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The Second Conference on Lunar Bases and Space Activities of the 21st Century

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The Second Conference on Lunar Bases and Space Activities of the 21st Century

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NASA Lyndon B. Johnson Space Center
Houston, Texas

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1924-1991

Advocates for lunar bases, professionals exploring the frontiers of astronomy, and enthusiasts for scientific investigation of the unknown all have lost a champion with the passing of Harlan J. Smith. As Director of McDonald Observatory for 26 years and as chairman of the Astronomy Department at the University of Texas at Austin for 15 years, Harlan's academic and scientific credentials are stellar. However, his inexhaustible energy and boundless enthusiasm for developing innovative approaches to astronomical research set him apart from the mainstream.

My only opportunity to work personally with Harlan came about during my studies of a permanent lunar base as a goal for the U.S. space program in the first decade of the twenty-first century. At the Johnson Space Center in the early 1980s, Mike Duke and I came to understand that any lunar base program needed a legitimate scientific component. (By the term "legitimate," we meant the program science could be advocated successfully on scientific grounds by scientists in scientific forums.) Mike formed an advisory group to help us think through the problems, and he asked Harlan to represent astronomy.

Harlan's reaction to the request was a healthy skepticism as to whether any large, manned space project could be seen by astronomers as a prudent investment in science. Nevertheless, he agreed to participate in order to ensure a knowledgeable representation of the views of astronomers.

Harlan worked with us at his usual high energy level and helped organize the Lunar Base Working Group, which met at Los Alamos in April 1984. The Report of the Working Group includes a thoughtful discussion of the advantages to making astronomical observations from the the lunar surface. In particular, the seismically stable lunar surface permits optical interferometry with microarcsecond angular resolution in the observational data. Bernie Burke developed the concept of the lunar optical interferometer.

As Harlan began to appreciate the unique qualities of the lunar environment for high-resolution, high-sensitivity optical observations and for wide-spectrum radio observations on the radio-quiet farside, he became not only an advocate but a champion of lunar-based astronomy.

Harlan was familiar with the need for persistence in advocating high-quality scientific projects. He helped organize a one-day workshop on lunar astronomy following the annual meeting of the American Astronomical Society in 1986. As I led off the meeting with a short talk on lunar base concepts, a young man in the front row asked, "If there is not going to be a lunar base for 20 years, why are we having this workshop now?" I turned to Harlan, who was sitting a few seats away, and asked when he had started talking about the Large Space Telescope (now called the Hubble). Harlan answered simply, "1962."

Harlan was ubiquitous and indefatigable in his advocacy. When he traveled to Moscow in late 1988, he wasted no time in bringing the Moon to Soviet scientists, most of whom had not considered a lunar base program. (In the Soviet Union, human exploration missions were discussed in the context of Mars landings within a paradigm established by Roald Sagdeev and his American colleague, Carl Sagan.)

In the spring of 1989, I met the Soviet planetary scientist Mikhail Marov and we discussed the relative merits of a manned lunar base and piloted missions to Mars as candidates for the next great step into space exploration. Mikhail was unfamiliar with the lunar base concepts although he knew Sagdeev's ideas well. When I saw Mikhail again at the International Space University session in Stras-

bourg in August 1989, he told me that he had written a "white paper" for the Soviet Academy of Sciences on a lunar strategy. It seems that after his conversation with me, he had spent two days with his dear friend, Harlan Smith, who had persuaded him of the logic of the lunar step.

When cancer manifested itself in his body, Harlan optimistically pursued experimental therapies, which proved to be more debilitating than he anticipated. Nevertheless, he traveled to scientific meetings and worked on new ideas whenever his energy level permitted it. He left us in the midst of spawning a concept to place an automated telescope on 20,000-foot Mt. Auconquilcha in the Andes to serve as a technology demonstration for a lunar instrument.

Those of us trying to support his efforts had trouble matching his schedule even in his last days. I hope to see that lunar telescope in action, and I hope to see it named after one of modern astronomy's staunchest proponents, Harlan J. Smith.

Wendell Mendell
Houston, Texas
November 20, 1991

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Prologue

Plans for manned bases on the Moon were first conducted by professional engineering organizations in the late 1950s as part of Project Horizon, a classified study sponsored by the U.S. Department of the Army. The civilian space program began to think about such things after President Kennedy's directive to land a man on the lunar surface by the end of the decade. Within a few years, NASA was working on concepts for extended human presence on the Moon under the Apollo Applications Program, as a continuation of the Apollo initiative. Journeys to Mars were also mapped out in the EMPIRE studies funded by NASA.

Detailed plans for lunar bases were simultaneously being developed in the Soviet Union as part of their secret manned lunar program. Although details of the Soviet N-1 rocket and their lunar transportation system have been released publicly, their lunar base plans have not yet been discussed.

In 1969-70, the Nixon Administration commissioned a study of the future of the space program under the stewardship of Vice-President Agnew. The report of the Space Task Group (STG) presented a sequence of manned programs, beginning with a low Earth orbit space station and continuing on to bases on the surface of the Moon and on Mars. The plan offered three different levels of effort with schedules dependent on funding commitments. In outline, the Space Task Group Report strongly resembles the current Space Exploration Initiative.

President Nixon and his staff decided that budgetary constraints would not permit commitment to a long-range program of human exploration of the solar system. In fact, they canceled the final two missions of the ongoing Apollo program. NASA conducted the Skylab program and the Apollo-Soyuz Test Project with spare Apollo hardware, but the long-range plans of the space agency were reduced to the development of the reusable space shuttle. In the STG Report, the Earth-to-orbit cargo vehicle, which later became the shuttle, had not received major emphasis. In NASA of the 1970s, it became the center of focus in the manned program and was to have major impacts on the unmanned program also.

The 1970s also saw a sequence of presidents who did not assign a high priority to ambitious goals for the space program. The NASA administrators reflected this philosophy and concentrated on being "team players" in restricted budgetary environments. The situation can be illustrated by considering the fiscal projections published in the STG Report for their three proposed approaches to expanding human presence in space. The graph that contained the funding estimates also featured a dotted line running along the bottom part of the chart. This dotted line was included to give the reader a reference point for the funding levels reflecting a hypothetical elimination of the manned space program. Looking back on the actual NASA funding (in constant year dollars) for the decade of the 1970s, we can see that it fell approximately 20% below the hypothetical dotted line.

Struggling to keep fiscal body and soul together, NASA invested few resources in true strategic planning. The last lunar base studies of the Apollo era have publication dates of 1972 or 1973. The agency looked seriously at solar power satellites as solutions to the energy crisis and dabbled in the space colony phenomenon, but generally the organizational mind-set embraced incremental programmatic evolution rather than bold landscapes with new initiatives. The phasing out of one major engineering development program (the space shuttle) and the start-up of another (the space station) occupied all the energy of the policy process within NASA of the early 1980s.

The space shuttle was operated by the Office of the National Space Transportation System (NSTS), a designation that could encompass other elements such as a space station and orbital transfer vehicles (OTV) for launching payloads from LEO to higher orbits. At the inauguration of the Reagan Administration, configurations for a LEO space station were being explored, and the performance requirements for a future OTV were being inserted into NASA databases.

This was the state of planning that I and my colleagues found in 1981 when we set off to explore the possibility of launching a Lunar Polar Orbiter (LPO) mission on the (then) new space shuttle. As NASA scientists involved in planetary exploration, we had little familiarity with manned programs. We were interested in resuming exploration of the Moon with implementation of the rather simple LPO spacecraft that had been "under consideration" for almost 10 years.

From our point of view, the NSTS, in its configuration circa the year 2000, appeared to provide routine access to the Moon. An OTV designed to deliver a communications satellite from the space station to geostationary orbit should be able to take satellites (or even landers) to the Moon because the ΔV (change in orbital velocity) required for both missions was essentially the same. As we pursued the matter further, we wondered whether consideration should be given to sizing the NSTS to carry humans and supporting cargo to the Moon for a lunar base.

Within NASA we encountered a number of reasons why a lunar base was a bad idea. A lunar base would be unaffordable or would compete with the space station; or Congress might scuttle the space station if it was believed to be a precursor for a lunar base. As we insisted on closer examination of an obvious extrapolation of the Space Transportation System, we became known as "lunar base advocates." We were told that advocacy of any particular objective was improper. The job of NASA planning was to produce a comprehensive list of all possible futures and study each to the same level of (superficial) detail as "options."

In 1983 we began a process to which I now refer as the legitimization of the lunar base discussion. We perceived the need to create forums wherein individuals and groups with accepted credentials could raise the relevant questions. Thereby the subject could become legitimate to evaluate within NASA. Critical steps in that process were the Report of the Lunar Base Working Group, from a workshop held at Los Alamos in April 1984, and the book *Lunar Bases and Space Activities of the 21st Century,* which recorded the proceedings of a symposium held at the National Academy of Sciences in Washington, DC, in October 1984. These meetings were conceived and organized by a small group of aerospace leaders from government, industry, and academia, who had been attracted to the lunar base as a long-term policy objective. Within NASA, funding was secured with the help of Deputy Administrator Hans Mark.

From that time forward, the planning environment evolved rapidly. A working group internal to NASA completed in 1985 a review of the technical constraints for manned Mars missions. The National Commission on Space delivered to the President in early 1986 a vision for the next 50 years. Astronaut Sally Ride led a NASA Task Group to produce the influential report, Leadership and America's Future in Space. NASA formed an Office of Exploration, staffed to the Administrator, in 1987. All these study groups relied heavily on technical information developed primarily at the Johnson Space Center a year or two earlier when lunar bases were not de rigeur. That work was performed on a tiny budget, but had the explicit support of Center Directors Chris Kraft and Aaron Cohen.The Second Symposium on Lunar Bases and Space Activities of the 21st Century was convened when interest in permanent presence on the Moon was growing rapidly. The Office of Exploration had become a funding source for new studies, replacing updated versions of older work. Internal funds in various organizations were being used to evaluate fresh ideas. New faces were appearing at aerospace meetings, particularly from the constructor-engineer companies, which possess unique and valuable expertise in building and operating facilities in harsh and isolated environments.

The current volume consists of a peer-reviewed selection of the papers delivered at the Second Symposium, held in Houston, Texas, on April 5-7, 1988. Compared to the 1984 symposium on lunar bases, these papers tend to go into more technical depth, reflecting a higher content of currently funded research. Participation from NASA is higher. Like the first symposium, the subject matter covers a broad range of topics, including discussions beyond the usual bounds of engineering and science. The selections are representative of the level of planning during the first year of operation of NASA's Office of Exploration.

During the preparation of this volume, many changes have occurred in what is now called the Space Exploration Initiative (SEI); and some of the assumptions in the papers here are dated. Legitimization of the lunar base concept has been completed with President Bush's sweeping vision of a return to the Moon ("... this time to stay ...") followed by piloted missions to Mars. At this writing, the fate of Space Station Freedom is uncertain, and progress in planning the SEI is awaiting the report from the Synthesis Group led by General Tom Stafford. The Report of the Advisory Committee on the Future of the Space Program is being cited and debated throughout the aerospace community.

Impatient enthusiasts supporting the human exploration of the solar system despair over the current turmoil. However, we must remind ourselves of the enormous progress that has been made in creating a real dialogue within the American body politic on the promise of the space frontier. This vision of the future must not be trivialized by identifying it with any single program or mission. Our movement to the planets must be made on a broad front with the active involvement and participation from many institutions in our society and from many of the peoples of the world. No longer is it sufficient to concentrate all space activities in one organization and expect all other constitutencies to support it. Yet fundamental change in the assignment of responsibility and authority is neither easy nor self-evident. We now are seeing the beginnings of change to a "new space order" that must be established before we can "boldly go where no one has gone before."

Wendell Mendell
Houston, Texas
May 23, 1991

Acknowledgements

Providing peer review for a collection of professional papers is always a demanding task for an editor. He must find associate editors—responsible and knowledgeable volunteers each of whom is willing to take charge of six to eight manuscripts and solicit qualified referees for them. In an eclectic volume such as this one, the editorial board must span a variety of disciplines and professional communities. Every community has its own culture and standards for judging worth, and these value systems must be integrated to provide a uniform quality to the finished collection.

For this volume, I was forced to go to 17 associate editors in order to provide reasonable workloads for each one, and I am grateful to them all; but I feel particularly obligated to Gordon Woodcock, Larry Taylor, and John Alred, all of whom jumped in at crucial times.

I thank Stephanie Tindell, Sarah Enticknap, Ronna Hurd, and other members of the Publications Services Department at the Lunar and Planetary Institute for exhibiting inexhaustible patience throughout the production phase. Bill Lagle from the Lockheed Engineering Services Company was invariably eager to do whatever was asked, and I should have taken advantage of his talents more often. Mark Cintala and Sarah Enticknap deserve special acknowledgment for creating the subject index.

Finally, I must acknowledge the efforts of Mike Duke and Barney Roberts in holding the original symposium and in helping to publish this book.

Wendell Mendell
Houston, Texas
February 7, 1992

1 / Lunar Transportation Systems

CONCEPTUAL ANALYSIS OF A LUNAR BASE TRANSPORTATION SYSTEM

N93-17415

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INTRODUCTION

The Report of the National Commission on Space (*National Commission on Space,* 1986) and the NASA/National Academy of Science Symposium on Lunar Bases and Space Activities of the 21st Century (*Mendell,* 1985) demonstrated that a return to the Moon would be a logical and feasible extension of NASA's goal to expand the human presence in space. Development of a permanently manned lunar base would provide an outpost for scientific research, economic exploitation of the Moon's resources, and the eventual colonization of the Moon.

Important to the planning for such a lunar base is the development of transportation requirements for the establishment and maintenance of that base. This was accomplished as part of a lunar base systems assessment study conducted by the NASA Langley Research Center in conjunction with the NASA Johnson Space Center. Lunar base parameters are presented using a baseline lunar facility concept and timeline of developmental phases. Masses for habitation and scientific modules, power systems, life support systems, and thermal control systems were generated, assuming space station technology as a starting point. The masses were manifested by grouping various systems into cargo missions and interspersing manned flights consistent with construction and base maintenance timelines.

A computer program that sizes the orbital transfer vehicles (OTVs), lunar landers, lunar ascenders, and the manned capsules was developed. This program consists of an iterative technique to solve the rocket equation successively for each velocity correction (ΔV) in a mission. The ΔV values reflect integrated trajectory values and include gravity losses. As the program computed fuel masses, it matched structural masses from General Dynamics' modular space-based OTV design (*Ketchum,* 1986a).

Variables in the study included the operational mode (i.e., expendable vs. reusable and single-stage vs. two-stage OTVs), cryogenic specific impulse, reflecting different levels of engine technology, and aerobraking vs. all-propulsive return to Earth orbit. The use of lunar-derived oxygen was also examined for its general impact. For each combination of factors, the low-Earth-orbit (LEO) stack masses and Earth-to-orbit (ETO) lift requirements are summarized by individual mission and totaled for the developmental phase. In addition to these discrete data, trends in the variation of study parameters are presented.

METHODOLOGY

The methodology for the lunar base transportation study is shown in Fig. 1. Requirements for the baseline lunar base mission model, derived by NASA Johnson Space Center, produced a set of functional requirements for the lunar base that included habitability, manufacturing, commercial applications, science, and exploration. System concepts were developed and analysis and technology option trade studies were conducted to define the mass, volume, power, and resupply requirements of the lunar base system. A manifest was prepared based on the priority requirements of equipment and hardware for the lunar base and the

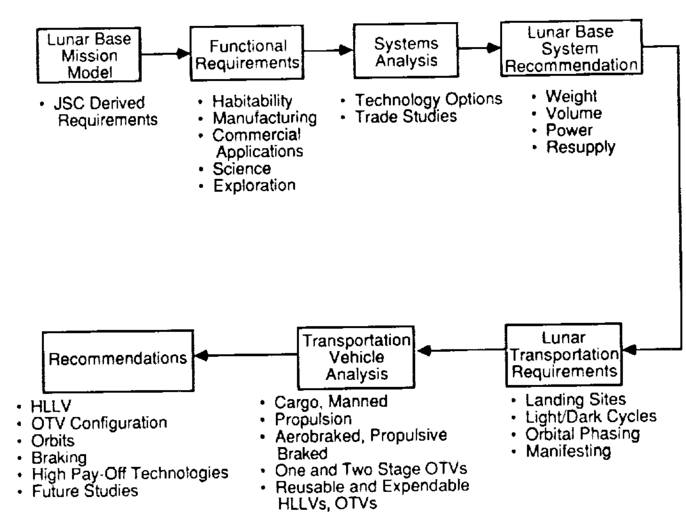

Fig. 1. Lunar base studies methodology.

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volume and mass requirements of the transportation system. The manifest information was then input into the analysis of transportation vehicle options. This analysis considered such factors as (1) separate manned and cargo missions; (2) reusable vs. expendable OTVs; (3) one- vs. two-stage OTVs; (4) aerobraking vs. propulsive braking on return to LEO; (5) specific impulse of cryogenic engines; and (6) impact of using lunar-derived oxygen in lunar vicinity.

MISSION DESCRIPTION

Development of a lunar base will probably progress in steps and phases as shown in Table 1 (*Roberts,* 1986). The first phase will incorporate unmanned reconnaissance or global mapping missions to expand the scientific database of the Moon (including lunar resource research). In the Phase II scenario, a temporary manned facility would be established on the lunar surface to provide limited research capability for science, materials processing, and lunar surface operations. Follow-on phases would establish permanent occupancy and self-sufficient bases, leading to colonization of the Moon. This study addresses the transportation requirements and system for the Phase II temporary facility.

The Phase II lunar base required a total mass of 207,865 lbm delivered to the lunar surface. A breakdown of the facility and equipment masses is given in Table 2. Manifesting the lunar base

TABLE 1. Lunar base phases.

Phase	Mission	Time Period	Crew Size	Power (kW)	Function	Facilities
I	Lunar surface mapping	1995-2000	—	—	• Preliminary site selection	• Unmanned lunar orbiter satellite
II	Lunar sorties to establish a small space port	2000-2008	0-5	100	• Final site selection • Site preparation • Exploration to 10 km • Core samples to 5 m • Materials processing	• Habitability module • Soil mover/crane • Pilot LOX plant • Core sampler • Surface transporter
III	Expand space port to increase functional capabilities	2008-2018	5-11	300	• Permanently manned • Expanded crew • Materials research • Closed loop research • LOX utilization	• 2 habitability modules • Science/astronomy • Expanded LOX plant
IV	Establish lunar base with minimum support from Earth for survival	2018-2028	11-30	1000	• Full LOX production • Habitat growth • Locally derived products/consumables	• 6 habitability modules • 2 science/astronomy modules • 1000 metric ton per year LOX plant • Closed ECLSS • LOX storage and servicing modules

TABLE 2. Lunar base facility and equipment masses.

	Lunar Base		90-day Resupply		
Facility	Mass (lbm)	Volume (ft^3)	Mass (lbm)	Volume (ft^3)	Power (kW)
Habitation Module 1	36,108	6,532	5,162	289	4.72
Node 1	16,983	2,860	325	32	4.68
Node 2	16,972	2,860	695	40	3.35
Node 3 LOX	17,627	2,860	226	35	73.41
Air Lock 1	5,879	1,006	70	7	1.16
Air Lock 2	5,879	1,006	68	7	1.16
Air Lock 3	5,671	1,006	40	5	0.99
Transporter 1	4,469	2,219	195	110	0
Crane/Regolith Mover 1	14,239	4,269	620	210	0
Launch/Lander Pad 1	27,600	15,150	50	2	0.05
Maintenance Shed 1	8,090	3,500	46	1	1.00
External Equipment	48,348	3,576	2,854	207	117.00
Total	207,865	46,844	10,351	945	207.50

material/components resulted in a requirement for 16 missions, 9 manned and 7 unmanned. A sample manifest for missions 1 and 2 (a manned and cargo mission) is presented in Table 3. The lunar base masses and manifest were developed in the NASA Langley assessment study from the NASA Johnson requirements.

To establish the Phase II lunar base, a transportation system capable of transporting manned capsules with a mass of about 13,000 lbm to and from the lunar surface and ferrying a cargo of 35,000 to 40,000 lbm to the lunar surface is required. For this study, the total mass (including payloads, modules, fuel, and crew) to be delivered to Earth orbit is approximately 3.0 million lbm to 4.5 million lbm, depending on the operational mode, engine efficiency, and reentry braking system.

TRANSPORTATION SYSTEM DESCRIPTION AND WEIGHT SUMMARY

The transportation system required for buildup and maintenance of a lunar base assumed Earth launch of a heavy-lift launch vehicle (HLLV) to a staging area (space station) in LEO and OTVs for transfer of all material to the Moon. The HLLV is capable of delivering approximately 150,000 lbm into LEO.

The space-based OTV concept that was used as the baseline for this study is the General Dynamics S-4C modular tank concept (*Ketchum,* 1986b). Figure 2 shows line drawings of the one-stage manned (with lunar ascent and descent vehicle) and two-stage cargo (with hab module payload) configurations.

TABLE 3. Sample mission manifest.

Mission 1 (manned)		Mission 2 (unmanned)	
Manned capsule	13,200 lbm	Regolith mover/crane	14,239 lbm
Core sampler	40 lbm	50% external power equipment	11,601 lbm
Stay time extension module (18-day supply)	3,300 lbm	Maintenance shelter	8,069 lbm
Lunar rover	4,469 lbm		
Crew and supplies	1,500 lbm		
Subtotal	22,509 lbm	Subtotal	33,390 lbm
Package (10%)	2,251 lbm	Package (10%)	3,393 lbm
Total mass approx.	24,800 lbm	Total mass approx.	36,800 lbm

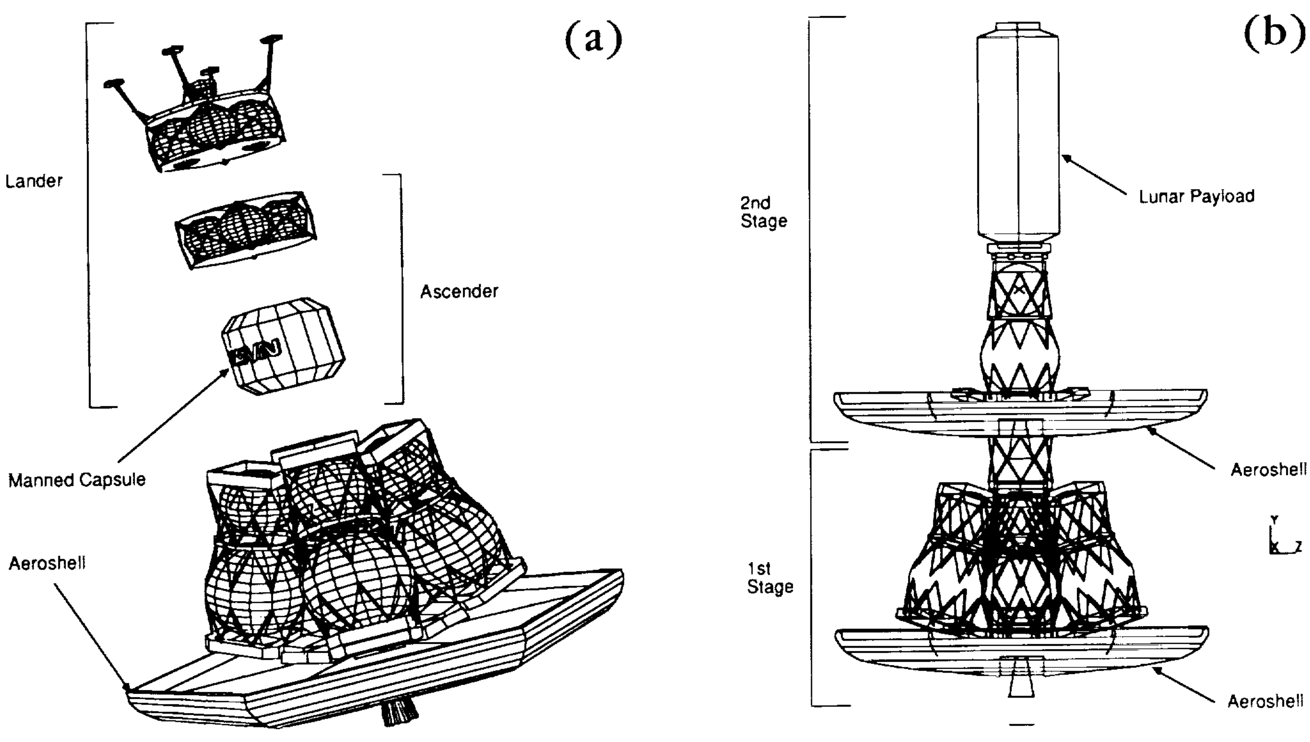


Fig. 2. Orbital transfer vehicle (OTV) line drawings: **(a)** one-stage manned and **(b)** two-stage cargo.

The S-4C OTV is composed of the following components: (1) twin engines; (2) geotruss aerobrake; (3) propellant tank sets (hydrogen and oxygen); (4) avionics package; and (5) payload.

In order to accommodate different payloads (masses), up to seven propellant tank sets can be accommodated on a single stage. The propellant capacity and the associated mass breakdown of the OTV for practicable numbers of tank sets are given in Table 4.

LUNAR MISSIONS TRANSPORTATION MODE SCENARIOS

Transportation mode scenarios for one-stage and two-stage lunar missions are shown in Fig. 3. Both manned and cargo, as well as expendable and reusable, missions are presented.

The mission scenario begins with the lunar transportation system (one- or two-stage) in LEO. For the manned missions, the transportation system consists of the OTV, a manned capsule, a lunar lander, and a lunar ascender. The cargo mission transportation system consists only of the OTV, the lunar lander, and the lunar payload. The OTV performs the translunar injection (TLI) burn and the lunar orbit insertion (LOI) burn. The OTV is discarded in lunar orbit, and the descender is discarded on the lunar surface. For the manned missions, the lunar ascender returns the manned capsule to lunar orbit to rendezvous with the OTV and is discarded. The OTV for all return missions (all manned and the reusable cargo missions) performs a trans-Earth injection (TEI) burn. Earth orbit insertion (EOI) is performed either propulsively or by aerobraking in the upper atmosphere along with a small ΔV burn. Once in LEO, the OTV and manned capsule will be refitted for reuse (reusable missions). For the expendable missions, a new OTV must be delivered by the HLLV for follow-on missions.

In the case of the two-stage OTV in Figs. 3c,d, stage one separates after TLI and is either discarded (expendable) or performs an Earth-orbit aerobraking in the upper atmosphere, along with a small ΔV burn to rendezvous with the space station for subsequent reuse. The second stage performs the LOI, and the OTV remains in lunar orbit while the lunar lander performs a powered descent carrying the payload (manned or cargo) to the lunar surface. For the expendable cargo missions, the lunar lander is discarded on the lunar surface and the OTV is discarded in lunar orbit.

COMPUTER PROGRAM DESCRIPTION

A FORTRAN program based on an iterative solution to the rocket equation was written to solve for the mass required to be delivered to LEO. The general form of the rocket equation is

$$\Delta V = g_e I_{sp} \, ln(M_o/M_f) \tag{1}$$

where ΔV is the change in velocity required for a specific maneuver (ft/sec), g_e is Earth gravity (32.174 ft/sec^2), I_{sp} is the specific impulse of the fuel (sec), M_o is the initial mass before the maneuver (lbm), and M_f is the final mass after the manuever (lbm).

Solving for the mass of fuel required for each manuever, the rocket equation takes the form of

$$M_{fuel} = M_f(e^{\Delta V/g_e I_{sp}} - 1) \tag{2}$$

where M_{fuel} is the mass of fuel required for the maneuver (lbm).

TABLE 4. Vehicle mass summary.

	Number of Fuel Tank Sets				
	1	3	4	5	7
Structure	2,732	3,514	3,905	4,296	5,078
Tanks	292	1,381	1,926	2,470	3,559
Propulsion system	1,178	1,828	2,153	2,478	3,128
Thermal control system	125	261	329	397	533
GN&C	150	150	150	150	150
Electrical systems	555	555	555	555	555
Aerobrake (reusable)	1,341	2,298	2,298	2,298	2,298
Propellant	40,843	122,529	163,372	204,215	285,901
Residual propellant	529	1,526	1,995	2,463	3,401
Pressurant	9	27	36	45	63
Reusable OTV					
Dry mass	6,374	9,987	11,316	12,644	15,301
Wet mass	47,217	132,516	174,688	216,859	301,202
Mass after maneuver	6,912	11,540	13,347	15,152	18,765
Expendable OTV					
Dry mass	5,033	7,689	9,018	10,646	13,003
Wet mass	45,217	130,218	172,390	214,561	298,904
Mass after maneuver	5,571	9,242	11,049	12,854	16,467

	Lunar lander	Lunar ascender
Structure	8,360	5,720
Propellant	29,920	11,000

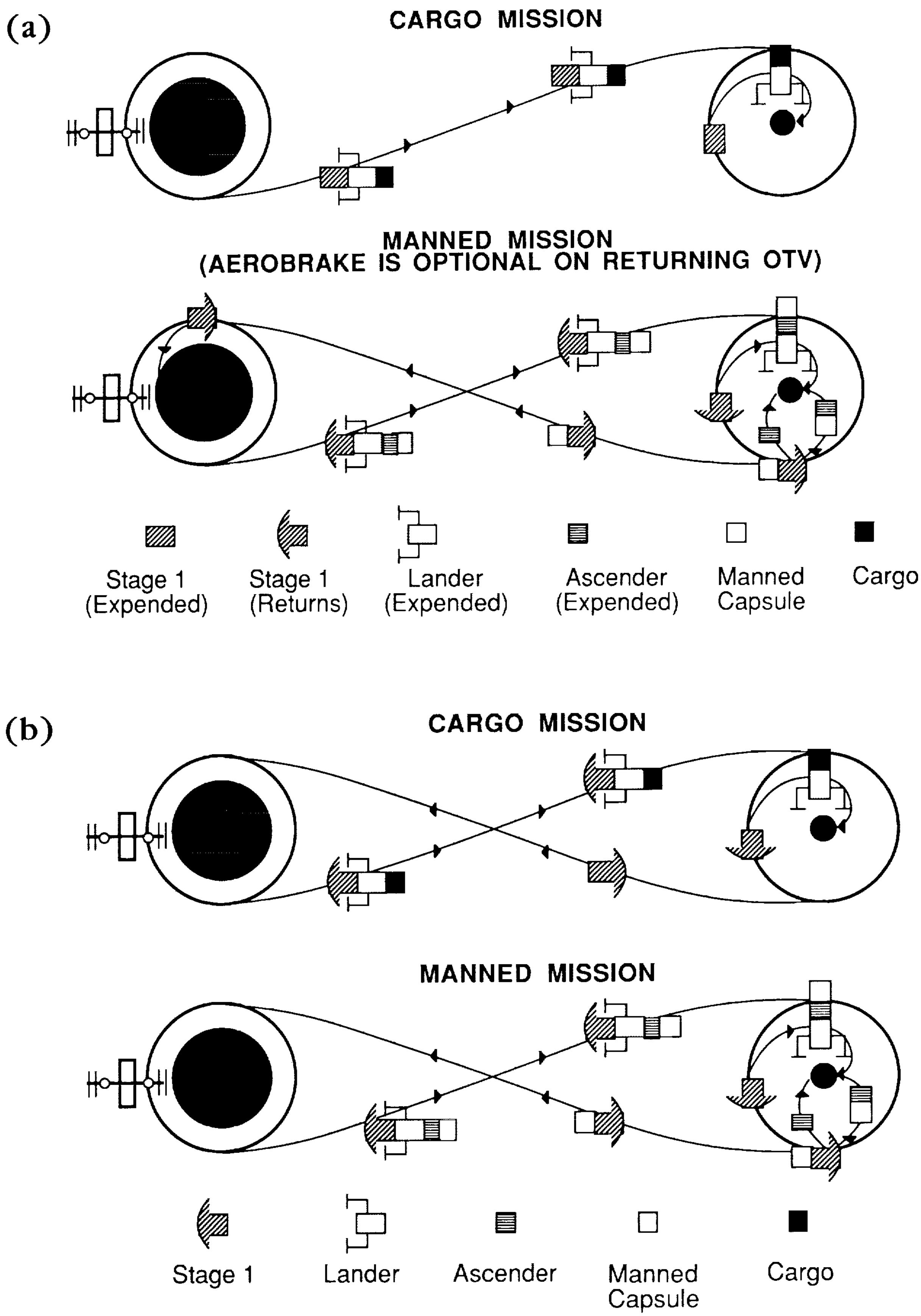
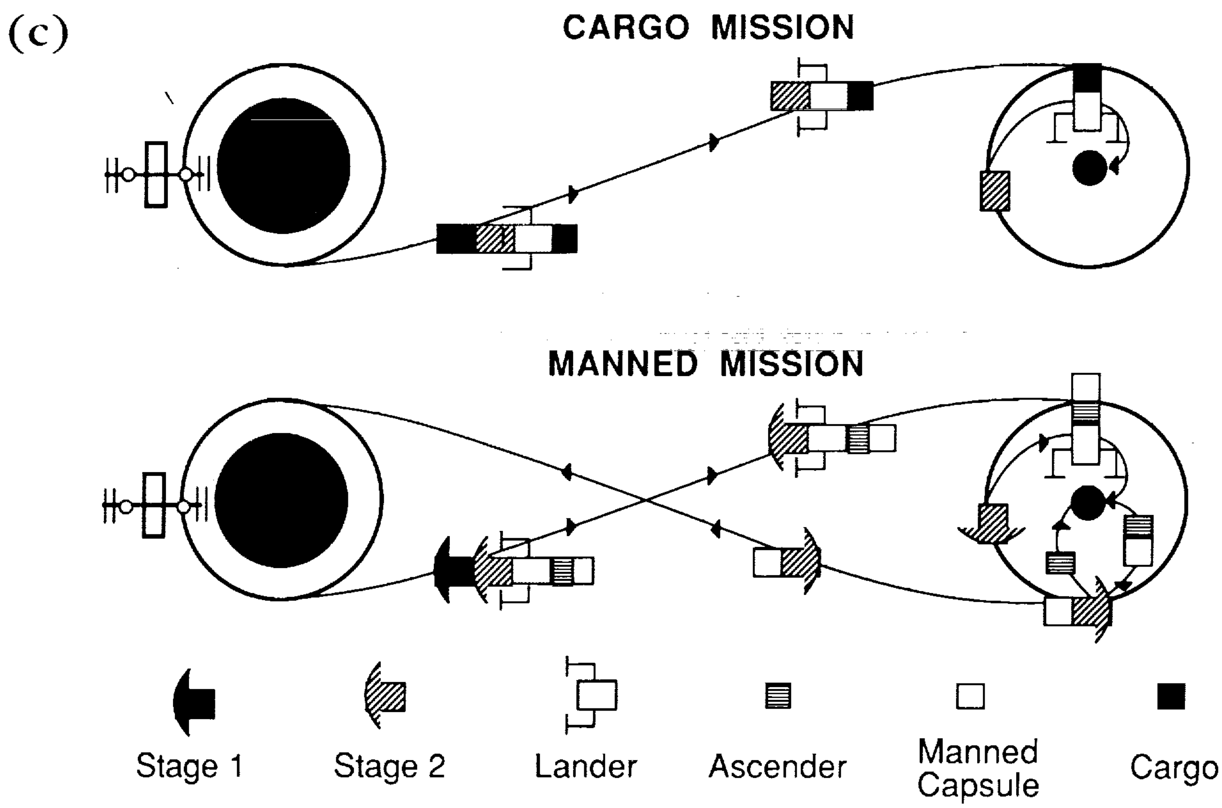


Fig. 3. Lunar mission scenarios: **(a)** one-stage, expendable OTV; **(b)** one-stage, reusable OTV.



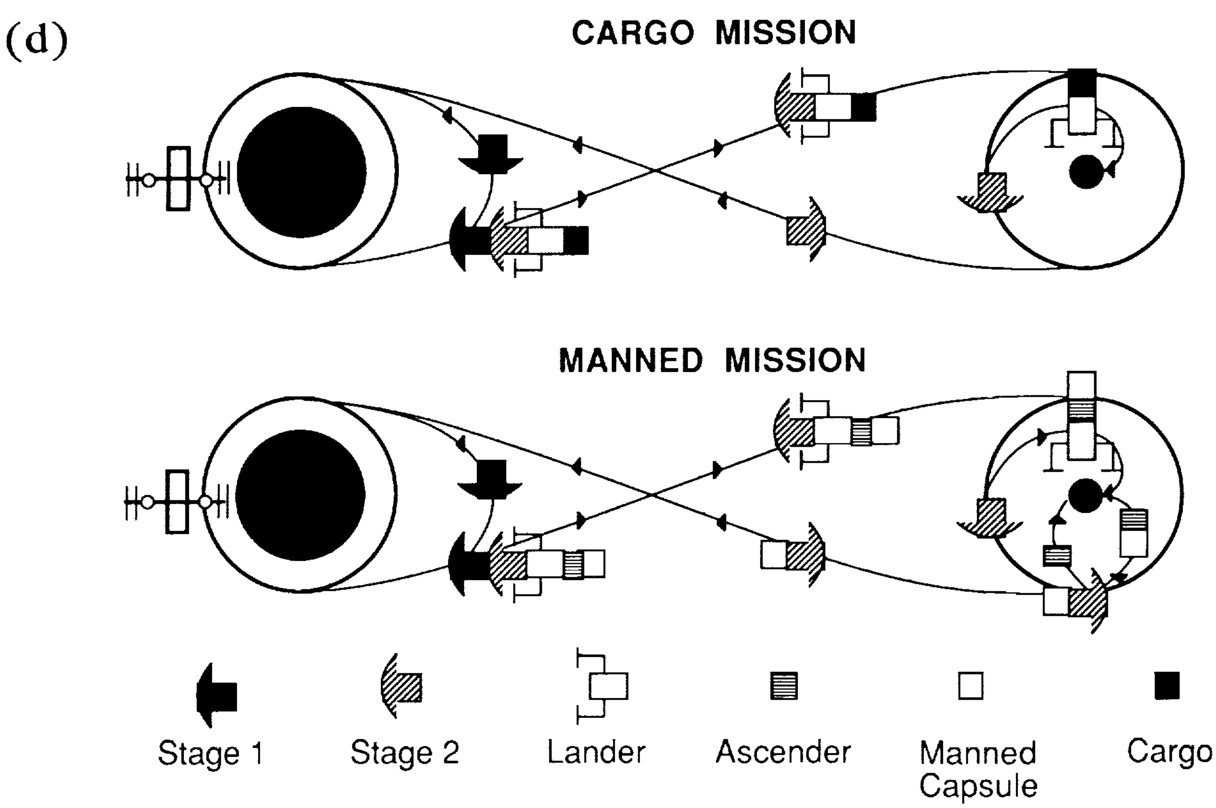


Fig. 3. (continued) **(c)** Two-stage, expendable OTV; **(d)** two-stage, reusable OTV.

The ΔV values shown in Fig. 4 are comparable to actual flight values from Apollo. The program starts with the manned module's ascent from the lunar surface and iterates backward from the lunar surface to determine the mass that must be delivered to LEO for the mission. This mass is the sum of the structure and fuel masses for all maneuvers plus the mass of the lunar payload (personnel, cargo, and supplies).

ETO MASS SUMMARY

The ETO masses were determined for all 16 missions in each transportation scenario. For manned missions, the initial delivery of the reusable manned capsule was not considered in the ETO mass. Also, the initial delivery of the OTV was not considered in the ETO mass for reusable missions. A sample 16-mission ETO mass summary for a one-stage, reusable, aerobraked OTV with a specific impulse of 460 sec is shown in Table 5.

Tables 6 and 7 provide the total mass to be delivered to LEO for the 16-mission lunar base buildup and the number of HLLV launches required for each scenario. Twelve scenarios covering all the trade-off options are shown. Mass to LEO varied from 3.03 million lbm to 4.91 million lbm, and the number of HLLV launches varied from 20 to 33. These total mission numbers and the ETO vs. lunar payload mass trend charts (to be discussed in the next section) were used to define the optimum lunar base transportation system.

TRADE-OFFS

A series of trade-off studies were conducted on key design parameters to determine the optimum transportation system for the manned and the cargo missions. Parameters affecting the design of the transportation system included (1) manned vs. cargo (unmanned); (2) reusable vs. expendable OTV; (3) one- vs. two-stage OTV; (4) aerobraking vs. propulsive braking on return to LEO; and (5) specific impulse of the cryogenic engines. Because of the large number of charts involved using the nine different variables, only sample trend charts for each set of variables are presented.

Trend charts of ETO mass required for varying manned capsule and lunar payload masses are presented in Figs. 5 to 9. Note that the step increases in ETO masses in the figures are due to the modular design of the OTV. As the deliverable lunar payload mass increases, the propellant requirement increases. When the propellant requirement exceeds the capability of the propellant tank set in the design, the computer program increases the number of tank sets to accommodate the new requirement, which, in turn, increases the structural mass of the OTV by a discrete amount.

Reusable vs. Expendable

The question of employing reusable as opposed to expendable OTV systems is very complex. Not only is the added mass (fuel) needed to transport and return the system to LEO a consideration,

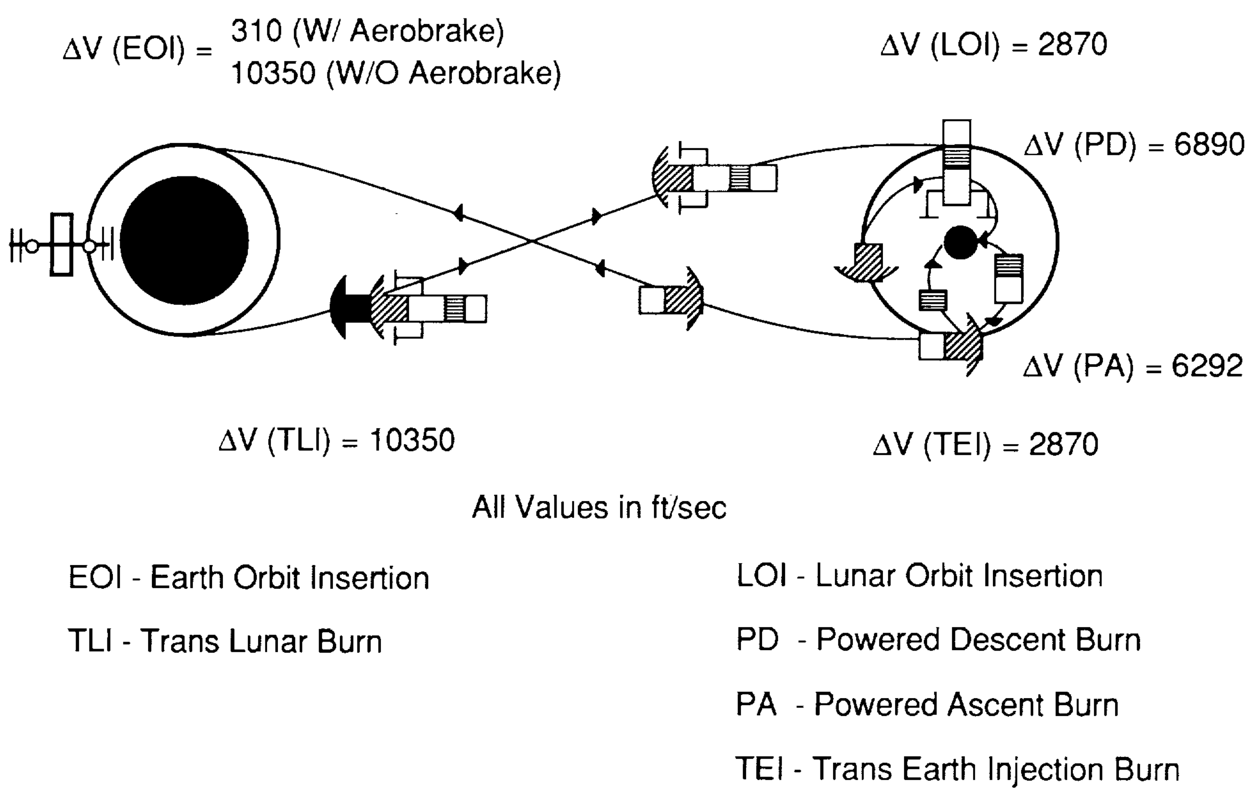


Fig. 4. Propulsive ΔV summary.

TABLE 5. Sample ETO mass summary.

Mission #	Mass (in lbm)	Mission #	Mass (in lbm)
1	260,595	9	228,525
2	202,814	10	228,525
3	228,525	11	228,525
4	204,370	12	143,111
5	228,525	13	217,084
6	214,488	14	196,587
7	154,007	15	217,084
8	154,007	16	217,084

One-stage, reusable OTV, aerobrake used; I_{sp} = 460 sec. Total mass to low Earth orbit = 3,323,856 lbm. Requires 22 launches of an HLLV (150,000 lbm payload capability).

TABLE 6. Transportation summary for a one-stage OTV.

I_{sp} (sec)	Mission Designation	Case No.	First Stage Aero	First Stage Nonaero	Weight to LEO, lbm × 10^6	No. of HLLV Launches Req'd. (150 k lbm)
440	Reusable	1	X		3.61	24
	Expendable	2	X*		3.74	25
	Reusable	3		X	4.83	32
	Expendable	4		X*	4.70	32
460	Reusable	5	X		3.32	22
	Expendable	6	X*		3.43	23
	Reusable	7		X	4.41	30
	Expendable	8		X*	4.33	29
485	Reusable	9	X		3.03	20
	Expendable	10	X*		3.16	21
	Reusable	11		X	3.98	27
	Expendable	12		X*	3.96	27

*For manned missions, stage 1 returns to LEO; for cargo missions, stage 1 is expended.

Phase II (16 missions: 9 manned, 7 unmanned).

TABLE 7. Transportation summary for a two-stage OTV.

I_{sp} (sec)	Mission Designation	Case No.	First Stage Aero	First Stage Nonaero	Second Stage Aero	Second Stage Nonaero	Weight to LEO, lbm × 10^6	No. of HLLV Launches Req'd. (150 k lbm)
440	Reusable	1	X		X		3.57	24
	Expendable	2			X*		3.75	25
	Reusable	3		X		X	4.91	33
	Expendable	4				X*	4.57	31
460	Reusable	5	X		X		3.32	22
	Expendable	6			X*		3.49	23
	Reusable	7		X		X	4.44	30
	Expendable	8				X*	4.22	28
485	Reusable	9	X		X		3.03	20
	Expendable	10			X*		3.21	22
	Reusable	11		X		X	4.02	27
	Expendable	12				X*	3.85	26

*For manned missions, stage 2 returns to LEO; for cargo missions, stage 2 is expended.

Phase II (16 missions: 9 manned, 7 unmanned).

but the structural and developmental cost of the reusable system, as well as the replacement cost of expendable systems for resupply and follow-on missions, must also be considered. An accurate cost comparison of these two types of vehicles is beyond the scope of this study. This study was concerned only with the ETO masses involved and did not consider any cost factors. The developmental cost of a reusable system could possibly offset its operating cost advantage over an expendable system.

Calculation of the total ETO mass for the reusable and expendable missions considered the added fuel to return the reusable system to Earth orbit for refit, whereas the expendable missions required a completely new OTV structure for each mission. Comparison of the ETO mass vs. lunar payload mass for both manned and cargo missions in the reusable and expendable configurations is shown in Fig. 5. The ETO mass of the reusable vehicle is consistently lower than that of the expendable vehicle

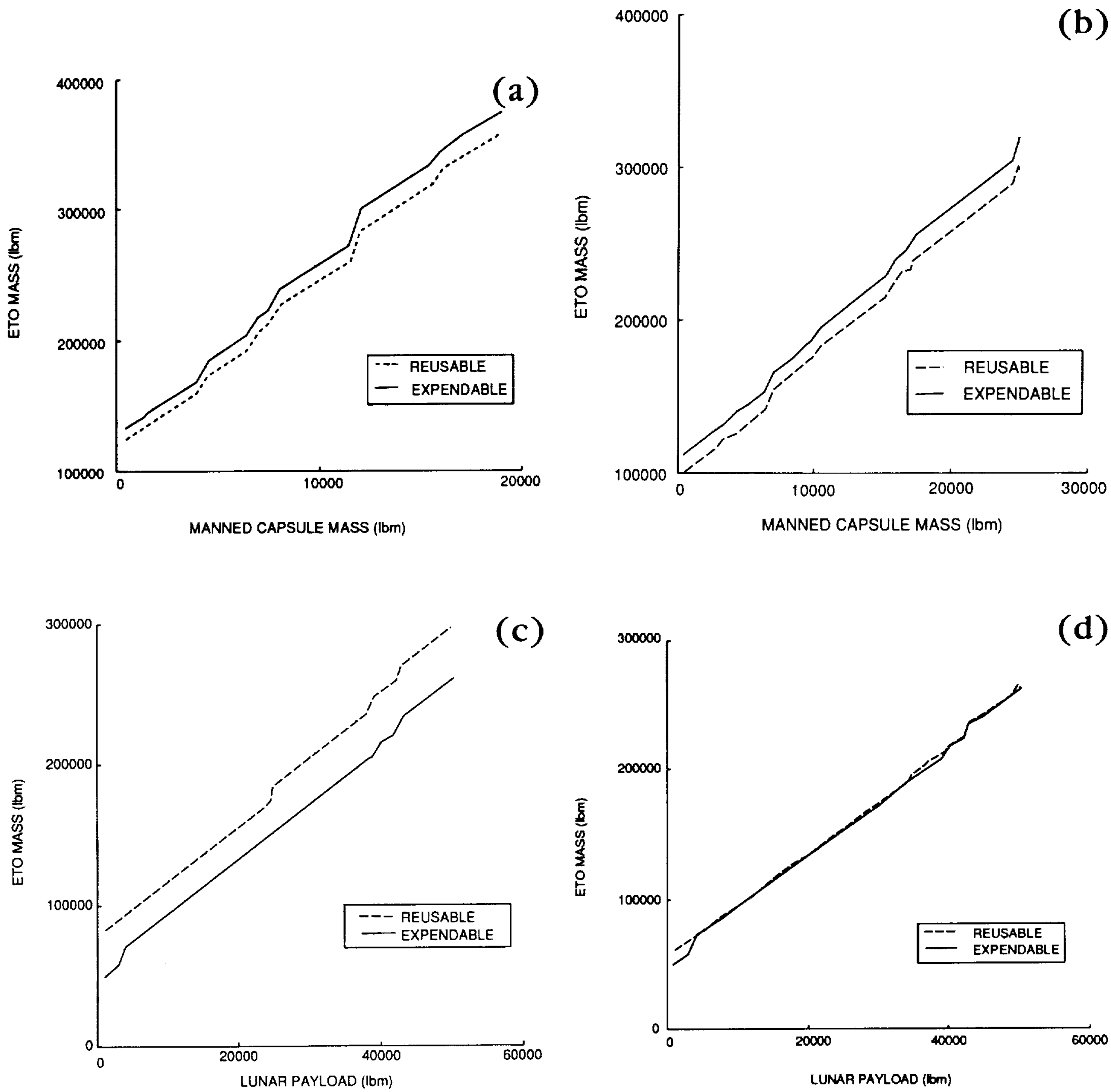


Fig. 5. ETO mass comparison of reusable and expendable OTVs: **(a)** one-stage, nonaerobraked, manned; **(b)** one-stage, aerobraked, manned; **(c)** one-stage, nonaerobraked, cargo; and **(d)** one-stage, aerobraked, cargo.

for the manned missions. The ETO mass for the reusable aerobraked cargo mission (Fig. 5d) is higher than that of the expendable mission. This is due to the large quantity of fuel required to return the reusable aerobraked cargo OTV to Earth orbit.

Over the 16-mission buildup of the lunar base, a saving of one HLLV ETO flight is achieved using aerobraking and reusable instead of expendable systems, regardless of staging (Tables 6 and 7). Without aerobraking, the expendable system is equal to or less costly (in terms of HLLV launches) than the reusable system, even though a new OTV is required for each mission.

One vs. Two Stages

The trend in ETO mass vs. manned capsule mass is almost identical for the one-stage and two-stage systems (Fig. 6). The same trend was noted in the cargo missions. This becomes more obvious when the total number of HLLV launches for the Phase II buildup is considered (Tables 6 and 7). In only three scenarios did the total mass to LEO using one vs. two stages vary by more than 80,000 lbm, thereby requiring one less HLLV for the two-stage missions. Each of these three scenarios involved expendable, nonaerobraked missions. Logistically, then, it is not necessary to consider a two-stage system in the lunar base transportation scenario. (Note that these results differ from the classical one-stage vs. two-stage comparison. In this study, the expended propulsive stages were not discarded; however, as indicated in Table 6, the one-stage OTV returns to LEO for manned missions and, for the two-stage manned OTV case, stage 2 returns to LEO. These returning stages require the addition of aerobrakes and other recapture components, thereby complicating the classical staging trade.)

Aerobraking vs. Propulsive Braking

The trends for both manned and cargo aerobraked vs. propulsive-braked systems are shown in Fig. 7. Using aerobraking for the cargo missions means a saving of 20,000 lbm to 30,000 lbm. The manned missions show a more drastic decrease in ETO mass with aerobraking. Here, the savings vary from 30,000 lbm for a 5000-lbm manned capsule to 100,000 lbm for a 20,000-lbm manned capsule. This translates into a savings of 8

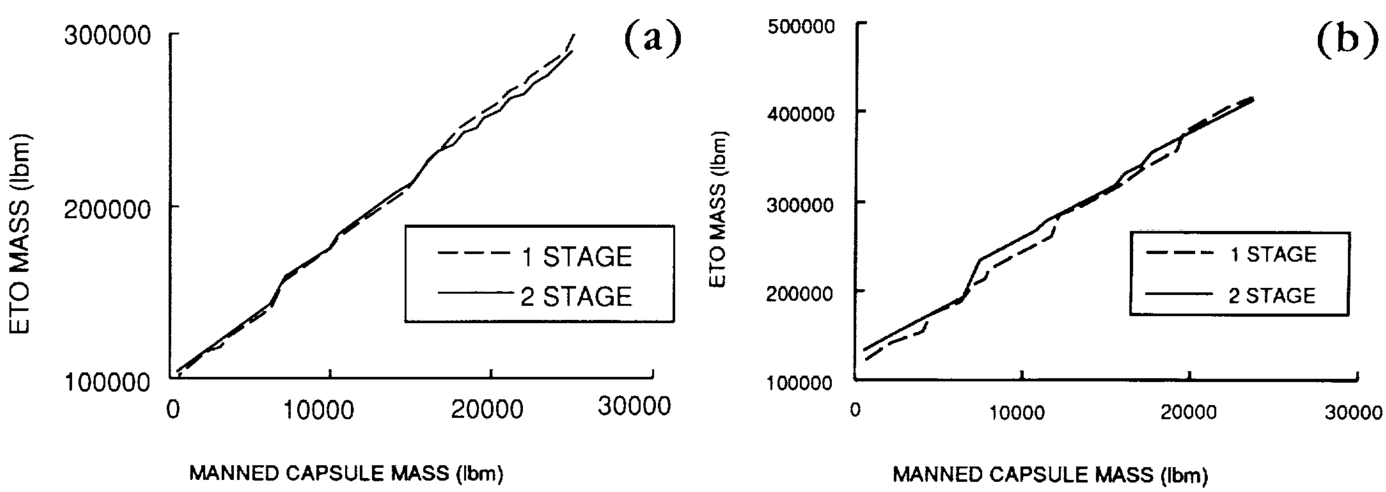


Fig. 6. ETO mass comparison of one-stage and two-stage OTVs: **(a)** reusable, manned, aerobraked and **(b)** reusable, manned nonaerobraked.

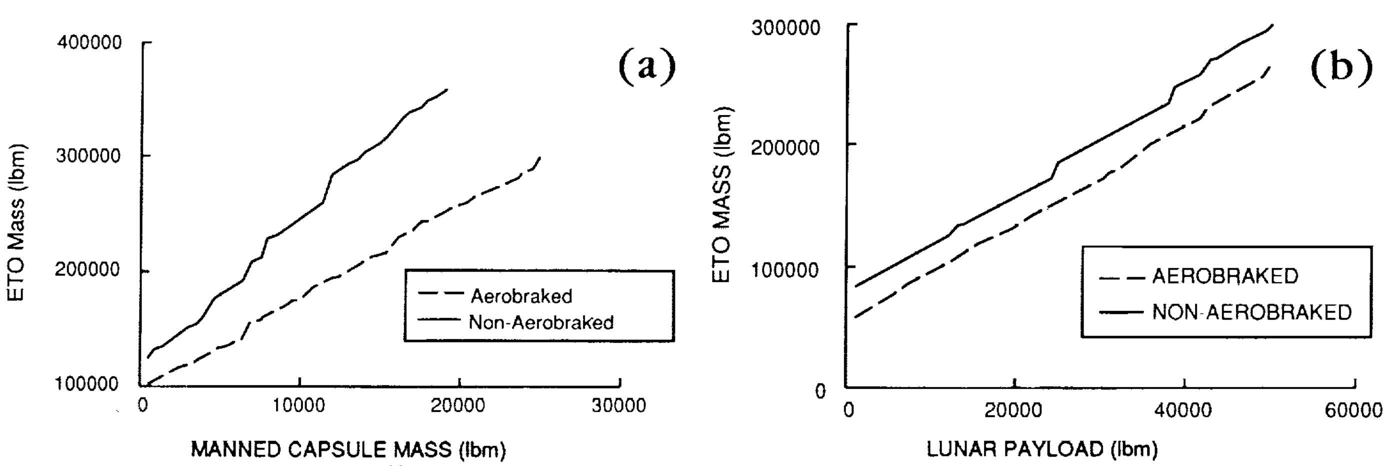


Fig. 7. ETO mass comparison of aerobraking and nonaerobraking (I_{sp} = 460 sec): **(a)** reusable, manned, one-stage and **(b)** reusable, cargo, one-stage.

HLLV launches over the 16-mission buildup of the lunar base (Tables 6 and 7). The savings in HLLV launches (ETO mass) when using the aerobraked system is due to the reduced amount of fuel necessary for Earth-orbit insertion. The much larger savings in mass in the manned mission case results from the larger mass that is being returned to low Earth orbit. The development and use of an aerobraking system becomes a distinct enhancing technology for lunar base missions.

Specific Impulse of the Cryogenic Engine

The trade study concerning the effect of varying specific impulse assumed only engines using cryogenic propellants, liquid oxygen, and liquid hydrogen. Three I_{sp} values (440, 460, and 485 sec) were considered, relative to state-of-the-art engine technology. An I_{sp} of 440 sec corresponds to current RL-10 engine technology, 460 sec considers a modified RL-10 engine using a large expansion ratio, and 485 sec corresponds to an engine based on advanced technology.

Trends in the I_{sp} effect on ETO mass are presented in Fig. 8. As expected, in all cases the higher the I_{sp}, the lower the ETO mass for a given manned capsule or lunar payload mass. The effect of the aerobrake in reducing the number of HLLV launches for the 16 missions is less dramatic for higher I_{sp} values. For a reusable OTV with an I_{sp} of 440 sec, use of the aerobrake saves eight or nine HLLV launches, while the same OTV with a 485-sec I_{sp} saves only seven HLLV launches (Tables 6 and 7).

LUNAR LOX IMPACT

The lunar surface is rich in minerals from which oxygen can be derived. *Roberts* (1986) showed that a transportation system using lunar-derived oxygen offers substantial ETO mass savings over a totally Earth-based system. For the present study, the use of lunar oxygen was only considered for lunar descent and ascent, trans-Earth injection, and Earth circularization maneuvers of reusable missions. Comparisons of ETO masses for variations in lunar payload mass for reusable cargo and manned missions are shown in Figs. 9a-d.

For a reusable cargo mission (one stage with an I_{sp} of 460 sec) with a 30,000-lbm lunar payload (Figs. 9a,b), the ETO mass for the nonaerobraked transportation system using Earth-derived LOX is 3.3 times that of the lunar-derived LOX system (204,000 lbm vs. 62,000 lbm). The addition of aerobraking reduces the ETO mass to 172,000 lbm for the Earth-derived LOX system with no appreciable change in the lunar-derived system ETO mass (the Earth-derived LOX system is still a factor of 2.8 higher).

The effect of using lunar-derived LOX is even more dramatic for the manned missions (Figs. 9c,d). Assuming a 19,000-lbm manned module (one-stage system with an I_{sp} of 460 sec), the ETO mass is 100,000 lbm for a lunar-derived LOX nonaerobraked transportation system as opposed to 355,000 lbm (a factor of 3.5 higher) for an Earth-derived LOX system. With aerobraking, the same manned capsule requires an ETO mass of 88,000 lbm for a lunar-derived LOX system and an ETO mass of 266,000 lbm for an Earth-derived system (3 times higher than the lunar-derived system).

With lunar LOX, the ETO mass of cargo missions can be reduced to 25-50% of that required with Earth-derived LOX. For manned missions using lunar LOX, the ETO mass can be reduced to 16-25%. For the 16-mission buildup, the total ETO mass can be reduced from 3.32 million lbm to 1.10 million lbm with the use of lunar-derived LOX (Fig. 10). Those mass savings are due

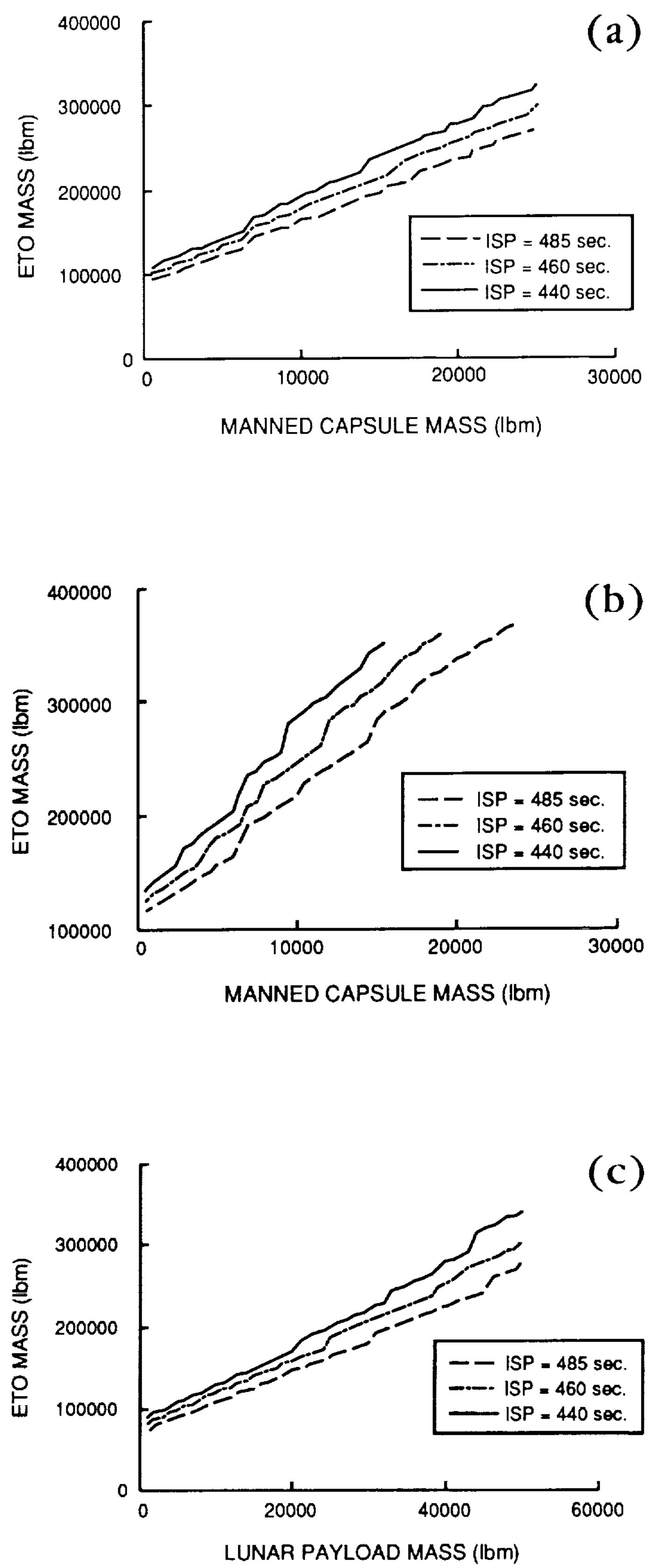


Fig. 8. ETO mass comparison of effect of specific impulse (I_{sp}): **(a)** reusable, manned, one-stage, aerobraked; **(b)** reusable, manned, one-stage, nonaerobraked; and **(c)** reusable, cargo, one-stage, aerobraked.

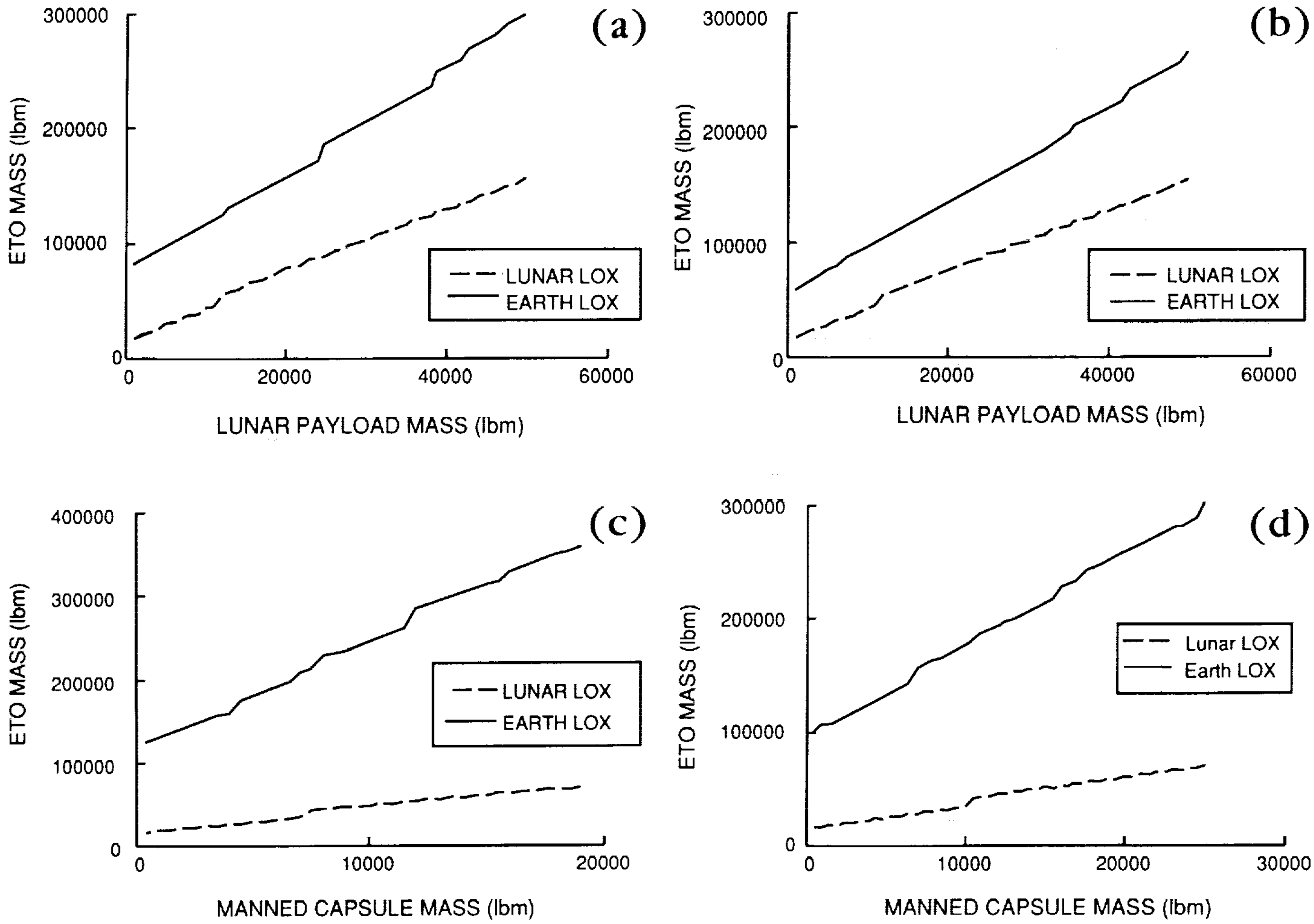


Fig. 9. Impact of lunar-derived LOX: **(a)** reusable, cargo, one-stage, aerobraked; **(b)** reusable, cargo, one-stage, nonaerobraked; **(c)** reusable, manned, one-stage, aerobraked; and **(d)** reusable, manned, one-stage, nonaerobraked.

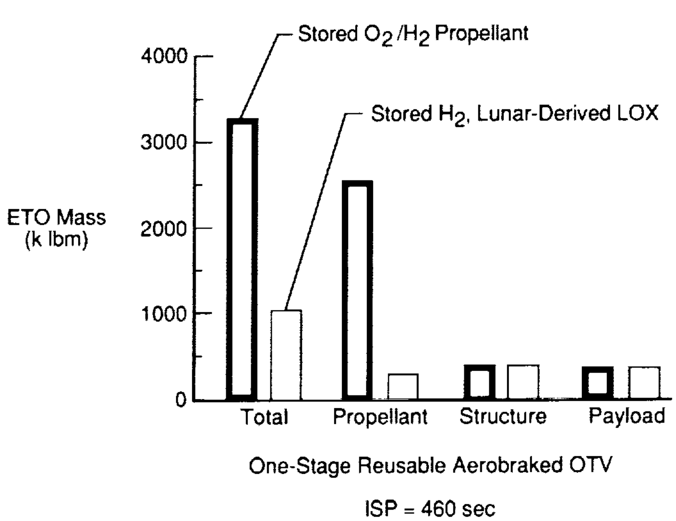


Fig. 10. Impact of lunar-derived LOX on total ETO mass.

primarily to the propellant mass reduction from 2.5 million lbm (Earth-derived LOX) to 0.35 million lbm (lunar-derived LOX). The estimated mass of a pilot LOX plant is included in the lunar base facility and equipment mass (Table 3), but a LOX production plant with an estimated mass of 8400 lbm (*Williams et al.,* 1979) is needed to derive the benefits shown here.

CONCLUSIONS

A systems analysis and assessment has been conducted on the transportation requirements to support a Phase II lunar base mission. The objectives of the study were to assess the relative impact of lunar base support requirements on a LEO-based transportation system and to identify key and/or enabling technologies.

It is immediately evident from the analysis that construction and support of a Phase II lunar base will place a tremendous burden on any space transportation system. The development of the Phase II lunar base will require 3 million lbm to 4 million lbm total weight in LEO over the course of some 20-30 launches of

a 150,000-lbm HLLV. Considering trajectory limitations for specific Earth-to-Moon missions, coupled with even the most optimistic ETO and LEO turnaround scenarios (not addressed in this report), this translates into a commitment of several years of dedicated lunar missions.

From an ETO mass standpoint, only small differences were noted between the use of reusable or expendable systems. However, the cost of expendable modules and vehicles must be considered relative to the developmental cost of the reusable system. It is possible that the developmental cost of a reusable system may offset its operating cost advantage over an expendable system. It appears that using a two-stage OTV yields no significant advantage in mass savings. In terms of operational logistics, then, a one-stage OTV makes the most sense. Aerobraking stands out as a critical, if not enabling technology. Over the course of 16 lunar missions, aerobraking can reduce LEO masses and corresponding ETO lift requirements on the order of 1.5 million lbm to 2 million lbm. Aerobraking is also critical in making a reusable OTV advantageous. As expected, the higher the I_{sp} of the engine, the lower the fuel needs and ETO masses. The ETO masses were also observed to be more sensitive to I_{sp} in reusable and all-propulsive modes. The use of aerobraking reduced the impact of increasing I_{sp}. An engine with an I_{sp} of 485 sec is probably beyond the near-future state of the art, but an I_{sp} of 460 sec appears definitely achievable. Utilizing lunar-derived oxygen for lunar landing, ascent from the lunar surface, and return to Earth orbit can reduce mission start mass to 16-50% of that required with Earth-derived LOX.

Overall, the trend analysis of this study indicates that the optimum transportation system would be a one-stage, aerobraked, reusable vehicle with the highest engine efficiency attainable. The use of lunar oxygen is advisable.

REFERENCES

Ketchum W. J. (1986a) *Orbital Transfer Vehicle Concept Definition and Systems Analysis Study, Final Report, Volume II—OTV Concept Definition and Evaluation.* GDSS Rpt. SP-86-011, General Dynamics Space Systems Division, Huntsville. 136 pp.

Ketchum W. J. (1986b) *Orbital Transfer Vehicle Concept Definition and Systems Analysis Study, Final Report, Volume IX—Detail Summary.* GDSS Rpt. SP-86-011, General Dynamics Space Systems Division, Huntsville. 215 pp.

Mendell W. W., ed. (1985) *Lunar Bases and Space Activities of the 21st Century.* Lunar and Planetary Institute, Houston. 865 pp.

National Commission on Space (1986) *Pioneering the Space Frontier.* Bantam, New York. 214 pp.

Roberts B. B. (1986) Mission analysis and phased development of a lunar base. *37th Congress of the International Astronautical Federation,* Paper No. IAF 86-512, Innsbruck, Austria. 11 pp.

Williams R., McKay D., Giles D., and Bunch T. E. (1979) Mining and beneficiation of lunar ores. In *Space Resources and Space Settlements* (J. Billingham, W. Gilbreath, and B. O'Leary, eds.), pp. 275-288. NASA SP-428, Washington, DC.

LUNAR BASE MISSION TECHNOLOGY ISSUES AND ORBITAL DEMONSTRATION REQUIREMENTS ON SPACE STATION

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The International Space Station has been the object of considerable design, redesign, and alteration since it was originally proposed in early 1984. In the intervening years the station has slowly evolved to a specific design that was thoroughly reviewed by a large agency-wide Critical Evaluation Task Force (CETF). As space station designs continue to evolve, studies must be conducted to determine the suitability of the current design for some of the primary purposes for which the station will be used. This paper will concentrate on the technology requirements and issues, the on-orbit demonstration and verification program, and the space station focused support required prior to the establishment of a permanently manned lunar base as identified in the National Commission on Space report. Technology issues associated with the on-orbit assembly and processing of the lunar vehicle flight elements will also be discussed.

INTRODUCTION

In early 1987, the Office of Aeronautics and Space Technology, NASA Headquarters, requested that the Langley Space Station Office perform a study to assess the impact of a manned lunar base mission on the Critical Evaluation Task Force (CETF) IOC space station. An agency-wide team was formed to investigate the space station support necessary to accommodate such a mission, with emphasis on precursor research requirements, lunar mission support requirements in low Earth orbit (LEO), concurrent science applications, technology requirements and issues, and station resource requirements including crew, power, and volume. The results of this study are published in *Weidman et al.* (1987).

From a review of recent studies conducted by NASA and in concert with the Civil Space Leadership Initiative (CSLI) activities, a baseline lunar base mission scenario was postulated, and the top-level technology requirements and issues needed to support such a mission were identified. These top-level issues were then analyzed to determine technology areas needing early or accelerated emphasis, and a statement of near-term and far-term requirements was formulated in terms of applicability to the lunar base initiative. From this analysis, the systems-level technologies that were considered enabling were identified, and an orbital demonstration and verification program for the major flight hardware elements of the lunar vehicles was developed.

Key lunar base mission technology implications are summarized in terms of the space station requirements and on-orbit support activities. Technology areas requiring additional study are identified and include in-space processing and serviceability, space-storable cyrogenics, automation and robotics, automated rendezvous and docking, etc. Some basic requirements for an orbital maneuvering vehicle (OMV) -type vehicle with increased capability and operational flexibility are presented.

LUNAR BASE ACCOMMODATION STUDY OVERVIEW

Before addressing the specific technology issues and on-orbit demonstration program requirements, a brief overview of the study results presented in *Weidman et al.* (1987) will be discussed.

The overall study objective was to establish and, where possible, quantify all the lunar base mission impacts on the IOC space station (on resources, interfaces, science, technology DDT&E, and configuration) resulting from accommodation of the lunar base mission. Of particular importance to the study were the on-orbit resource requirements in terms of crew, power, and volume, the impacts to the station science, and the enabling and enhancing technology requirements.

The basic assumptions and ground rules that were used in the study were (1) the CETF IOC configuration is the study baseline; (2) there will be an early manned lunar mission; (3) there will be lunar sample return and rover precursor missions with expendable launch vehicles (ELVs); (4) the Johnson Space Center (JSC) lunar base scenario is the primary basis for space station mass flow; (5) lunar mission vehicle buildup will take place in LEO; (6) a hydrogen/oxygen chemical proplusion system will be used; (7) orbital transfer vehicles (OTVs) and OMV will be man rated and space based; (8) heavy lift launch vehicles (HLLVs) will be operational; and (9) the study does not consider a post 2010 timeframe.

Unmanned precursor missions, which include lunar orbiters, sample return vehicles, and surface rovers, will be delivered by ELVs launched directly from Earth. From the onset of the early manned lunar missions to the establishment of a permanent lunar base, all lunar mission elements will pass through the space station. The mass-to-LEO necessary to support the flight rates assumed for the program dictated the need for an HLLV. The

station-based OTVs and OMVs were assumed to be available early in the program from the vehicles' on-orbit verification and man-rating programs beginning at station IOC.

In the study four possible on-orbit basing options for vehicle preparation and maintenance were considered: (1) all vehicle accommodations are based on the space station; (2) the vehicle hangar is based on the space station but propellant is located on a co-orbiting facility; (3) all vehicle accommodations except the crew habitation module are based on a co-orbiting facility; and (4) all vehicle accommodations including the crew module are based on a co-orbiting facility.

Only options 1 and 2 were analyzed in any detail for their impacts on the station configuration, control characteristics, and static microgravity profiles. In option 3 the major impact would be increased traffic to and from the station to accommodate the support crew shift changes. Option 4, by definition, would produce little or no effect on the station.

The station configuration shown in Fig. 1 shows option 1 with the vehicle hangar/service facility above the transverse boom and attached to the upper keel, while the propellant tanks are below the boom and attached to the lower keels. The JSC lunar base scenario, which provided the fundamental definition of the total mass flow through the station, consists of three phases and is shown in Table 1.

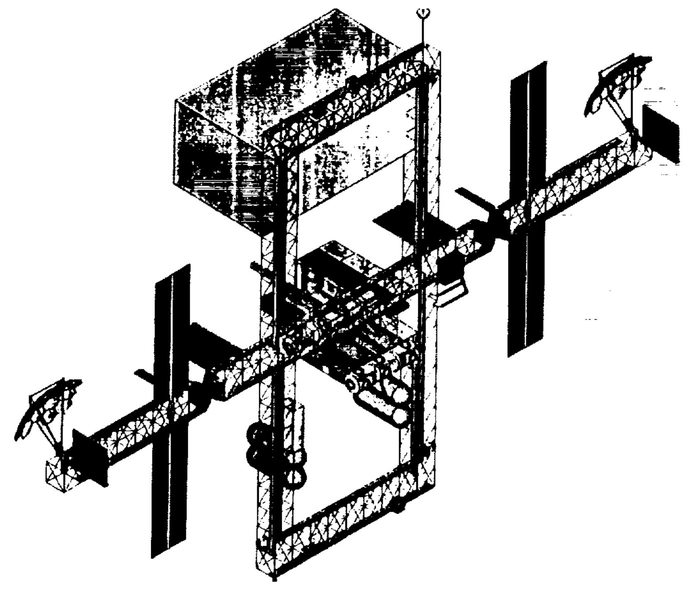


Fig. 1. Space station configuration—option 1.

TABLE 1. Lunar base scenario.

Phase I: Preparatory Exploration (Robotic)
- Lunar orbiter explorer and mapper
- Site selection
- Possible automated site preparation

Phase II: Research Output (0-4 Personnel)
- Man tended
- Habitat module
- Total Earth supply
- Science module
- Lunar oxygen pilot plant
- Surface mining pilot operation
- Power unit

Phase III: Operation Base (4-12 Personnel)
- Permanently occupied facility
- Additional habitats and laboratories
- Expanded mining facility
- Oxygen production plant
- Additional power

The first phase begins in 1994 with the primary objective being to assess and select a candidate landing site. This phase would commence with a lunar orbiting satellite to provide detailed mapping of the entire lunar surface. This would be followed by sample return missions and delivery of unmanned rovers for detailed landing site evaluations. The final step in this phase could be delivery of automated construction equipment to the surface for initial site preparation.

The second phase of the scenario establishes a man-tended research outpost and begins with the delivery of a small power plant, a habitat, an unpressurized rover, and various scientific experiments. A crew of four will operate the outpost for up to two weeks at a time during the first two years. As more facilities and equipment are delivered, stay times will increase and small-scale mining operations and oxygen production experimentation will commence.

Phase III begins about 2005 with the goal of establishing a permanently manned lunar base. During this phase the number of crew will increase to 12 with the habitats, facilities, and equipment necessary to support large-scale oxygen production. A lunar orbiting support facility will have been established as a storage/transfer depot for the lunar-produced oxygen and as a staging area for the arriving and departing lunar mission crews.

The major development milestones necessary for implementing the phased lunar base program are shown in Fig. 2. Key space station events are indicated. As mentioned earlier, OTV and OMV development and orbital verification should start at station IOC, as well as the orbital assembly and outfitting of the OTV and lunar vehicle and hangar/service facility. The milestones for the lunar vehicle elements reflect a very ambitious and success-oriented schedule considering that all the flight hardware elements must be assembled on orbit, tested, and verified in two years!

The space station support requirements that need to be addressed in order to successfully meet the schedule milestones are shown in Table 2. In this table, the primary activities required by the station to support a lunar base are shown as a function of time and include all the program phases discussed. The early activities, 1997-2000, affecting the station requirements support are primarily the on-orbit technology development and demonstration program, the on-orbit facility support buildup, and the lunar vehicle testing and verification program. The station support requirements in the 2000-2010 timeframe include (1) the capability to support routine vehicle servicing, refurbishment, and missions operations and (2) the advanced technology development programs necessary to establish the permanent manned facility on the lunar surface. These advanced programs and their implications on the evolutionary growth of the LEO and lunar orbit infrastructures will undoubtedly be challenging topics for future study activities such as those emerging from NASA's Office of Aeronautics and Space Technology's Project Pathfinder. Also, during this latter time frame, the orbital activities and mass-to-orbit requirements necessary to support the lunar base (and quite possibly the manned Mars initiative) will most likely have established the need for an LEO transportation node as part of the in-space infrastructure.

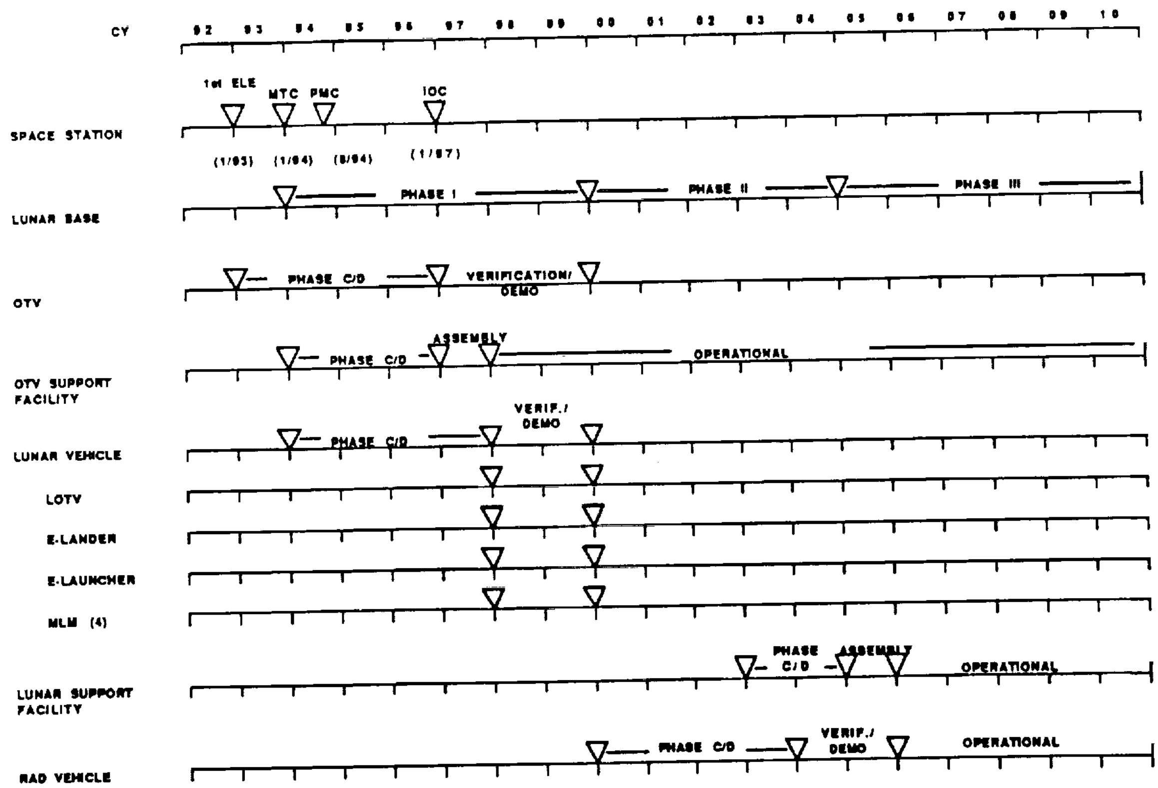


Fig. 2. Major milestones for the lunar base.

TABLE 2. Space station support requirements.

1997-2000
• On-orbit facilities buildup
• Technology development/demonstration
• Lunar vehicle demonstration/verification
2000-2010
• Lunar vehicle servicing
• Lunar base mission support
• Advanced technology development/demonstration
• Advanced lunar vehicle development/verification

To summarize the lunar base overview, the majority study accomplishments are (1) mission and mission vehicle are defined; (2) detailed operations analysis are concluded; (3) strawman Kennedy Space Center (KSC) flight schedule is developed; (4) space station accommodation options are identified and analyzed; (5) space station science effects are analyzed; (6) technology requirements for lunar base support are examined; and (7) on-orbit development program requirements are developed.

The remainder of this paper will concentrate on the last two items, the overall development of the technology requirements and the on-orbit technology demonstration and verification programs necessary to support this initiative.

TECHNOLOGY REQUIREMENTS/ISSUES

In order to assess the specific technology requirements and their impacts on the station, it was necessary to first identify the top-level technology issues that must be addressed in order to establish a permanently manned presence on the Moon. These technologies, shown in Table 3, are "across the board" or generic in nature, and are relevant to the entire initiative.

TABLE 3. Top-level technology issues.

• Advanced Environmental Control and Life Support System (ECLSS) - Air, water, waste management, food processing
• Crew Systems - Advanced EVA suits - Habitability considerations - Health care and maintenance considerations
• Surface Transportation - Rovers (unmanned, manned)
• Automation and Robotics - Cargo handling - Assembly - Remote site exploration
• Structures - Aerobrake/aeroshell - Assembly and handling
• Power/Thermal - Solar - Nuclear - Chemical
• Long-life Mission Systems/Subsystems - Radiation/temperature effects - Propellant storage - Maintenance/activation

The technologies indicated on the table were not prioritized or time-phased, but do serve as a basis for a point of departure in the study to determine areas of specific emphasis for the space station support. For example, the structures, automation/robotics, and life-support technologies being developed under the space station program are directly transferable to lunar base applications. Technology areas such as surface transporation, power generation, and thermal protection could best be done on the ground with prototype and final hardware demonstration and verification being done on the lunar surface.

In the following discussion, only those technologies that needed the space station for direct support will be considered at any depth. These "station focused" technology issues are shown in Table 4. The first five technology issues listed were those the study identified as needing early or accelerated emphasis. These may be looked at as enabling technologies, whereas the items listed under "Space Station Supporting Technology and Development" could be considered as enhancing and would be accommodated by the station in any event.

TABLE 4. Technology issues—space station focused.

Accelerated Emphasis
Automation/robotics
Aerobraking
Autonomous rendezvous and docking
Space propulsion systems
Space cryogenics
Space Station Supporting Technology and Development
Environmental Control and Life Support Systems (ECLSS)
Guidance, navigation, and control (GN&C)
Communications and tracking (C&T)
Extra vehicular activity (EVA)
Data management system (DMS)

Table 5 shows the technology issues just discussed with a brief statement as to their application to the near-term and long-term lunar program requirements. For example, the automation/robotics technology, while key to the success of the lunar vehicle on-orbit servicing/refurbishment requirement, is also an essential technology necessary to support the lunar base surface operations. This is equally true for the automated rendezvous/docking issue, where sophisticated systems are required to support both the numerous LEO and lunar orbital operations that have been identified. Guidance, navigation, and control and Comm/Tracking are also key technology issues when the amount of traffic that can be expected in the space station and the lunar vicinity is considered.

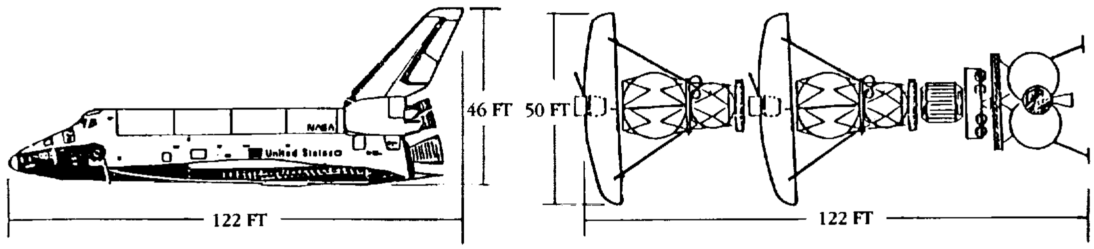
As mentioned earlier, the handling of space cryogenics needs early emphasis in that the transfer, storage, and management of space-storable propellants is critical to mission success. This becomes even more apparent later in the program when lunar oxygen production becomes a reality. Fuel-related issues include (1) fuel transfer (tank to tank/tank to vehicle), (2) fuel storage/boil off; (3) on-orbit tank handling (automated rendezvous/docking and OMV capabilities); and (4) robotic/teleoperator servicing/operations. Solutions to these issues are also keyed to the supporting automation/robotics and the automated rendezvous/docking technologies.

Technology issues include (1) space-based diagnostics/prognostics (in-space systems checkout, onboard/orbit decision making for safe systems operations, and systems health prediction/status; (2) in-space shelf life of lunar-base hardware/spares inventory in LEO, lunar vicinity; (3) in-space processing of hazardous (wet) systems; and (4) pressurized transfer of mission crew to fueled lunar vehicle. These issues evolved from the analysis of the lunar vehicle in-space processing and turnaround requirements developed by the KSC study participant.

TABLE 5. Near-term and long-term lunar program technology requirements.

Automation/Robotics
• Lunar vehicle preparation/servicing in LEO
• Lunar base surface operations
Aerobraking
• OTV LEO operations
Automated Rendezvous/Docking
• OTV, OMV, HLLV, LEO operations
• Lunar vehicle lunar orbit operations
Space Propulsion Systems
• OTV, E-lander, E-launcher engine development
• OTV, OMV propulsion systems reusability, maintainability, refurbishment
Space Cryogenics
• Propellant transfer and storage
ECLSS
• Manned lunar module (MLM)
• LEO/LO support operations
• Lunar base operations
GN&C
• Traffic control in LEO
• OMV, OTV LEO operations
• Lunar vehicle translunar and lunar orbit operations
• Lunar orbit system
Comm/Tracking
• Traffic control in LEO
• OMV, OTV LEO operation
• Lunar vehicle translunar and lunar orbit operations
• Lunar orbit systems
EVA Systems
• Lunar surface operations
• LEO support operations
DMS
• LEO support operations
• Lunar base support
• MLM support

The space-based diagnostics/prognostics issue is key to successfully meeting the rigid turnaround schedule requirements developed in the study and for establishing the high degree of confidence required for safe systems operation. The degree of modularity, the level of component changeout and replacement, engine/tank reusability, spares inventory, etc. will be real challenges to designers to provide "serviceability" to all the lunar vehicle systems. The Lewis Research Center (LeRC) is proposing studies on reusable space propulsion systems that are directly applicable to in-space vehicle processing, especially in the area of expert system intelligence for monitoring, diagnostics, and control. The issues of on-orbit processing of hazardous (wet) systems and the pressurized transfer of crewmen to fueled space vehicles will also require new and innovative "operational philosophies" in order to provide timely and safe solutions to these problems.



SPACE SHUTTLE	
Length	122 ft
Width	78 ft
Height	46 ft
Dry Weight	165,000 lb
On-Orbit Weight	230,000 lb
Engine Systems	
SSME	LOX/LH
OMS	MMH/NTO
RCS	MMH/NTO
Subsystems	
ECLSS	
GN&C	
C&T	
EPS	
DMS	
EVAS	

MANNED LUNAR VEHICLE	
Length	122 ft
Width	50 ft
Dry Weight	56,000 lb
Earth Departure Weight	248,000 lb
Engine Systems	
Space Prop.	LOX/LH
RCS	MMH/NTO
E-Lander	LOX/LH
E-Launcher	MMH/NTO
Subsystems	
ECLSS	
GN&C	
C&T	
EPS	
DMS	
EVAS	

Fig. 3. Comparison of space shuttle and the manned lunar vehicle.

Figure 3 graphically depicts the magnitude of some of the challenges associated with the on-orbit vehicle processing and servicing mentioned above. This figure shows the space shuttle orbiter and the manned lunar vehicle configuration to approximately the same scale. Not only is the lunar vehicle as large, in many ways it is as complex as the orbiter. It has more engine systems and more elements that need to be serviced, integrated, and checked out, all on orbit with limited "hands-on" personnel.

Having identified the key technology areas relative to the station support role, the next step was to define the systems-level technology issues. Tables 6, 7, and 8 address these issues for the major flight hardware elements of the lunar vehicles. Each of the new development items that comprise the manned lunar vehicle is listed along with the major subsystems/functions that make up that element. Table 6 depicts those elements unique to the manned module.

TABLE 6. Systems-level technology issues—manned module only.

Element/Function	SS-Derived	New	LEO Dev. Test STS	LEO Dev. Test SS
ECLSS	Yes	Some	No	Yes
EPS	No	No	No	Yes
GN&C	Yes	Yes	Yes	Yes
Comm/Tracking	Yes	Yes	Yes	Yes
EVA Systems	Yes	Yes	Some	Yes
DMS	Yes	No	No	Yes
Command/Control Interface	Some	Yes	Yes	Yes

TABLE 7. Systems-level technology issues—orbital transfer vehicle.

Element/Function	SS-Derived	New	LEO Dev. Test STS	LEO Dev. Test SS
Automated Rendezvous/Docking	Yes	Yes	Yes	Yes
ACS	Yes	No	No	Yes
GN&C	Yes	Yes	Yes	Yes
C&T	Yes	Yes	Yes	Yes
Propulsion System	No	Some	yes	Yes
(Reusability Tech)	No	Yes	Yes	Yes
Aerobrake/Aeroshell	Yes	Yes	Yes	Yes
Command/Control Interface	Some	Yes	Yes	Yes

TABLE 8. Systems-level technology issues—expendable elements.

Element/Function	SS-Derived	New	LEO Dev. Test STS	LEO Dev. Test SS
E-Lander				
GN&C	Yes	Yes	Yes	Yes
C&T	Yes	Yes	Yes	Yes
ACS	Yes	No	No	Yes
Propulsion System	No	Yes	No	No
Command/Control Interface	No	Some	Yes	Yes
Rover	No	Yes	No	No
E-Launcher				
GN&C	Yes	Yes	Yes	Yes
C&T	Yes	Yes	Yes	Yes
ACS	Yes	No	No	Yes
Propulsion System	No	Some	No	No
Command/Control Interface	Some	Yes	Yes	Yes

In an attempt to define the technology readiness of the flight hardware, an overall assessment was made of the availability of the technology as shown in the first two columns. These technology requirements were identified as being station derived (required by the station program itself), new technology, or some combination of both. As can be seen, over half of those identified were found to be highly dependent on space station heritage. The applicability of using the shuttle and/or space station experience for the on-orbit development and testing for the lunar base elements is indicated in the last two columns of the figure.

In Table 7, the OTV main propulsion system is an excellent example of capitalizing on the experience base to be accumulated on the space shuttle main engines (SSMEs). This base, along with the proposed LeRC research on reusable space propulsion systems, will be invaluable in finding solutions to the challenges associated with on-orbit processing and refurbishment.

In Table 8 the systems-level issues for the expendable elements are shown. As the program matures into the Phase II timeframe, these elements will be replaced by reusable vehicles. The systems/subsystems technology requirements for these reusable vehicles will have benefited from the early development activities associated with expendable elements.

From this systems-level analysis, the single common thread that ran through all the elements was the command and control interface function. This requirement was due primarily to the "man in the loop," who is an integral part of all vehicle systems. For example, no matter how sophisticated the automated rendezvous and docking system becomes, the crew must have the capability to monitor, assess, and intervene if necessary, to take active, real-time control of any vehicle or situation of which they are a part.

ON-ORBIT TECHNOLOGY DEVELOPMENT AND DEMONSTRATION CONSIDERATIONS

The purpose of the on-orbit technology development and demonstration program was to evaluate and demonstrate the operation of the systems, the techniques, and the components of the mission elements and functions to insure a high degree of confidence in their operations.

Long-term, dependable operation is achieved by high reliability, maintainability, repairability, and/or replacement. The on-orbit technology program must insure that the proper balance of these attributes has been determined for the particular system or subsystem selected. In developing the orbital demonstration/testing program discussed here, the subsystem selection, the development of the operational procedures, and the space asssembly techniques should be made as early in the program as possible, while maximizing the use of space station hardware and operations experience. As much testing and verification as is feasible must be done before flight hardware is committed to orbit.

The primary items that must be considered in the on-orbit demonstration program are identified in Table 9. In this table, the lunar vehicle systems are shown with the major testing and verification requirements listed for each of the flight hardware elements. In addition to those listed, end-to-end testing and all-up mission simulations with the totally integrated lunar vehicle configuration will be required.

TABLE 9. On-orbit program demonstration considerations.

Testing/Verification

OTV
- Rendezvous/docking with OMV
- Rendezvous/docking with MLM
- Separation test—OMV, MLM, cargo module
- Serviceability/turnaround procedures
- Fueling
- Aeroshell performance

OMV
- Rendezvous/docking with HLLV
- Rendezvous/docking with lunar vehicle (OTV/MLM, OTV/cargo)
- Serviceability/turnaround procedures
- Fueling

Manned Lunar Module (MLM)
- Subsystems verification
- Command/control interface verification
- Serviceability, maintenance
- Mission simulations
- Crew transfer, premission/postmission C/O procedures

E-Lander/Launcher
- Separation, rendezvous, and docking demonstration
- Landing and ascent demonstration
- Mission simulation (manned, unmanned)
- Fueling

Aerobrake/Aeroshell
- Assembly
- Serviceability/refurbishment procedures

ON-ORBIT PROGRAM RESOURCE REQUIREMENTS

As stated earlier, the primary thrusts of the paper were the on-orbit technology requirements and the on-orbit demonstration and verification programs with emphasis on station impacts in terms of crew, power, and volume requirements. The on-orbit resource estimates developed for the thrusts are shown in Table 10, and the term "user" refers to those requirements over and above basic station capabilities or allotments.

TABLE 10. On-orbit resource estimates for lunar mission support.

Activity	User Crew	User Power	User Volume
Precursor Program			
Technology development demonstration*	4	15 kW	0.5 lab
Mission Support			
Vehicle assembly, servicing, and checkout	6-12	30 kW†	1 lab
Mission crew	4-12		

* Includes systems testing verification.
† Includes cryo management.

The estimates indicated for the precursor program activity were derived primarily from detailed analysis of the on-orbit demonstration program just discussed. The rather high crew estimates include the personnel requirements for vehicle systems/subsystems monitoring and for crew support, while tests of the rendezvous and docking, fueling, landing/ascent, aeroshell perfor-

mance, etc. are in progress. Also included is the crew needed for the hangar/service facility and construction and assembly in the 1997 timeframe and for the manpower required to develop, test, and validate the vehicle processing and turnaround procedures during the two years prior to phase II initiation.

The power estimate includes the base load necessary to sustain the systems/susbsystems monitoring functions and an allowance to support a command/control capability on the station. This base load averaged about 6 kW/yr over the 1997-2000 technology development period. The bulk of the power usage, approximately 9 kW, was due primarily to requirements from the vehicle hangar/service facility and to the technology program associated with storage, reliquefaction, and transfer techniques of the space-storable cryogenics. The volume requirements shown represent the pressurized/internal volumes needed to accommodate the monitoring and command/control functions associated with the demonstration and verification support demands.

The mission support activity, which begins at the onset of phase II, puts the most severe demands, in terms of crew, on the basic station resources. Vehicle assembly, servicing, and checkout can require from 6 to 12 additional crewmen depending on the flight rates and turnaround times assumed in the program scenario. If we assume we need to maintain the baseline crew of 8 in order to preserve the basic research mission of the station, there is now an on-orbit crew requirement that ranges from 14 to 20 people. This equates to an additional two habitat modules in order to support routine station and lunar mission operations. The lunar base/mission crew will grow from 4 to 12 by the year 2010. However, these are transient personnel and could probably be accommodated by "doubling up," so to speak, in the additional habitat modules.

The 30-kW power requirement shown for the mission support activity includes the energy necessary to support the vehicle assembly, tests, and servicing functions, as well as providing the power needed for on-orbit space cryogenic management. During the operational time period, a dedicated pressurized service and assembly facility, equivalent in size to a lab module, will be required to manage daily activities associated with vehicle processing and mission control.

SUMMARY

The lunar base program and its attendant requirements can be characterized by long-duration, operationally intense missions. The program's success will depend upon an ambitious flight support schedule requiring a substantial expansion of our current Earth-to-LEO launch capabilities, and significant advances in the automation and robotics technology.

The primary focus on the space station activities in support of the lunar base mission early in the program will be the on-orbit technology development, testing, verification of flight hardware, and some orbital demonstration experimentation. The operational phase will require significant support for the assembly, refurbishment, and maintenance of the lunar mission elements.

If the lunar vehicles and elements are station based, the assembly, servicing, and maintenance functions will require extensive station interfaces such as those for a large hangar/service facility attached to the station.

The OTV and the OMV particularly must be designed to accommodate the massive mission vehicles, and they must be man rated. Traffic control around and at the station, and contamination due to increased vehicular traffic, must be studied to provide workable procedures and solution.

CONCLUDING REMARKS AND OBSERVATIONS

Some of the key conclusions derived from the referenced study and this paper are summarized below.

1. The CETF space station configuration (dual keel) will accommodate the lunar mission.
2. Crew requirements point to the need for a crew carrier.
3. The lunar vehicle size, complexity, and allocated in-space processing time requires it to be of modular design with high reliability and robotic interfaces.
4. Application of automation and robotics principles is required to improve productivity and increase efficiency of operations.
5. On-orbit servicing and refurbishment, space storable cryogenics, and automated rendezvous and docking technologies should be accelerated.

REFERENCES

Weidman D. J., Cirillo W. M., Llewellyn C. P., Kaszubowski M., and Kienlen E. M. (1987) *Space Station Accommodation for Lunar Base Elements—A Study.* NASA TM-100501. 243 pp.

ADAPTION OF SPACE STATION TECHNOLOGY FOR LUNAR OPERATIONS

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Space Station Freedom technology will have the potential for numerous applications in an early lunar base program. The benefits of utilizing station technology in such a fashion include reduced development and facility costs for lunar base systems, shorter schedules, and verification of such technology through space station experience. This paper presents an assessment of opportunities for using station technology in a lunar base program, particularly in the lander/ascent vehicles and surface modules.

INTRODUCTION

Current concepts for a lunar base program (*Duke et al.,* 1985; *Hoffman and Niehoff,* 1985; *Woodcock,* 1985; *Ride,* 1987; *National Commission on Space,* 1986) assume the presence of a low Earth orbit (LEO) space station as part of the overall mission infrastructure (Fig. 1). Such a station will function as a staging platform between Earth launch systems and lunarbound orbital transfer vehicles (OTVs), providing services such as vehicle assembly, checkout, and fuel storage.

Space Station *Freedom* (Fig. 2) represents the first step to creating such a LEO facility. This Phase I station will serve both as a testbed to develop the servicing capabilities mentioned above, and as a life sciences laboratory to gain better understanding of how life can function in space. Eventually, it could evolve into the staging platform for lunar missions.

An equally important aspect of Space Station *Freedom* is that the systems-level technologies that NASA is developing specifically for this program, such as data management, guidance and navigation, and communications, represent basic capabilities that in many cases can be applied directly to lunar base elements. This approach of using existing systems has been followed throughout the long history of lunar base planning (*Lowman,* 1985; *Johnson and Leonard,* 1985). Now, with the advent of the design, development, test, and evaluation portion of the space station program, it is possible to assess such technology transfer at a finer level of detail. This paper reports on a preliminary internal study by McDonnell Douglas of such opportunities for the space station avionics.

BASELINE LUNAR BASE AND SUPPORTING ELEMENTS

This study assumes a Phase II lunar base, as defined by *Duke et al.* (1985) and *Ride* (1987; also known as the "Ride Report"). (Phase I in renewed lunar exploration would entail robotic exploration of the Moon during the 1990s, with the specific goal of finding a suitable site for the eventual lunar base. Phase II would then follow in the 2000 to 2005 timeframe and represents the initial return of people to the Moon. The associated surface facility would grow into the permanently occupied Phase III base, with up to 30 inhabitants by 2010.) Although there are various versions of such a base, they share common requirements and features. Table 1 lists these items, as well as representative values.

Of all the possible elements, only the lander/ascent vehicles and lunar surface modules are considered here for potential applications. Although a lunar orbiting space station would help logisitics and operations, it is not needed until the succeeding Phase III lunar base. The OTV is not included because it may be developed independently of the lunar base program, much like Space Station *Freedom* and the orbital maneuvering vehicle, and therefore is assumed to already exist by the time this program gets under way.

Lander and Ascent Vehicles

Several NASA-sponsored studies (*Babb et al.,* 1984; *NASA,* 1987a) defined a set of expendable/reusable, manned/cargo landers and ascent vehicles. Only the expendable elements are

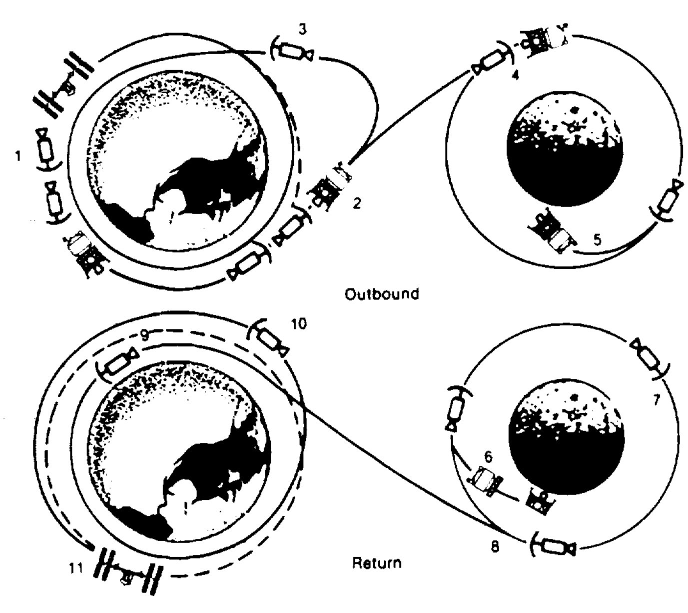

Fig. 1. Lunar base transportation infrastructure.

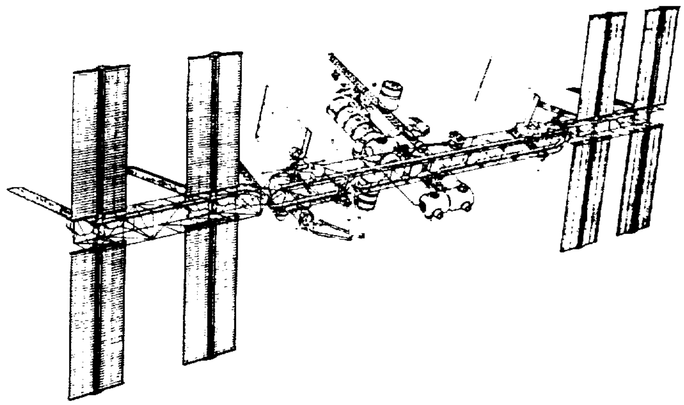

Fig. 2. Phase I Space Station *Freedom*.

TABLE 1. Mission parameters for Phase II lunar base.

Surface stay time	1-3 months
Crew size	3-5
Utilization of lunar resources	Soil for radiation shielding; otherwise, total resupply from Earth
ECLSS closure	Same as space station
Power	75-100 kW
Communications to Earth	Real time video (22 Mbps)
Location	Equatorial, nearside

considered here (Fig. 3) because they are the ones used during Phase II base operations. A large percentage of the avionics and software developed for these expendable landers can be adapted to the reusable versions when they are developed 10 years later.

Although the overall vehicle is expendable, it may prove feasible to recover high-value avionic components and reuse them either in new landers, or else somewhere in the growing lunar base.

Surface Modules

To achieve top-level commonality between the lunar base and Space Station *Freedom*, an initial lunar base design will incorporate space station-type modules. *Hoffman and Niehoff* (1985) propose one such initial operations configuration that consists of three main modules (habitation, laboratory, and service) and several interface nodes, as well as two rovers and a 100-kW nuclear reactor, while *Duke et al.* (1985) present a more generic module arrangement configuration. Figure 4 presents a lunar base model developed as part of our general studies in this area.

The interface elements are derived from the space station resource nodes, while the airlock is comparable to that on the station. A disposable logistics module is used for resupply.

As stated earlier, this review considers only the module systems, not the actual internal module configurations. The impact of the 1/6-*g* level on the microgravity-driven design of the station module interiors merits a separate study.

SPACE STATION TECHNOLOGY APPLICATIONS

Data Management System

The Space Station *Freedom* data management system (DMS) represents a major evolutionary step in onboard space processing capabilities. In contrast to previous space vehicles, which employ a centralized architecture that is based on a main computer (plus backups), the station DMS functions will be distributed among over 20 stand-alone computers, termed standard data processors (SDPs), and several hundred embedded data processors (EDPs). This decentralized approach is intended to provide adequate flexibility to accommodate future station growth, technology improvements, and functional redundancy.

The SDPs and EDPs use the same 32-bit microprocessor (a space-qualified version of the Intel 80386) and present a family of processing capabilities that can fit a variety of user needs (Fig. 5). Other DMS hardware components include the 100 Mbps fiber optic core network, smart multiplexer/demultiplexers (MDMs), work stations, optical and tape mass storage units, and Mil-Std-1553 local data busses.

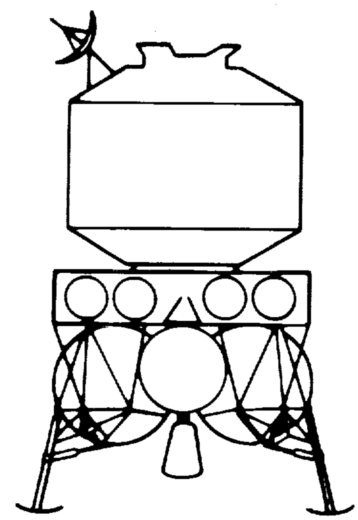

Fig. 3. Expendable lunar excursion module (from *Babb et al.,* 1984). LLMM total weight = 3.25 t. E-launcher propellant weight = 5.0 t; dry weight = 2.6 t; total weight = 7.6 t. E-lander (delivers 17.5 t to lunar surface) propellant weight = 13.6 t; dry weight = 3.8 t; total weight = 17.4 t.

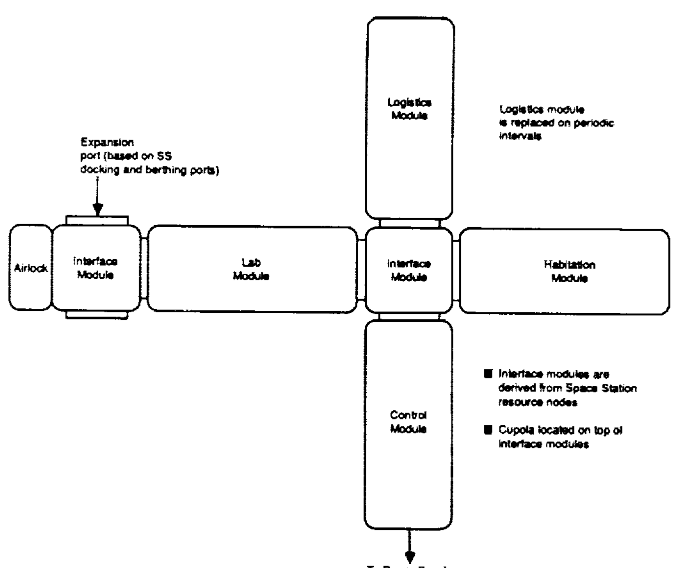

Fig. 4. Lunar base modules.

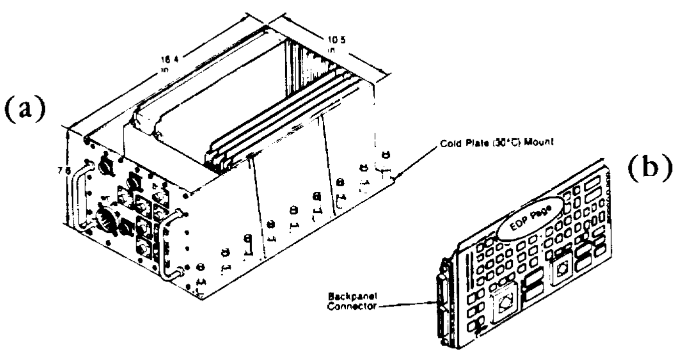

Fig. 5. DMS standard data processor: **(a)** Standard data processor (SDP): 4 Mips/growth to 8 Mips; 4 Mbytes/growth to 64 Mbytes per slot (1995); FDDI optical network interface; optional optical or wire local busses; radiation tolerant; VHSIC class parts. **(b)** Embedded data processor (EDP): 32-bit 80386 industry standard ISA for ground/onboard compatibility.

A single prime SDP, plus backup, using Mil-Std-1553 local busses to access MDMs and EDPs, should be able to provide all the data processing for a lander vehicle. For the surface base, the 100-Mbps core network can link the various elements together, while MDMs support monitoring and control functions. Both kinds of mass storage units could be called on to archive research data.

Software

NASA is undertaking two specific steps to ensure that space station software and the tools used to develop it will be transportable to future systems like the lunar base. First, all station software (with the exception of commercial off-the-shelf programs) shall be written in Ada, a structured language that is written for such transportability. The most "visible" software component will be the dedicated operations management system (OMS), consisting of a ground and on-orbit segment (OMGA and OMA, respectively), which will coordinate station operations and can serve as a model for subsequent lunar base software operating systems. Assuming that DMS hardware is used, the lunar base can also employ lower-level software, such as data display formats, encoding techniques, and built-in test. In general, the base will resemble Space Station *Freedom* in that it will generate a substantial amount of data that can undergo extensive on-site processing before transmission to Earth.

The station software will also include expert systems to provide highly autonomous operations, independent learning, and more efficient resource scheduling. The longer distance from Earth and limited manpower will make these features even more desirable at the lunar base, particularly during the interim periods when there is no crew.

The second relevant software issue is the software support environment (SSE) that NASA is creating to develop this station software (Fig. 6). It will consist of software production facilities (SPFs) at the various NASA centers and their associated contractors for software development, system development facilities (SDFs) for system-level integration of software and hardware, and a single multiple system integration facility (MSIF) where the top-level software integration will take place. All these facilities will incorporate flight-equivalent DMS hardware and operational software, with associated computer-based simulation programs to duplicate payloads and interfaces.

These various facilities will represent important national resources when Space Station *Freedom* is placed in orbit. Because they are functional and not physical equivalents of station systems, the MSIF, SPF, and SDF can easily be rearranged (generally by altering cable connections and rewriting simulation software) to new configurations such as a lander/ascent vehicle or a surface habitation module.

Communications and Tracking

For space-to-ground communications, Space Station *Freedom* will use TDRSS. Dedicated baseband processor units, Ku-Band transceivers, and a 2.75-m steerable antenna provide up to 300 Mbps throughput for real-time video and data transfer (Fig. 7).

The transmission segment of this system will be inappropriate for communications from the Moon to Earth, primarily because the TDRSS satellites are in geosynchronous orbit with their antennas pointing toward Earth. A direct microwave or laser link to Earth, or a dedicated relay satellite, would provide easier access (the microwave system would require larger antenna, ground receivers, and/or up-front amplifiers than those on the station to compensate for the greater distance).

Far better opportunities exist for applying Space Station *Freedom's* multiaccess proximity communication system, as well as internal audio/video and data collection equipment (TV cameras, pan tilt units, etc.). With respect to the proximity communication system, up to four users, such as EVA astronauts and approaching OTVs, can access the station through a second,

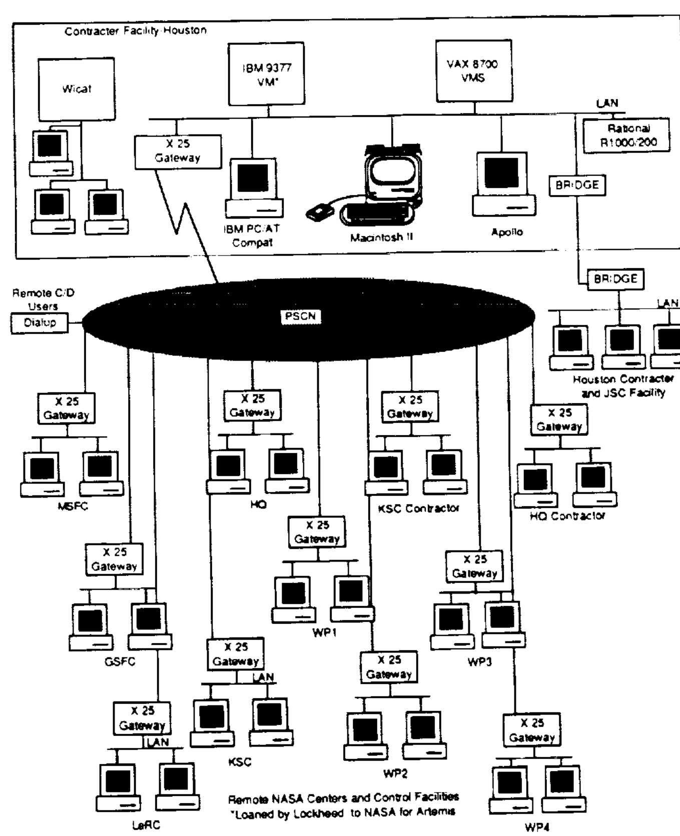

Fig. 6. Interim SSE system hardware and communications (derived from *LMSC*, 1987).

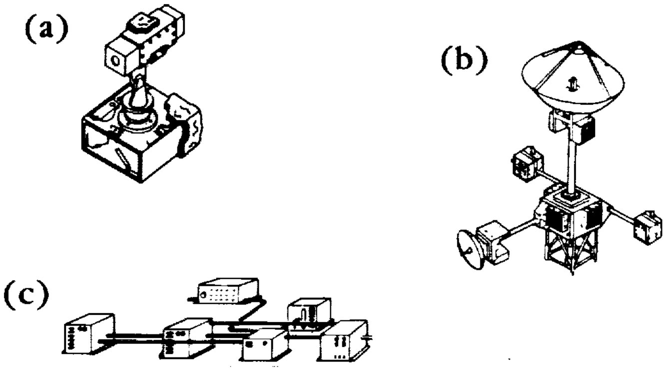

Fig. 7. Communications and tracking hardware: **(a)** video camera/pan tilt unit; **(b)** antenna boom, antenna-mounted equipment; **(c)** TDRSS high data rate frame multiplexer.

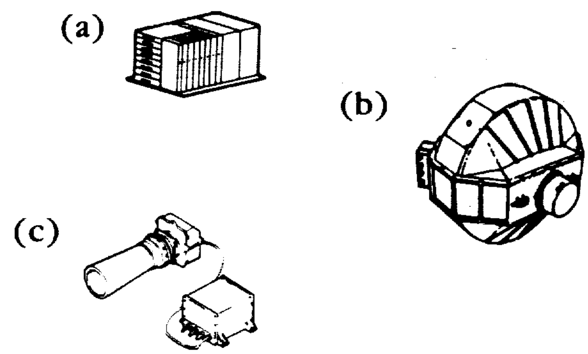

Fig. 8. Space station guidance, navigation, and control components: **(a)** Attitude determination system: Solid-state star tracker (ST) gives reliable, accurate performance; ISA provides reliable attitude data continuity when star tracker data unavailable; ISA/ST accuracy of 0.003°. **(b)** Control momentum gyro: Intell 80C86 processor; 1553B interface; double gimbal; 3500-ft-lb-sec momentum storage; dual electronics for each gimbal; 200-ft-lb torque; passive thermal cooling; BIT/BIT; minimum 10-year life. **(c)** Star tracker alignment ring innovation ensures boresight to navigation base alignment (0.0015°).

separate Ku-Band link that utilizes frequency division multiple access. This capability would serve well on the surface base for links to a lander and EVA work parties.

Guidance, Navigation, and Control

The space station Guidance, Navigation, and Control (GN&C) design incorporates ring laser gyros (RLGs) and star trackers to determine the attitude of a reference "Nav Base" to an accuracy of at least 0.01°. The companion coorbiting and polar orbiting platforms will also have Earth sensors for contingency purposes. Modified off-the-shelf GPS receivers will obtain data to determine position and velocity to a 3σ accuracy of 26 m and 0.1 m/sec, respectively.

Control is implemented through six 6760 N-m-s control momentum gyros (Fig. 8) and several sets of reaction control system thrusters that use gaseous hydrogen and oxygen for propellants.

This equipment is generally not useful for the surface modules, which are intended to retain fixed attitudes and positions on the lunar surface (some surveying tools may be needed for initial site studies and any intentional movements of modules). The main use of station attitude determination technology will be on the landers. The star trackers, in conjunction with lunar ephemeris data and/or Earth sensors, would generate periodic update references with respect to the Moon, while the RLGs would provide continuous information. If the Earth sensors are used, some software modifications will be required to address the different conditions at the Moon (no atmosphere, sharper terminator contrasts, etc.).

The control momentum gyros are probably too large and expensive for the landers, especially if the latter are expendable. The station's RCS technology could be called on if the lander has a H/O propulsion system.

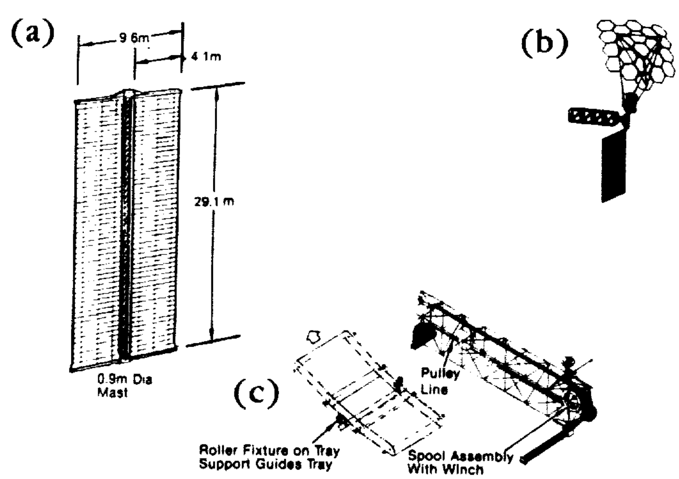

Fig. 9. Space station power components: **(a)** photovoltaic solar array (18.75 kW); **(b)** solar dynamic receiver (25 kW); **(c)** utility tray installation.

Power

The total Space Station *Freedom* power facility consists of the electrical power system (EPS) and power management and distribution (PMAD) (*NASA*, 1987b). The EPS also performs the power storage task for the night portion of every orbit. Figure 9 depicts major components of these systems.

EPS will use four 9.6×29.1-m photovoltaic (PV) arrays during Phase I to generate 75 kW of power (end of life). Solar dynamic generators (SDs) are planned for Phase II of the station program and will add an additional 50 kW of power. The station will represent the first on-orbit application of this technology. Nickel-hydrogen batteries are used to store PV output, while molten eutectic salts undergo a phase change to maintain a set temperature difference in the SD receivers while the sun is eclipsed.

Like the SDs, the station PMAD entails major changes over current space vehicle power distribution systems (these changes are driven by the large size of the station). It will distribute 20 kHz ac at 440 V_{ac} along primary feed lines and 208 V_{ac} to users,

in contrast to the 28 V_{dc} used on most current spacecraft and 400 Hz ac on aircraft. This high frequency is expected to lead to lower transformer and switching equipment weights.

For the landers, only the Ni-H batteries may have some application. Otherwise, the rest of the power system can incorporate more traditional spacecraft components that operate at 28 V_{dc}.

The potential applications of Space Station *Freedom* power generation technology for the surface elements is less clear than for the previous technologies. Although the lunar base will have a grid architecture and power levels comparable to the station, the long duration of the lunar night will place drastically different requirements on the base's generation and storage systems. This has led many to consider nuclear power for the primary power source instead of solar energy (*Hoffman and Niehoff,* 1985; *Buden and Angelo,* 1985; *French,* 1985). However, as listed in Table 2, there are still a number of viable opportunities for supplemental solar power systems that could utilize the station elements.

The transferability of the 20-kHz PMAD elements is also uncertain. However, the utility tray design (Fig. 9c) can accommodate low-frequency cables and would provide easy deployment during base construction. Operating the lunar equivalent to a backhoe, lunar construction workers would dig a trench between a module and the power generation facility, unroll and connect the utility tray, and then cover it with soil for extra protection against micrometeorites and rover vehicles.

TABLE 2. Applications for solar energy power generation systems.

Initial construction sorties—stay time <2 weeks
Short-term peak power surges
Drilling, heavy machinery
Energy-intensive material processing experiments
Autonomous mobile surface vehicles
Lunar base situated at the lunar poles

SUMMARY

The above discussion demonstrates that even at this early date, many opportunities can be identified for using Space Station *Freedom* technology in the design of lunar base systems and elements, with subsequent benefits of lower up-front costs, reduced technical and schedule risks, and program commonality. Table 3 summarizes such opportunities for the space station avionic systems. An additional benefit of such a study is awareness of what functions cannot be performed by space station technologies and therefore need further research and development.

Future efforts will include (1) a comparable assessment of other Space Station *Freedom* systems and elements (i.e., thermal, EVA, the mobile transporter, ECLSS, resource node, lab/hab module structure, manned systems); (2) continued refinement of the above analysis, particularly to assess cost implications; and (3) application of such a review to manned Mars missions.

TABLE 3. Applications of space station avionics technology in a lunar base.

	Lander/Ascent Vehicle	Surface Module
Data management system		
SDP	X	X
EDP	X	X
100 Mbps F/O network		X
Local data busses	X	X
MSU		X
Ring coupler		X
MDMs	X	X
Software		
MISF	X	X
SPF	X	X
SDF	X	X
SSE	X	X
OMA	X	X
OMGA	X	X
Built-in test	X	X
Expert systems	X	X
Other low-level code	X	X
Communications and tracking		
Multiple access Ku-band transceiver	X	X
Multiple access Ku-band antenna	X	X
Internal audio, video	X	X
EVA radio	X	X
Data collection equipment	X	X
Guidance, navigation, and control		
Star trackers	X	
ISA	X	
CMGs		
RCS	?	
Earth sensors	X	
Power		
Photovoltaic array		?
Solar dynamic receiver		?
Batteries	X	?
20 kHz PMAD		?
Utility tray		X

REFERENCES

Babb G. R., Phillips P. G., and Stump W. R. (1984) *Impact of Lunar and Planetary Missions on the Space Station, Final Report.* Eagle Engineering Report No. 84-85D. Eagle Engineering, Houston.

Buden D. and Angelo J. A. Jr. (1985) Nuclear energy—Key to lunar development. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 85-98. Lunar and Planetary Institute, Houston.

Duke M. B., Mendell W. W., and Roberts B. B. (1985) Strategies for a permanent lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 57-68. Lunar and Planetary Institute, Houston.

French J. R. (1985) Nuclear powerplants for lunar bases. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 99-106. Lunar and Planetary Institute, Houston.

Hoffman S. J. and Niehoff J. C. (1985) Preliminary design of a permanently manned lunar surface research base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 69-75. Lunar and Planetary Institute, Houston.

Johnson S. W. and Leonard R. S. (1985) Evolution of concepts for lunar bases. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 47-56. Lunar and Planetary Institute, Houston.

LMSC (Lockheed Missiles & Space Corp.) (1987) *Interim SSE System User's Guide—Space Station Software Support Environment.* DRL Number 23, NASA Contract NAS9-117830. Houston, TX.

Lowman P. D. (1985) Lunar bases: A post-Apollo evaluation. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 35-46. Lunar and Planetary Institute, Houston.

NASA (1987a) *Lunar Base Mission Accommodation at Space Station.* Status report presented to CSLI STAS meeting, NASA Langley, May 7, 1987.

NASA (1987b) *Power System Description Document.* NASA Lewis Research Center, Cleveland.

National Commission on Space (1986) *Pioneering the Space Frontier.* Bantam, New York. 211 pp.

Ride S. (1987) *Leadership and America's Future in Space: A Report to the Administrator.* U.S. Govt. Printing Office, Washington, DC. 63 pp.

Woodcock G. R. (1985) Mission and operations modes for lunar basing. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 111-124. Lunar and Planetary Institute, Houston.

OPERATIONAL CONSIDERATIONS FOR LUNAR TRANSPORTATION

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Transportation of people and cargo between low Earth orbit and the surface of the Moon will be one of the most important elements in a lunar base program. This paper will identify some of the important lessons from the space shuttle program and discuss their application in future lunar vehicle operations. Also, some unique challenges in flight planning, training, vehicle servicing, payload integration, and flight control for lunar transportation will be outlined. This paper relies heavily on recent studies of space shuttle development and operations with the goal of applying shuttle experience in the design of a practical and efficient lunar transportation system.

INTRODUCTION

The two basic program components of a lunar transportation system are development and operations. The focus of this paper is operations. However, in many ways, efficient operations result from careful planning in the development phase. This planning must be applied in the vehicle design process and in the definition of program goals. Transportation operations will be a major cost factor in a lunar base program, and so the vehicles must be designed for efficiency and low life-cycle cost. The space shuttle program had similar goals at its inception, and so the lessons of shuttle operations provide a good experience base for development of operations concepts for routine lunar transportation. Several significant lessons from the shuttle progam that may apply to lunar vehicle development and operations will be discussed in this paper.

There are many ways to define and subdivide operations functions. The operations functions to be discussed in this paper are flight planning, training, vehicle servicing, payload integration, and flight control. These are the major activities that contribute to operational cost in the space shuttle program (*JSC*, 1988a). In this paper, each operational function will be discussed, and some significant aspects related to lunar operations will be described.

As a starting point, it is necessary to define a basic scenario for the lunar transportation system. It is assumed that all missions in support of lunar base construction and operations will originate at a servicing facility in low Earth orbit. This facility could be similar to the currently planned space station, or it could be a derivative with very different characteristics. Vehicle elements, payloads, propellant, and people will be assembled at the servicing facility to begin a flight to the Moon. The vehicle will depart from Earth orbit under the power of an orbital transfer stage. The vehicle will enter lunar orbit and a descent craft will separate from the transfer stage to land on the lunar surface. Return trips will begin with an ascent into lunar orbit where the transfer stage is waiting. The transfer stage then carries the payload, and possibly the lander craft, back to Earth orbit. It is assumed that insertion into Earth orbit will be accomplished with an aerobraking maneuver followed by small propulsive maneuvers to circularize the orbit and rendezvous with the servicing facility.

There are many variations of the transportation scenario that must be considered. Orbital transfers and lunar descent and ascent can be accomplished with vehicles of one or many stages. A libration point could be used as the staging point in the lunar vicinity rather than lunar orbit. The lunar landing craft could be returned to Earth orbit for servicing, left in lunar orbit for later reuse, or expended after each use. These options and others must be studied in depth to gain an understanding of their implications for system performance and operational efficiency. This paper will not resolve those issues, but it will describe some of the factors that must be considered in the solution.

LESSONS LEARNED FROM SHUTTLE OPERATIONS

There are a number of lessons that have been learned in the initial years of operating the space shuttle that also apply to operation of a lunar transportation system.

Define the Operational Scenario as Part of the Conceptual Design

In the conceptual design phase of the space shuttle, initial assumptions were made about the operational scenario for the vehicle. The crucial element of the scenario was the assumed flight rate of 60 flights per year. As the shuttle design evolved, changes were made due to weight restrictions, budget limitations, and the risks of new technology. The definition of the optional scenario did not remain in step with the vehicle design, and this contributed to underestimation of operations costs by over 400% (*Petro*, 1986).

In the conceptual design of lunar spacecraft it will be important to define the operational scenario for these vehicles. The important scenario elements are flight frequency, lifetime of vehicles, location of transportation nodes, availability of facilities and personnel for servicing, and the traffic model for cargo and people. The relationship between lunar transportation and other

space operations must also be determined. Clear definition of both vehicle designs and the operational scenario will make possible a reliable and realistic estimate of life-cycle cost for the lunar transportation system.

Establish a Strategy for Evolutionary Growth Including the Transition from Development to Operations

The space shuttle can be described as a research and development vehicle with an operational mission. It is a first-generation system, and expectations for using it in a totally routine manner were probably never realistic. It might be realistic, however, to expect that the shuttle system will evolve into second- and third-generation systems that will approach and eventually attain the level of routine operations performed by commercial airlines. This evolution can be accomplished by improving vehicle systems, streamlining operations, and modifying vehicle configuration on the basis of operational experience.

One of the fundamental requirements for a lunar base program will be the need to periodically transport crews to and from the base and to deliver supplies. The scenario being used in the Johnson Space Center (JSC) Lunar Base Systems Study calls for 5 to 10 missions per year in the initial phase of base development. In later phases, the need for supplies from Earth may diminish, but scientific and resource objectives imply an ongoing and even increasing need for routine transportation between the lunar surface and other points in space (*JSC*, 1988b).

Because of the relatively high flight rate, even in the early phase, it will be very important that the lunar transportation system be operated efficiently; otherwise, transportation costs will absorb progam resources to the extent that expanding the base will become difficult or impossible. Since the only previous experience with lunar transportation is from the Apollo Program in the 1960s, the lunar spacecraft being considered here are essentially first-generation vehicles. It will be a major challenge to design these vehicles with the foresight needed to ensure efficient operations in the uncertain environment of the first lunar base development program.

The development program for lunar vehicles should follow a carefully planned strategy, leading from spacecraft that will support the first landings to routine crew rotation and supply flights. As the base grows, the flight rate and the complexity of cargo operations will gradually increase. Most importantly, as the base operations mature, there will be increasing pressure to reduce the level of resources devoted to transportation activity. This means that a plan should be in place to transition smoothly from experimental-type vehicle operations to more routine operations.

The transition from transportation system development to transportation operations creates several requirements that should be addressed in vehicle design. One requirement is that vehicle designs be flexible. It should be possible for vehicle configurations to evolve over time to accept larger cargo loads, larger numbers of passengers, and more demanding performance requirements. Consideration should be given to the fact that vehicle servicing may originally occur in Earth orbit and later in lunar orbit or on the lunar surface. Also, if vehicle servicing is initially performed by human crews, those functions may be later taken over by robots. Designs should also account for the need to periodically upgrade vehicle systems such as avionics, power, or propulsion.

Another means to effectively transition to operations is to specify the long-term requirements for routine operations. Operations requirements include, for example, specifications for software tools for automated flight planning and real-time flight control. Another requirement is the vehicle performance and environmental data needed to validate and implement the planning and flight control tools. Specification of operational requirements provides the rationale needed to support an effective flight test program. A flight test program should not only prove that a vehicle functions acceptably, but it should define the overall performance envelope and provide all the data needed to support streamlined operations. This performance definition will help to eliminate the need for continuing engineering analysis in an era of routine operations (*JSC*, 1988a).

Invest in Technology Development Programs

Early investment in technology development was very beneficial in reducing overall space shuttle development costs (*Petro*, 1986). Areas where advanced technology might be beneficial in lunar vehicle development are (1) thermal protection systems; (2) aerodynamic analysis of atmospheric braking; (3) adaptive flight software for guidance and control; (4) reusable, low maintenance engines; and (4) propellant storage techniques. Any technology area that will increase the degree to which lunar vehicles can operate autonomously should also be pursued. These technology needs are common to most advanced space transportation systems.

Balance Maintainability Against Performance in Design Decisions

Maintenance of high-performance systems is the primary schedule and cost driver in space shuttle orbiter processing, and it is likely to be the same for lunar vehicles (*Petro*, 1986). Lunar vehicles, as currently envisioned, will have high-performance engines using liquid oxygen and liquid hygrogen propellants. These engines will have to be restartable and in some cases must have a wide throttle range. Shuttle experience would indicate that the engines would require a great deal of refurbishment between uses and have a short operational lifetime. A high-performance propulsion system reduces the mass of propellant that must be launched into Earth orbit to support a lunar mission. The advantages of high performance must be weighed, however, against the potential cost of system maintenance. The cost of supporting maintenance facilities in space could be enormous, especially if human crews are required. Emphasis has to be placed on maintainability, even at the expense of performance. Maintenance requirements must be minimized and refurbishment, when required, should be automated as much as possible.

Simplicity is always a good design goal. For routine lunar transportation, it is essential. The interfaces between vehicle elements, such as an orbital transfer vehicle and a lunar lander, should be standardized. The same is true for interfaces between vehicles and payloads. Every effort should be made to keep vehicle configurations simple. Complex shapes for aerobraking vehicles should be avoided, as should configurations in which there can be a wide variation in the center of gravity location.

OPERATIONAL FUNCTIONS FOR LUNAR TRANSPORTATION

Transportation operations can be divided into five functional areas: (1) flight planning; (2) training; (3) vehicle servicing; (4) payload integration; and (5) flight control. Considerations related specifically to each function in lunar transportation operations are described in the paragraphs that follow.

Flight Planning

There are a number of complicating factors involved in lunar flight planning due to the periodic variation of orbital planes and the resulting variation in performance requirements. With a particular vehicle, payload, and destination, there will be an optimum time of departure and flight duration (*Woodcock,* 1985). There may be some flexibility in timing, however, if there is enough excess vehicle performance to support some amount of orbital plane change. Flight planners will have to consider, and in some cases specify, the mass of the payload, length of the departure window, flight duration, stay-time in lunar orbit, stay-time on the lunar surface, and the amount of propellant loaded. In addition, mission abort trajectories will have to be planned.

All the complicating factors mentioned are interrelated. It will require extensive trade studies to fully explore all the lunar trajectory options. However, it should be a goal, by the time lunar base construction begins, to automate the flight planning process. The flight test program must verify the automated flight planning system for the entire range of possible trajectories and flight conditions. Ideally, the flight planning software should be part of an onboard adaptive guidance and control system. Onboard flight planning would make the transportation system less sensitive to uncertainties about the payload and to delays in vehicle servicing and departure. If problems develop during a flight, the onboard system could modify the flight profile or even plan an abort trajectory without assistance from an extensive mission support facility.

Experience from the space shuttle program might lead to the conclusion that an attempt should be made to standardize flight profiles. Such an approach might apply to flights from Earth to a space station, but the same is not true for lunar missions. The variables for lunar flight planning are so numerous and subject to frequent change that development of an automatic system would be a wise investment.

The concern about rapid-response flight planning applies in particular to potential aerobraking maneuvers. There are a number of factors that would affect aerobraking flight design that are subject to change during the course of a mission. These factors include the mass of the returned payload, the amount of remaining propellant, and the vehicle center of gravity. One goal of the development and test programs for specific aerobraking vehicles must be to build a sufficient database to support automated flight planning for the entire range of possible trajectories. Flight planning could become an operational burden if detailed engineering analysis had to be performed each time there was a vehicle, payload, or flight schedule change.

Training

The training of crews for lunar vehicles will present some new challenges. The most significant involves the potentially long mission durations combined with the need to assign multiple tasks to crewmembers. A flight to and from the Moon is not in itself a long mission. The round trip can be completed in less than one week. Training crews to perform their flight functions for early lunar missions with short surface stay-times will be similar to training space shuttle crews. Later, transportation operations for a mature base will probably be performed by specialist pilots or be automated.

Between the earliest lunar missions and the mature operations era there is likely to be a transition period lasting many years. In this transition period, crews who operate the vehicles may also have to remain on the lunar surface for extended periods before piloting the vehicles back to Earth. These astronauts will require extensive training in engineering and scientific functions related to surface base operations, in addition to vehicle operations.

The response to the training challenge can take at least two forms. One approach is to provide facilities for pilot proficiency training as part of the lunar base. This could be done by building a training simulator capability into operational vehicles. This would be an efficient use of existing equipment, and the technique might be applied in other space activities, especially a Mars mission.

A different approach is to design vehicles that are essentially automatic. If common vehicles are used for both manned and unmanned missions, lunar spacecraft that carry people should already be capable of operating without human intervention. However, it would take a significant change in philosophy to no longer train crews to manually control spacecraft. The approach taken in lunar transportation is likely to be a combination of enhanced automatic capabilities along with continuation of flight training during stays at a lunar base.

There are many things that can and will be done to make lunar spacecraft operations less demanding than during the Apollo program. The most crucial step is to emphasize operational simplicity as a vehicle design goal. Advanced computer technology should help reduce crew work load with features such as automatic failure detection and recovery. Programs to enhance the space shuttle and develop other space systems should provide an experience base to support this area of lunar vehicle design.

One aspect of lunar transportation that should become less demanding is descent and landing on the lunar surface. Since the vehicles will be flown repeatedly to the same site, the terrain will be well known, radio and visual navigation aids can be provided, and landing areas can be prepared. On the other hand, one of the consequences of reusable vehicles may be a larger range of mission abort options, including the option of aborting an ascent from the Moon. More abort options could significantly complicate the crew training task.

Vehicle Servicing

The vehicle servicing requirements for lunar spacecraft may include the following functions: refurbishment of engines, refueling or replacement of propellant tanks, resupply of consumables, repair or replacement of failed components, inspection of aerobrakes, and testing of propellant tanks. Of these activities, engine maintenance is likely to be the most time consuming. The most efficient method of handling engine maintenance will probably be to design engines for easy removal and replacement as a single unit. Engines can then be serviced in a large pressurized volume in space or returned to Earth.

In general, every effort must be made to design systems for easy servicing, either with robots or from within pressurized volumes. Maintenance by crews in pressure suits should be a last resort.

Payload Integration

The most important payload for the lunar transportation system will be people, but, if space shuttle experience is an indication, readiness of human crews will not be the most critical item in a lunar transportation system schedule. Readiness of cargo is likely to be a major concern at a transportation node because of the need to coordinate the delivery and preparation of a vehicle, its propellant, and the payload within very tight launch window

constraints. Schedules for delivery of payloads will be subject to all the factors that can delay a launch from Earth.

One way to reduce sensitivity to delivery schedule problems is to make the payload integration process as simple as possible. That can be done by minimizing the interfaces between the payload and vehicle. This will be especially important in later phases of lunar operations when payloads may have to be moved from an orbital transfer vehicle to a lunar landing craft after arrival in the lunar vicinity. Payloads on unmanned missions should require only a simple mechanical connection with the vehicle. Crew modules will require some data connections and possibly electrical connections, but these should be minimized. One option for data connections is to use radio frequency or optical links to avoid a physical connection. The complicated interface engineering and compatibility analysis that is performed in the space shuttle payload integration process must be avoided by design.

Unloading cargo and people on the lunar surface is also an important part of payload integration. Trade studies must be done to determine which unloading devices should be part of the landing craft and which should be provided at the surface base.

Flight Control

The goals for flight control are closely related to those for flight planning. Ideally, there should be no ground facilities for real-time control of lunar spacecraft. Reliance on ground control creates an enormous institutional requirement for development and maintenance of facilities for tracking, communication, data reduction, computation, training, and management. A better allocation of resources would be to develop and verify onboard adaptive guidance, navigation, and flight control systems. Along with this, automated systems could be developed to monitor spacecraft systems with little or no additional workload placed on a human crew. In addition to the direct benefits, automated systems are going to be mandatory for planetary spaceflight and the lunar transportation system would provide a good test bed for their development.

CONCLUSIONS

Studies of space shuttle development and operations provide several major lessons that could benefit the effort to design an efficient lunar transportation system. The program goals and the transportation scenario must be defined as part of the conceptual design process. There must be a strategy for evolutionary growth of the system, including a plan for the transition from development to operations. An important part of the transition plan is the definition of what is required to operate the system without an ongoing vehicle and flight analysis effort. Vehicle maintainability has to be weighed against flight performance in design trade studies, and early investments should be made in technology development.

There are two major challenges in the development of an operational lunar transportation system. One is the development of high-performance vehicle systems, such as engines, which do not require extensive and frequent maintenance. The other challenge is to develop and validate highly adaptive onboard guidance, control, and flight planning systems. Onboard autonomy will require a large initial investment, but it will help to control operational costs, and its development will be mandatory for future interplanetary spaceflight. The long-range goal for transportation development should be to design systems that operate themselves.

Transportation system design will continue as part of ongoing lunar base studies. The effort to understand the lessons of past and current space vehicle programs will also continue with the goals of avoiding pitfalls and building on success.

REFERENCES

JSC (1988a) *Design Goals and Technology Requirements for Future Launch Systems, Draft Report.* Advanced Programs Office, NASA Johnson Space Center, Houston.

JSC (1988b) *Lunar Base Systems Study Review.* Advanced Programs Office, NASA Johnson Space Center, Houston.

Petro A. J., ed. (1986) *Space Shuttle Directions.* JSC-20939, Advanced Programs Office, NASA Johnson Space Center, Houston.

Petro A. J., Stecklein J. M., and Bass L. E. (1987) *Lunar Transportation Study, First Quarterly Report.* Advanced Programs Office, NASA Johnson Space Center, Houston.

Woodcock G. R. (1985) Mission and operations modes for lunar basing. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 111-124. Lunar and Planetary Institute, Houston.

ELECTRIC PROPULSION FOR LUNAR EXPLORATION AND LUNAR BASE DEVELOPMENT

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Using electric propulsion to deliver materials to lunar orbit for the development and construction of a lunar base was investigated. Because the mass of the base and its life-cycle resupply mass are large, high specific impulse propulsion systems may significantly reduce the transportation system mass and cost. Three electric propulsion technologies [arcjet, ion, and magneto-plasma-dynamic (MPD) propulsion] were compared with oxygen/hydrogen propulsion for a lunar base development scenario. Detailed estimates of the orbital transfer vehicles' (OTVs') masses and their propellant masses are presented. The fleet sizes for the chemical and electric propulsion systems are estimated. Ion and MPD propulsion systems enable significant launch mass savings over O_2/H_2 propulsion. Because of the longer trip time required for the low-thrust OTVs, more of them are required to perform the mission model. By offloading the lunar cargo from the manned O_2/H_2 OTV missions onto the electric propulsion OTVs, a significant reduction of the low Earth orbit (LEO) launch mass is possible over the 19-year base development period.

NOMENCLATURE

ACS	Attitude control subsystem
ASE	Advanced space engine
CDS	Command and data subsystem
H_2	Hydrogen
I_{sp}	Specific impulse (lb_f-sec/lb_m)
L/D	Lift-to-drag ratio
LEO	Low Earth orbit
LLO	Low lunar orbit
L-2	Earth-Moon libration point 2
MPD	Magneto-plasma-dynamic
MSFC	Marshall Space Flight Center
NEP	Nuclear-electric propulsion
NH_3	Ammonia
NSO	Nuclear-safe orbit
OTV	Orbital transfer vehicle
O_2/H_2	Oxygen/hydrogen
PPU	Power processing unit
RCS	Reaction control susbsystem
Telecom	Telecommunication subsystem
TVS	Thermodynamic vent system
T/W	Thrust-to-weight
VCS	Vapor-cooled shield
Xe	Xenon
ΔV	Velocity change (km/sec)

INTRODUCTION

To construct a lunar base, large propulsion systems to transport personnel and material to the Moon are required. Many missions are planned, including preliminary exploration of lunar base sites, lunar base construction missions, and base maintenance missions. The choice of the types of lunar transfer propulsion systems is dependent upon the factors of cost, trip time, safety, and capability. A mixed fleet of systems that can fulfill all the lunar base transportation system needs is a potential optimum or "best" solution.

In finding the best way to develop a lunar transportation system, a mix of several propulsion systems to be used for both unmanned cargo missions and manned assembly crew missions can be considered. Three electric propulsion options are available to perform complementary missions with the baseline chemical propulsion systems for the lunar base transportation missions. Each of these electric propulsion options is capable of delivering cargo to low lunar orbit (LLO). Because of the low thrust produced by the electric orbital transfer vehicles (OTVs), the lunar-transfer trip time is long: 100-300 days. Personnel are not transported on these OTVs; they are delivered with the high-thrust chemical propulsion OTVs. By offloading the cargo onto the low-thrust OTVs, the cost of constructing a lunar base, as measured by the initial mass required in LEO, may be significantly reduced.

LUNAR EXPLORATION AND THE LUNAR BASE

A lunar base is being considered as a possible major NASA initiative (*Ride,* 1987). At the base, a large number of scientific experiments will be conducted. Using lunar industrial processes to produce oxygen from the lunar soil is also a planned base activity (*Carroll,* 1983).

To construct and maintain the lunar base, a large number of people and a large mass of material must be delivered to the Moon. Table 1 provides the payload massses for the base (*Eagle Engineering,* 1984). The construction phase is 19 years. Prior to the lunar base delivery to the Moon's surface, a number of exploratory missions are needed. Small communication satellites and surface rovers will be placed into lunar orbit and on the surface respectively.

TABLE 1. Lunar base payload masses.

Payload	Mass (kg) Up	Down	Number of Payloads
GEO-Mapper	500	0	2
Surface Rover	4,000	0	6
Heavy Delivery	35,000	0	10
Base Set-Up and Ops	32,000	6,000	8
Ops and Supply	19,500	7,000	3
Heavy Delivery	22,500	1,000	16
Ops (+2T)	12,500	7,500	4
Resupply	19,500	7,500	3
Crew Rotation	14,500	7,500	3
L-2 Communications Satellite	2,000	0	1
Resupply	22,000	10,000	15
Crew Rotation	17,000	10,000	7

After the initial surface reconnoitering, a site will be selected for the base. A series of unmanned payload delivery missions is required for the base construction. Over the 19-year construction phase, a total of 1,602,500 kg is delivered to LLO.

In constructing a lunar base, the ability to continuously deliver large masses to lunar orbit will be essential. Using chemical propulsion, the cost of placing these masses in Earth orbit and finally in lunar orbit will be high. Figure 1 compares the mass of a chemical O_2/H_2 OTV to OTVs using ion and magneto-plasma-dynamic (MPD) propulsion. This analysis uses a 35,000-kg payload delivered to LLO from LEO; the OTV with no payload is returned to LEO. An O_2/H_2 OTV using a 475-lb_f-sec/lb_m specific impulse (I_{sp}) requires a propellant mass of 77,450 kg to perform this mission. With ion or MPD propulsion at a 5000-lb_f-sec/lb_m I_{sp}, the propellant mass is reduced to 13,300 kg and 12,250 kg, respectively. These electric propulsion systems can reduce the propellant mass needed by 64,150 and 65,200 kg per flight.

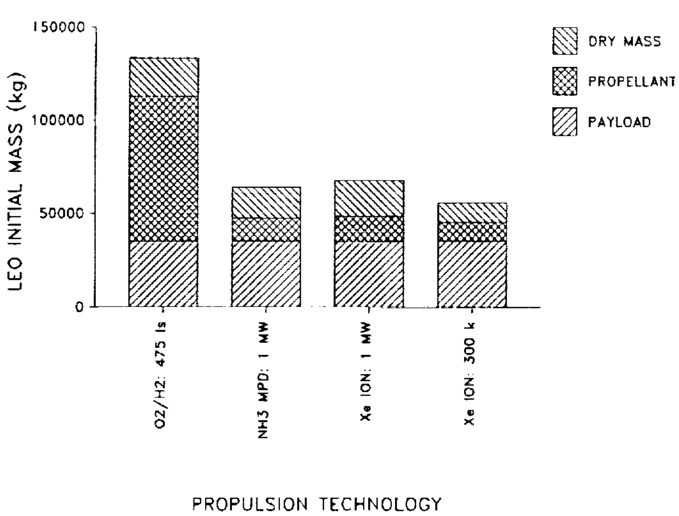

Fig. 1. Propulsion system mass comparison.

MISSION ANALYSIS

Mission analyses for each of the electric propulsion OTVs and the chemical propulsion OTVs were conducted. The ΔV for the various OTV maneuvers and their impact on the lunar transfer mission are discussed. The effect of nodal regression on the launch of the OTV payload delivery missions is described. These results are used to compute the trip times and the propellant mass for the various orbit-transfer maneuvers.

Propulsion requirements are driven by the orbit-transfer and the orbit nodal-regression ΔVs. Both low-thrust orbit transfers, high-thrust all-impulsive orbit transfers, and aerobraked orbit transfers are addressed. Nodal regression of the Moon's orbit constrains the servicing interval and the spacecraft departure time selection. Parametric analyses describing the minimization of the nodal-regression ΔV for a lunar orbit transfer are presented.

Mission ΔV

The primary ΔV for the lunar missions is the orbit-transfer ΔV. In the transfer from LEO to LLO, the OTV departs from LEO, a 28.5° inclination, 500-km-altitude orbit; the LLO is a 100-km-altitude, 0.0° inclination orbit. Table 2 provides the ΔVs used for the low-thrust and the high-thrust orbit transfers. The one-way high-thrust ΔV for the Earth departure (with no gravity losses) is 3.058 km/sec.

Gravity Losses

Gravity losses associated with the medium thrust-to-weight (T/W) nonimpulsive firings of the chemical propulsion systems were estimated using (*Robbins,* 1986)

$$\Delta V_{gl} = (\mu/24\, r_o^3)\, \Delta V_I\, t_b^2\, [1 - (\mu/(r_o(V_o + \Delta V_I)^2)]$$

where ΔV_{gl} = gravity-loss ΔV penalty (km/sec); μ = Earth gravitational constant = 398,601.3 km^3/sec^2; r_o = radial orbital distance (km); ΔV_I = impulsive ΔV (km/sec); t_b = thruster firing time (sec); and V_o = inital elliptical orbit velocity (km/sec).

For the chemical OTVs, the gravity losses were minimized by using a T/W of 0.1. The OTV thrust level was fixed at 133,340 N (30,000 lb_f); by selecting the high thrust level, the LEO-LLO ΔV_{gl} was less than 100 m/sec.

To reduce the high-thrust LEO-return ΔV, aerobraking is used. A 90-km entry altitude is assumed; the OTV provides the circularization ΔV from the 90-km aerobraking altitude to the 500-km Earth-return altitude. The OTV would then rendezvous with the space station. During the aerobraking maneuver, no orbit plane change occurs; the OTV delivers any required plane change

TABLE 2. Lunar orbit transfer ΔV.

OTV Type and Maneuver	ΔV (m/sec)
High Thrust	
LEO Departure and Trajectory Correction	3153
LLO Insertion	900
LLO Departure	900
Trajectory Correction and LEO Insertion	250
Low Thrust	
LEO Departure and LLO Insertion	8000
LLO Departure and LEO Insertion	8000

prior to the atmospheric entry. For an aerobraked return, including the circularization burn and the LLO departure, the ΔV is 1.093 km/sec. An added 57 m/sec is provided for gravity losses and the trajectory correction maneuvers between LLO and LEO.

With the low-thrust case, the ΔV is 7.80 km/sec (*Carroll*, 1983). For this study, a 200-m/sec ΔV was added for nonminimum energy LEO-LLO transfers; the total one-way ΔV is therefore 8.00 km/sec.

Servicing Requirements

In planning the OTV departures, the nodal regression of the LEO and the Moon must be considered. Nodal regression is the rotation of an orbit's line of nodes. This rotation is caused by the Earth's oblateness or nonsphericity. If the OTV departure time is not carefully planned, a large ΔV penalty may be incurred.

Figure 2 provides the LEO-Moon nodal-regression ΔV, using the method in *Edelbaum* (1961) and *Palaszewski* (1986). The ΔV is plotted against the servicing interval. A judicious selection of the orbit transfer departure time can significantly reduce the required OTV ΔV. Every 55 days, the nodal regression ΔV reaches a minimum. In this analysis, the OTV departures coincide with this minimum nodal ΔV.

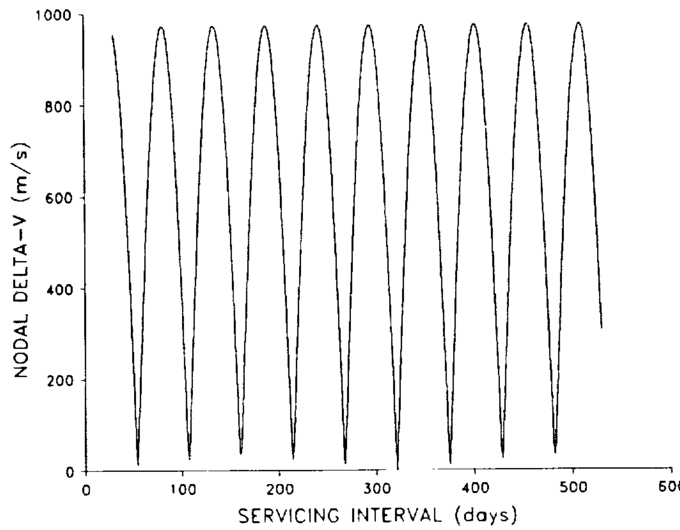

Fig. 2. Nodal regression ΔV.

Nuclear-Safe Orbit

A nuclear OTV may require a minimal deployment altitude called a nuclear-safe orbit (NSO). An NSO is an orbit that precludes a reactor reentry in less than 300 yr (*Buden and Garrison*, 1984). No official NSO altitude has been determined; a 500- to 1000-km altitude range is possible. If the NSO altitude is higher than the space station altitude, an added chemical-propulsion OTV, a nonnuclear electric propulsion OTV, or an orbital maneuvering vehicle (OMV) may be required. This OMV or OTV will deliver the nuclear OTV to its NSO and service it after every mission. In this study, a 500-km NSO was assumed. Therefore, no added servicing OMV or OTV was required.

Flight Performance Reserves

An added ΔV is provided for reaction control and flight performance reserves. During the rendezvous with the space station and for rendezvous in lunar orbit, a high-thrust reaction control subsystem (RCS) will be required; docking disturbances created by the contact of the OTV with the station must be negated. For each orbit transfer, there is also some variation in the main propulsion system performance. This RCS will provide the flight performance reserves if it is necessary to augment the OTV main propulsion system. In each OTV design, an O_2/H_2 RCS is provided; it is designed to deliver a 100-m/sec ΔV to a 45,360-kg (100,000-lb_m) initial-mass spacecraft. A 45,360-kg mass was chosen as a representative OTV wet mass. Using a 450-lb_f-sec/lb_m I_{sp}, the RCS propellant mass required is 1016 kg.

PROPULSION OPTIONS AND PROPULSION TECHNOLOGIES

OTV Designs

Cryogenic O_2/H_2 OTVs are being considered for lunar missions (*Ride*, 1987; *Carroll*, 1983; *Eagle Engineering*, 1984; *General Dynamics*, 1985; *Boeing*, 1986; *Martin Marietta*, 1985). Electric propulsion options considered in this study were the thermal-arcjet, the MPD, and ion propulsion. Both expendable and reusable OTVs are being considered for the resupply of a lunar base. In this study, only reusable OTVs were analyzed.

Chemical OTV. Figure 3 depicts the chemical OTV design (*Park*, 1987). A conical lifting-brake aerobrake is assumed. This OTV design has a low lift-to-drag (L/D) ratio: 0.1-0.2. Each of the OTV main engines retracts behind a thermally protected door in the aerobrake. To prevent reentry wake impingement on the payload during aerobraking, a 50-ft-diameter aerobrake was assumed (*General Dynamics*, 1985).

Nuclear-electric OTV. A nuclear-electric OTV is shown in Fig. 4 (*Jones*, 1986). In this design, the nuclear reactor is separated from the payload and the propulsion system by a boom. This separation of the payload and the reactor is required to minimize the radiation effects on the payload. The OTV will fly in a gravity-gradient-stabilized mode; the most massive part of the OTV will point toward the Earth with the boom aligned with the Earth gravity vector. For this OTV, inert gas Xe-ion, NH_3 MPD, and H_2 arcjet thrusters were considered.

Solar-electric OTV. A solar-electric OTV is depicted in Fig. 5 (*Aston*, 1986). A 100- and a 300-kW solar array are assumed. As with the nuclear-electric OTV, the ion-electric propulsion system

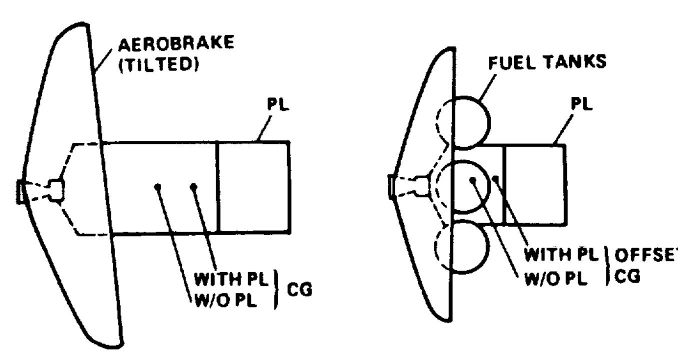

Fig. 3. Chemical OTV.

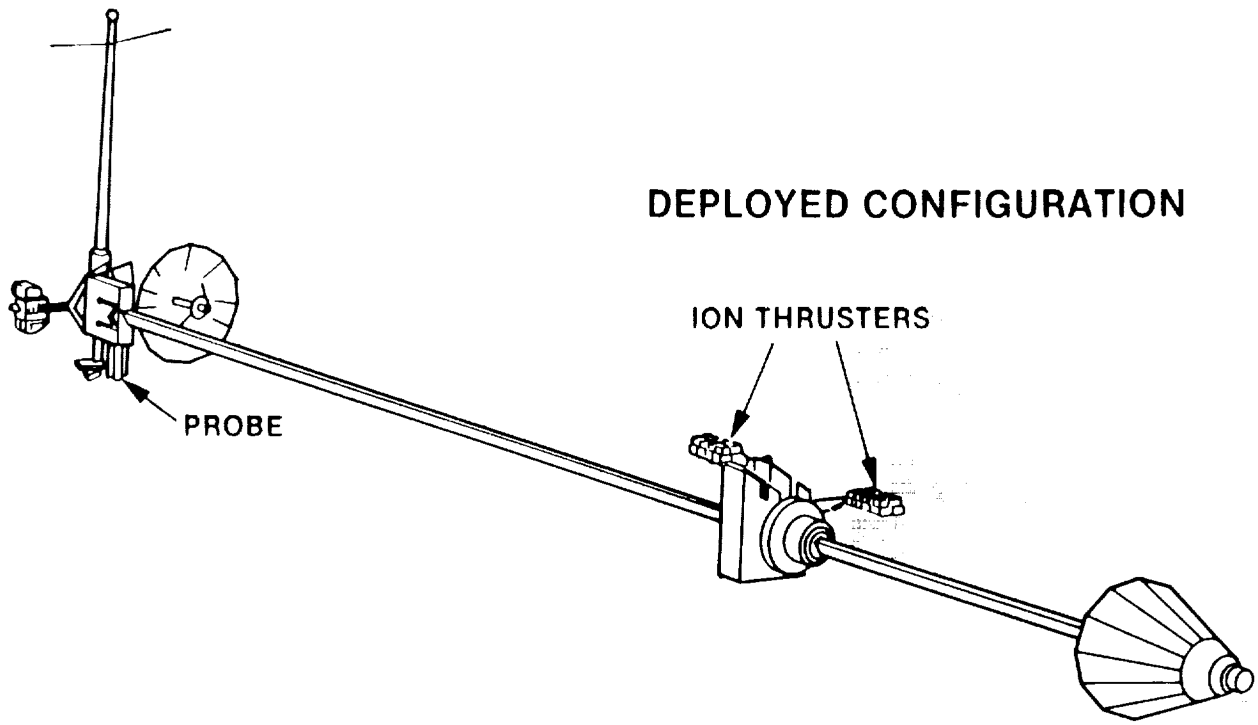

Fig. 4. Nuclear-electric OTV.

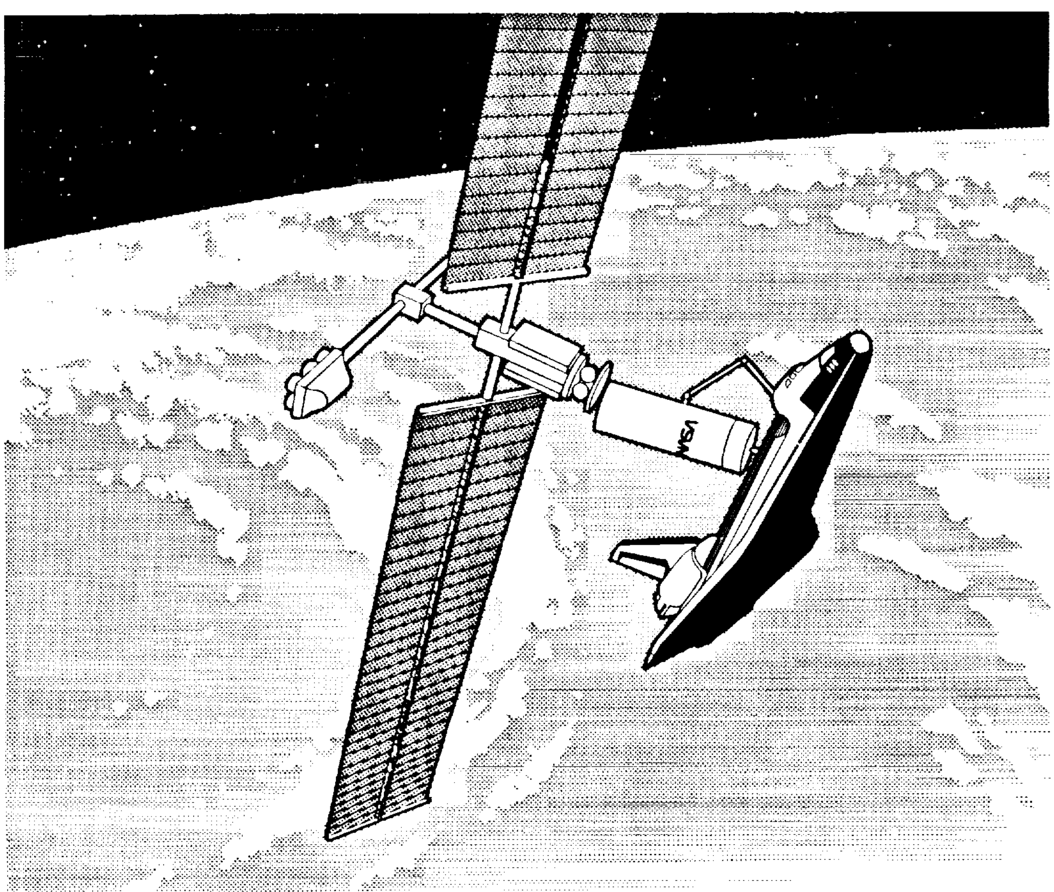

Fig. 5. Solar-electric OTV.

uses an inert gas Xe propellant. Similar OTVs were designed for the arcjet propulsion system; H_2 propellants were assumed for these OTVs. No solar-powered MPD systems were considered.

PROPULSION SYSTEM DESIGN

Main Engine and Thrusters

Table 3 shows the propulsion performance of the OTV designs. A 475-lb_f-sec/lb_m O_2/H_2 I_{sp} was assumed (*General Dynamics,* 1985). Each chemical OTV uses a 133,350-N thrust level. For the O_2/H_2 OTV the advanced space engine (ASE) mass and performance were assumed (*General Dynamics,* 1985).

For each of the low-power electric propulsion systems, a 50-kW power input per thruster was assumed. For a 100-kW propulsion system, three thrusters are required; one thruster is provided for redundancy. The 300-kW OTV needs six thrusters and two redundant thrusters are provided. At a 1-MW power level, a minimum of 20 50-kW thrusters are needed. The propulsion system complexity and mass will be reduced if a higher-power-level thruster is available.

The propulsion system mass reductions for OTVs with higher thruster power levels were investigated; a 500-kW ion thruster design for the high-power 1-MW Xe-ion OTV was assumed. One redundant thruster is provided on the OTV. For the 1-MW H_2-arcjet OTV, a 200-kW thruster power level was assumed. The OTV operates with five thrusters; three thrusters are added for redundancy. Each MPD thruster uses a 1-MW power level. For a 341-day trip time, five thrusters will be fired in series to deliver the total propulsion system firing time; three thrusters are added for redundancy.

TABLE 3. Propulsion system performance.

System	I_{sp} (lb_f - sec/lb_m)	Input P (kW)	Efficiency (Thruster and PPU)
O_2/H_2	475	n/a	
H_2 Arcjet	1,500	50	0.49
H_2 Arcjet	1,500	200	0.49
Xe Ion	5,000	50	0.72
Xe Ion	5,000	500	0.72
Xe Ion	20,000	500	0.89
NH_3 MPD	5,000	1000	0.50

Aerobrake

The aerobrake mass is 15% of the aeroentry mass (*Eagle Engineering,* 1984). Included in the aeroentry mass is the OTV dry mass, the payload that is returned to LEO, the propellant that is on board the OTV for the circularization firing after aerobraking, and the aerobrake itself. For the baseline O_2/H_2 system, the aerobrake mass is 2973 kg.

Electric Power System

For the chemical OTVs, a fuel cell-based power system was assumed (*Martin Marietta,* 1985). This power system provides a 0.33-kW power level for a 6- to 10-day mission. Power systems for the electric OTVs were solar arrays and nuclear reactors. Power levels of 100 kW to 1 MW were considered. An end-of-life 7-kg/kW solar array specific mass was assumed for the 100- and the 300-kW arrays (*Aston,* 1986), and for the 1-MW reactor, a 5-kg/kW and a 10-kg/kW reactor specific mass were assumed (*Sercel,* 1987). The reactor mass includes the OTV boom mass (the boom separates the payload from the reactor and isolates it from the reactor's radiation).

A solar array will experience radiation degradation as it passes through the Earth's Van Allen radiation belts. New solar-cell technologies, such as amorphous silicon, may significantly reduce the cell radiation damage (*Aston,* 1986). In the solar-electric OTV analyses, a 1-kg/kW effective mass penalty accounts for the radiation degradation to the array; an array with no degradation has a specific mass of 6 kg/kW. A 14.3% degradation margin is therefore included. After the array has degraded 14.3%, the array blanket would be replaced.

Power Processing Units

Power processing units (PPUs) for the electric propulsion systems used state-of-the-art power electronics and dc/dc-converter technologies (*Palaszewski,* 1986). H_2-arcjet-propulsion PPU specific masses of 0.11 kg/kW were assumed (W. Deininger, personal communication, 1986). The ion-propulsion PPU specific mass was 0.78 kg/kW (G. Aston, personal communication, 1986) for the 1-MW ion and MPD OTV and 3.1 kg/kW (G. Aston, personal communication, 1985) for the 100-kW and 300-kW OTVs.

At high power levels, the arcjet, MPD, and ion PPU specific mass will be reduced. The PPU is composed of a power-level-dependent mass and a fixed mass that is independent of the PPU power level. For a low power level, the fixed mass is a large fraction of the total PPU specific mass. At higher power levels, the PPU fixed mass is unchanged; with a high power level, the sum of the PPU fixed mass and the power-level-dependent mass correspond to a small total PPU specific mass.

Feed System Design

Detailed propulsion feed-system mass-scaling equations for all the OTVs were derived. Each feed system includes a propellant tank, pressurization system, and feed components to provide propellant to the OTV thrusters. Figure 6 provides an Xe feed system schematic. In each feed system, a 10% ullage was assumed. Each liquid propellant tank accommodates a propellant residual mass of 1.5% of the total of the usable propellant mass and the residual propellant mass. For the supercritical propellant, a 100-psia final tank pressure was assumed; for a 4500-psia initial tank pressure, this translates into a residual mass of 1.6% of the total propellant mass.

For the O_2/H_2 system, aluminum propellant tanks with a 30-psia maximal operating pressure were assumed. The tank factor of safety is 2.0; the flange factor is 1.4. Autogenous pressurization is assumed. A 20-psia nominal tank ullage pressure is assumed. A propellant boiloff rate of 0.27 kg/hr for the H_2 and 0.11 kg/hr for the O_2 was assumed. The total boiloff mass for the 10-day mission is 91.2 kg; this mass is carried as a fixed mass penalty on the OTV dry mass.

Included in the electric propulsion module designs are detailed propellant-feed systems (*Palaszewsi,* 1987). H_2 propellants for the arcjet propulsion systems, Xe propellant for the ion system, and NH_3 for the MPD propulsion system were considered. Storage pressures for the propellants are 20 psia for the liquid H_2, 150 psia for the liquid NH_3, and 4500 psia for the supercritical Xe. A 30-psia maximum operating pressure was the H_2 tank design point. For the NH_3 systems, a 170-psia maximal operating pressure was used and the maximal Xe tank pressure was 4500 psia.

A tank-wrapped vaporizer provides propellant to the HN_3-MPD thrusters. The H_2 system uses a thermodynamic vent system/vapor-cooled shield (TVS/VCS) system to reduce propellant boiloff. For both the NH_3-MPD and the H_2-arcjet feed system, the vaporizer and TVS/VCS are linked to the thruster feed system; the vapor or liquid from the thermal control system is conditioned and provided to the propulsion system. Because the Xe is stored as a supercritical fluid, the propellant temperature is noncryogenic: 298 K.

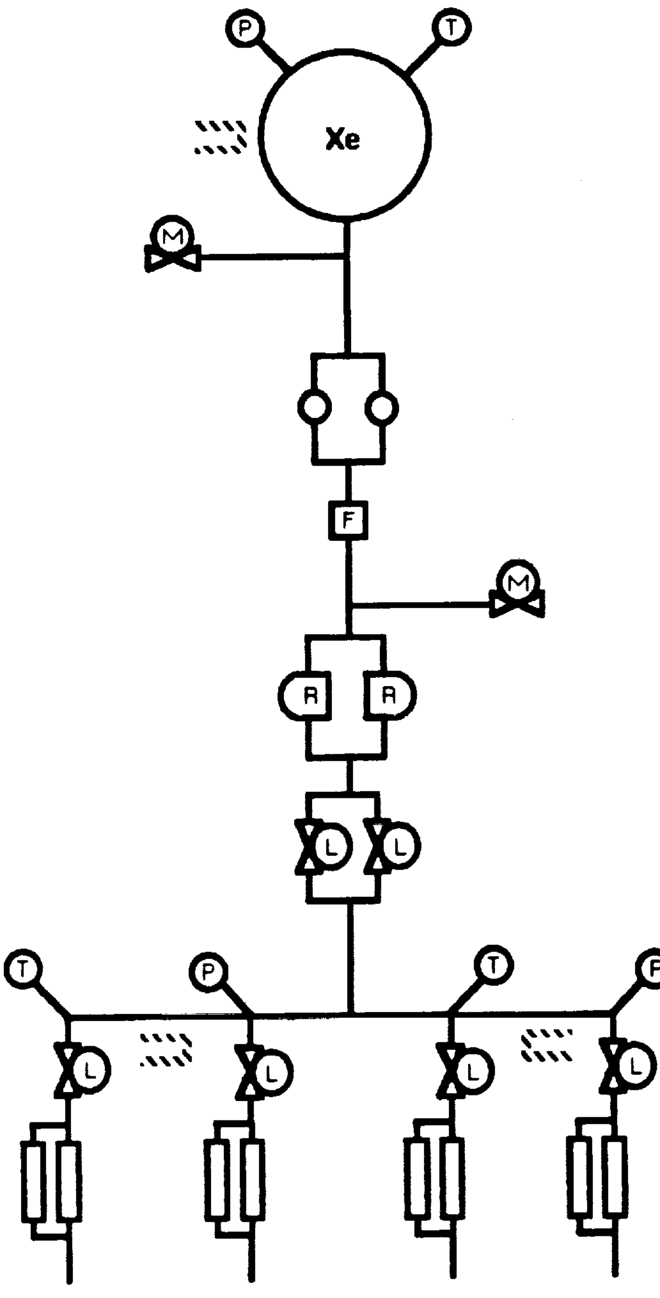

Fig. 6. Xe feed system schematic.

Other OTV susbsystems that are included are the OTV structure, the propulsion-system thermal control subsystem, the attitude control subsystem (ACS), the telecommunication subsystem (telecom), and the command and data subsystem (CDS).

RESULTS

OTV Masses

Table 4 provides a comparison of the 14 OTV dry masses. Each OTV was sized for the worst-case or largest propellant mass. Figure 7 shows the O_2/H_2 OTV sizing analysis; the largest OTV was chosen from this analysis and was used in estimating the mission model propellant mass. Staging of the O_2/H_2 OTV (a two-stage system) is required to reduce the LEO launch mass. The largest O_2/H_2 OTV is sized by two missions: the 35,000-0 mission (35,000-kg payload delivered to LLO and a 0-kg payload returned

TABLE 4. OTV masses.

System	Dry Mass (kg)	M_p, usable (kg)
I_{sp} = 475 lb_f - sec/lb_m		
O_2/H_2	9,506.70	40,000.00
O_2/H_2	5,742.95	14,200.00
I_{sp} = 1500 lb_f - sec/lb_m		
H_2 Arcjet (100 kW)	24,231.33	7,4650.48
H_2 Arcjet (300 kW)	29,701.00	8,5012.40
H_2 Arcjet (1 MW)	46,417.31	11,7882.20*
H_2 Arcjet (1MW)	35,081.76	9,5592.76†
I_{sp} = 5000 lb_f - sec/lb_m		
Xe Ion (100 kW)	6,282.25	8,949.49
Xe Ion (300 kW)	8,848.22	9,939.16
Xe Ion (1 MW)	17,540.14	13,291.54*
Xe Ion (1 MW)	11,766.36	11,064.66†
I_{sp} = 20,000 lb_f - sec/lb_m		
Xe Ion (1 MW)	13,861.50	27,11.00*
Xe Ion (1 MW)	8,709.05	22,73.21†
I_{sp} = 5000 lb_f - sec/lb_m		
NH_3 (1 MW)	14,837.59	12,249.20*
NH_3 (1 MW)	9,529.73	10,202.01†

* Power system mass = 10 kg/kW.
† Power system mass = 5 kg/kW.

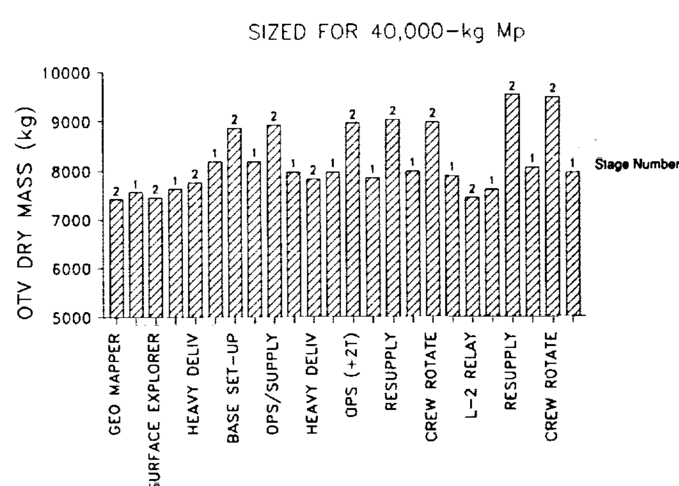

Fig. 7. O_2/H_2 OTV sizing analysis.

to LEO) sizes the 40,000-kg propellant load, and the 22,500-10,000 mission sizes the aerobrake. Figure 8 provides the 1-MW Xe-ion OTV analysis (I_{sp} = 5000 lb_f-sec/lb_m and the power system mass is 10 kg/kW). For the arcjet, the MPD, and the Xe-ion OTVs, the mission that sized the largest OTV is the 35,000-0 payload mission.

Table 5 presents a mass breakdown for the chemical-propulsion OTVs; each OTV has a mass contingency of 10% mass of the burn-out mass. All the OTVs have the same RCS, CDS, ACS, and telecom masses. The chemical O_2/H_2 OTV mass is 9507 kg. Table 6 gives the H_2-arcjet OTV mass summary; Table 7 provides the Xe-ion OTV mass summary. The Xe-ion OTV mass is 17,540 kg and the H_2-arcjet OTV has a 35,082-kg mass.

Propellant Masses

In Tables 8 and 9, the total mission model propellant masses for each OTV are shown. With the O_2/H_2 system, the total propellant mass is 4.7×10^6 kg. The maximum propellant mass delivery is needed in the eighteenth year of the mission model: 6.7×10^5 kg.

Each of the Xe-ion and the MPD OTVs can significantly reduce the total propellant mass required for the lunar base mission model. If the payloads for the base buildup were transported with low-thrust propulsion and the manned cargo crews were delivered separately, a large LEO launch mass savings is possible. Table 8 shows the mass reduction that this operations scenario provided for the total mission model.

The timing of the payload delivery to lunar orbit is also important. With a low-thrust system, the payloads that do not require a manned presence can be sent on ahead of the personnel.

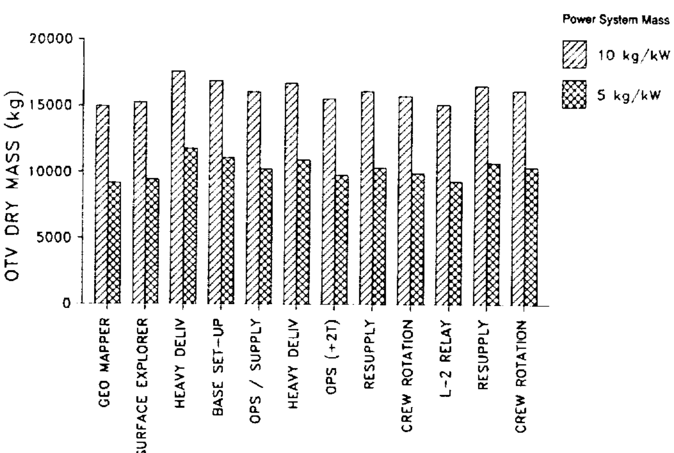

Fig. 8. Xe-ion OTV sizing analysis.

TABLE 5. Chemical OTV mass breakdown.

O_2/H_2 (M_p, usable = 40,000 kg)	
Subsystem	Mass (kg)
Aerobrake	2973.21
Propulsion Main Engines	167.83
Propellant Storage and Feed	1039.24
RCS	1137.65
Power	291.66
Structure	2000.00
Thermal Control	172.72
ACS, Telecom, CDS	251.00
Residuals	609.14
Contingency	864.25
Total	9506.70

TABLE 6. H_2-Arcjet (1-MW) OTV mass breakdown.

H_2 (M_p, usable = 95,592.76 kg)	
Subsystem	Mass (kg)
Propellant Storage and Feed	9,465.20
RCS	1137.65
Power System, PPU, and Thrusters	5,618.92
Structure	4,779.63
Thermal Control	9,184.39
ACS, Telecom, CDS	251.00
Residuals	1,455.72
Contingency	3,189.25
Total	35,081.76

Power system mass = 5 kg/kW.

TABLE 7. Xe-ion (1-MW) OTV mass breakdown.

Xe (M_p, usable = 13,291.54 kg)	
Subsystem	Mass (kg)
Propulsion Main Engines	128.10
Propellant Storage and Feed	3,098.89
RCS	1,137.65
Power System and PPU	10,164.40
Structure	930.42
Thermal Control	18.70
ACS, Telecom, CDS	251.00
Residuals	216.42
Contingency	1,594.56
Total	17,540.14

Power system mass = 10 kg/kW.

TABLE 8. OTV propellant mass requirements.

System	Total M_p (kg)	M_p Delivered* in Year 18 (kg)
$I_{sp} = 475\ lb_f$ - sec/lb_m		
O_2/H_2	4.65×10^6	6.73×10^5
$I_{sp} = 5000\ lb_f$ - sec/lb_m		
Xe Ion (100 kW)	1.66×10^6	7.10×10^4
Xe Ion (300 kW)	1.74×10^6	8.19×10^4
Xe Ion (1 MW)	2.00×10^6	1.19×10^4†
Xe Ion (1 MW)	1.83×10^6	9.42×10^4‡
$I_{sp} = 20{,}000\ lb_f$ - sec/lb_m		
Xe Ion (1 MW)	1.35×10^6	2.31×10^4†
Xe Ion (1 MW)	1.31×10^6	1.84×10^4‡
$I_{sp} = 5000\ lb_f$ - sec/lb_m		
NH_3 MPD (1 MW)	1.92×10^6	1.07×10^5†
NH_3 MPD (1 MW)	1.76×10^6	8.50×10^4‡

* Electric OTV propellant only.
† Power system mass = 10 kg/kW.
‡ Power system mass = 5 kg/kW.

TABLE 9. OTV propellant mass requirements.

System	Total M_p (kg)	M_p Delivered* in Year 15 (kg)
O_2/H_2	4.65×10^6	6.58×10^4
H_2 Arcjet (100 kW)	6.16×10^6	7.32×10^5
H_2 Arcjet (300 kW)	6.96×10^6	8.56×10^5
H_2 Arcjet (1 MW)	10.03×10^6	1.25×10^6†
H_2 Arcjet (1 MW)	7.79×10^6	9.83×10^5‡

* Electric OTV propellant only.
† Power system mass = 10 kg/kW.
‡ Power system mass = 5 kg/kW.

All electric propulsion OTV total propellant mass estimates include 1.19×10^6 kg of O_2/H_2.

A smaller high-thrust vehicle can be used to rendezvous with the cargo modules once they are in lunar orbit.

In each of the total propellant masses for the electric OTVs listed in Tables 8 and 9, a 1.2×10^6-kg O_2/H_2 propellant mass is included. This mass is the total propellant mass required to fly the manned missions in the model; to make the most effective use of electric propulsion, the cargo from the manned sorties is offloaded onto the low-thrust OTVs. In this "remanifesting" of the payloads, the only payloads that fly on the O_2/H_2 OTVs are manned modules for the crew. The crews aboard the chemical OTVs would rendezvous with the payloads delivered by the low-thrust OTVs once they had arrived in LLO.

Each manned mission in the remanifested model is flown with an O_2/H_2 OTV that is sized for a 6000-kg mass flown on a round-trip lunar mission. This mass represents a 5500-kg manned mission module that suppports a four-man crew (*Eagle Engineering*, 1984) and 500 kg for added support systems (power, etc.) for the module. The OTV dry mass is 5743 kg and has a usable propellant load of 14,200 kg.

In the remanifested payload delivery scheme, the payload delivered to LLO by the electric OTVs is the difference between the manned sortie missions listed in Table 1 and the 6000-kg mass for the manned module. For example, for the 32,000-kg up, 6000-kg down mission, the electric OTV would deliver a 26,000-kg up payload and return 0 kg to LEO. An O_2/H_2 OTV performs a round trip with the 6000-kg manned module.

All the missions that are unmanned in the baseline chemical propulsion scenario are conducted using electric propulsion; no payload mass changes are made with these payloads.

All the H_2-arcjet OTVs were rejected because the total mission-model propellant mass for each design exceeds the O_2/H_2 OTV mission-model propellant mass. The relatively low I_{sp} of the arcjet system combined with the high ΔV the system must deliver makes the arcjet system noncompetitive with the chemical propulsion options.

Fleet Sizes

Table 10 compares the fleet sizes for all the OTVs. For the chemical-propulsion OTVs, the minimal fleet size for all scenarios is two OTVs. The chemical propulsion trip times are short; a chemical OTV requires four to five days for a LEO-LLO orbit transfer. In an actual OTV deployment, four to six OTVs would be required; because of hardware failures, damaged OTVs, missed orbit-transfer opportunities due to nodal regression, or other unanticipated problems, a number of added OTVs over and above the minimal fleet size is desirable.

The 100-kW Xe-ion OTVs require very large fleet sizes. The minimum total number of 100-kW electric OTVs required is 47. Array shadowing (passage of the OTV into the Earth's shadow during the orbit transfer) was included. Due to the extended trip times for the low-thrust OTVs, a large number of them are needed. As with the chemical OTVs, additional vehicles will be required to replace OTVs that are being repaired or have been damaged. Because of the large fleet sizes required, these OTVs were rejected from further consideration.

In this analysis, the effect of solar-array shadowing was included; by not including shadowing, the effects of the OTV power level and the shadowing on the OTV trip time are decoupled. If OTV shadowing is included the total fleet size increases by 20%. For example, the 100-kW Xe-ion OTV fleet size if shadowing is ignored is 39 OTVs; with shadowing included, the fleet's size is 47 OTVs.

A 1-MW OTV design can reduce the total fleet size required over the low-power OTVs. Figure 9 compares the 1-MW Xe-ion and MPD OTV fleets ($I_{sp} = 5000\ lb_f\text{-sec}/lb_m$, 10-kg/kW power system. A minimum of seven Xe-ion and nine NH_3 MPD OTVS are needed. As with the chemical OTVs, additional vehicles will be required to replace OTVs that are being repaired or have been damaged.

In Fig. 9, the OTV fleet size varies from year to year. This variation is caused by the differing delivery schedules in each mission model year. For example, in year 10, there are 6 payloads, 12 payloads in year 15, and 11 payloads in year 18.

A high-power OTV can significantly reduce the LEO-LLO trip time; this causes the significant fleet-size reduction for high-power OTVs. Figure 10 provides the trip times for the Xe-ion OTVs. All the trip times are for round trips. For the 300-kW OTVs, the maximum trip time (with shadowing) is 769 days. At the 1-MW power level, the trip time is significantly reduced: 257 days. Figure 11 gives the MPD OTV trip times. A 341-day maximum trip time is required for the 1-MW OTV (10 kg/kW power system).

An important result of these fleet size and propellant mass analyses was that the fleet size of the 300-kW Xe-ion OTVs (5000-$lb_f\text{-sec}/lb_m$ I_{sp}) and the 1-MW Xe-ion OTVs (20,000-$lb_f\text{-sec}/lb_m$ I_{sp}) is comparable. Though the propellant mass required for the

TABLE 10. OTV minimum fleet size requirements.

System	Minimum Fleet Size	Year
O_2/H_2	2	All
$I_{sp} = 5000\ lb_f - sec/lb_m$		
Xe Ion (100 kW)	47	18
Xe Ion (300 kW)	18	15, 18
Xe Ion (1 MW)	7	15, 18*
Xe Ion (1 MW)	6	18†
$I_{sp} = 20,000\ lb_f - sec/lb_m$		
Xe Ion (1 MW)	17	18*
Xe Ion (1 MW)	13	15, 18†
$I_{sp} = 5000\ lb_f - sec/lb_m$		
NH_3 MPD (1 MW)	9	15, 18*
NH_3 MPD (1 MW)	7	15, 18†

* Power system mass = 10 kg/kW.
† Power system mass = 5 kg/kW.

20,000-lb_f-sec/lb_m I_{sp} OTVs was significantly lower than the 5000-lb_f-sec/lb_m I_{sp} OTVs, the fleet size was similar: 18 for the 300-kW system and 17 for the 1-MW system. If the cost of the 300-kW solar-powered OTV were significantly lower than the 1-MW nuclear-powered OTV, the solar-electric OTV may have a cost advantage over the 20,000-lb_f-sec/lb_m I_{sp} OTVs.

Payload Remanifesting

To reduce the total mission-model propellant requirements and the OTV fleet size, variations of the OTV payload delivery capability were investigated. In this sensitivity study, the total payload of the mission model is variable. For missions in the model with multiple payload deliveries and retrievals per year, the total number of LLO missions is variable; for example, if the payload delivered to LLO on each OTV is doubled, the total number of missions flown to LLO is halved. With missions that are flown only once per year, the total mass flown to orbit it multiplied by the payload factor; no remanifesting of the other LLO payloads is addressed.

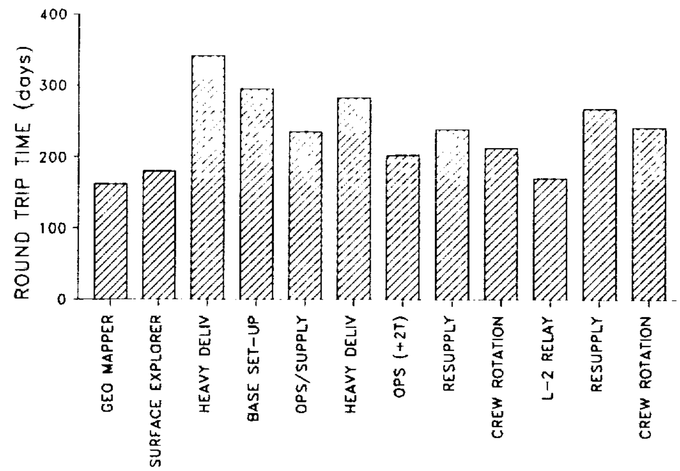

Fig. 11. MPD trip times.

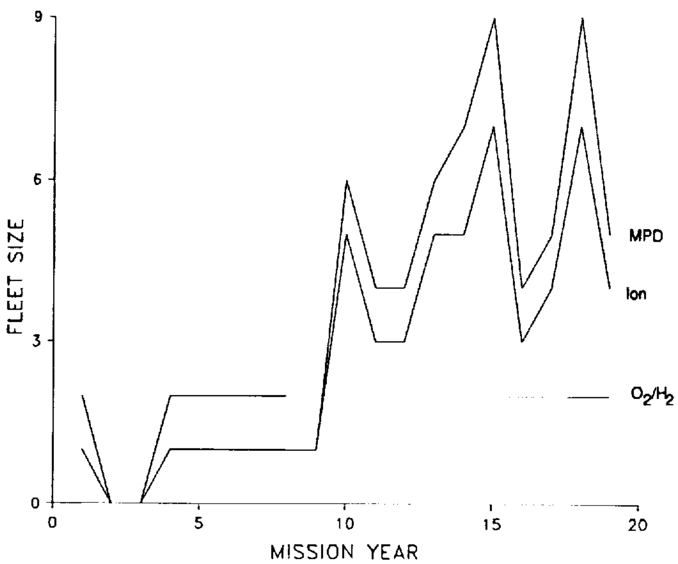

Fig. 9. Fleet sizes: 1-MW ion, 1-MW MPD, and O_2/H_2.

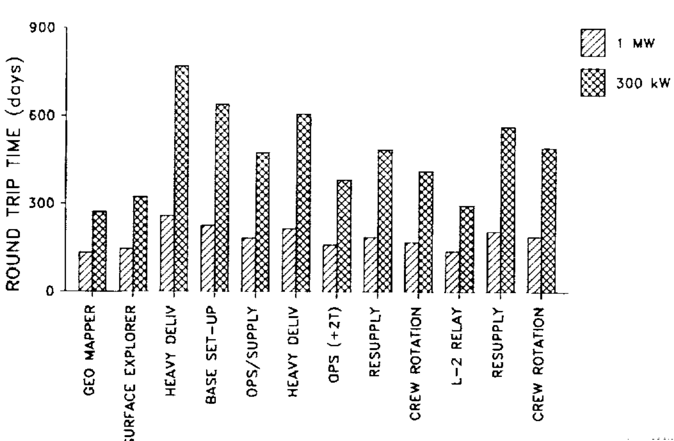

Fig. 10. Xe-ion trip times.

This multiplication of the single-flight LLO payloads results in a significant payload mass increase. In the baseline model, five 22,500-kg up, 1000-kg down, five 22,000-kg up, 20,000-kg down, and one 17,000-kg up, 10,000-kg down flights are planned. For the Xe-ion 1-MW OTVs in year 18, at a payload factor of 2.89, the total delivered payload mass is 3.2×10^5 kg; the nominal year-18 payload mass is 2.4×10^5 kg.

By remanifesting the payload mission model, a large savings in propellant mass is possible. Remanifesting implies that the payload masses of the various missions are not fixed; they can be offloaded onto other OTV flights or combined with other OTV delivery missions. Currently, the mission model payload masses on each flight are not fixed. For example, with the heavy delivery missions, the number of cargo elements delivered on each mission is variable.

In this analysis, the mass of the OTV payload is multiplied by the OTV payload factor. For each payload factor, the required OTV mass was computed using an OTV mass-scaling equation; therefore, the OTV mass is not a fixed number for each payload factor. At each payload factor, the number of OTVs required was computed; an optimum or minimum number of OTVs for any mission model can be estimated. An OTV payload factor ranging from 0.2 to 5.0 was considered.

Figure 12 presents the minimum Xe propellant mass required for 1-MW Xe-ion OTV (power system mass is 10 kg/kW and a 500-lb_f-sec/lb_m I_{sp}) vs. the OTV payload factor. A 1-MW OTV was assumed. At a payload factor of 2.89, a minimum propellant mass is obtained.

In the data from Fig. 12, there are several local minima. The minima are the result of two effects. The first effect is the increase in the OTV size as the payload factor increases. Because the OTV size is increasing, the fleet size for each payload factor is dropping.

As the payload factor increases, the number of OTVs to perform the mission model decreases. However, as the number of OTVs decreases, there is always an integral number of them (there are either 1, 2, or n OTVs, not 2.5). The variation of the number of OTVs with the payload factor is shown in Fig. 13. The fact that the number of OTVs is an integral number and not a smooth function of the payload factor is the second effect.

Combining the effect of the OTV size increase and the fact that the number of OTVs is always an integral number causes the local minima. As shown in Fig. 12, at a payload factor of 2.89, the total propellant mass is a minimum. Another local minimum occurs at a payload factor of 2.42. The increase in propellant mass between the two payload factors is the result of the payload mass increasing on each of the OTVs and the number of OTVs remaining constant (see Fig. 13).

The minimum 1-MW Xe-ion OTV fleet size is shown in Fig. 13. A minimum fleet also occurs at a payload factor of 2.89; for the Xe-ion OTV, the minimum fleet size is 5. This represents a reduction of the total number of OTVs from seven to five. Table 11 provides the optimum payload factor and the propellant savings for the 1-MW MPD OTV and the 1-MW Xe-ion OTV. The Xe-ion propellant mass is reduced by 1.3×10^5 kg. In the MPD case, the minimum fleet occurs at a 2.42 payload factor; the number of OTVs is reduced from nine to six and the propellant mass is reduced by 5.2×10^4 kg to 6.8×10^5 kg.

TABLE 11. Optimal OTV payload factors.

System	Payload Factor	Fleet Savings	Total Propellant Savings (kg)
$I_{sp} = 5000\text{-}lb_f\text{-}sec/lb_m$			
Xe Ion (1 MW)	2.89	2	1.3×10^5*
NH_3 MPD (1 MW)	2.42	3	5.3×10^4*

* Power system mass = 10 kg/kW.

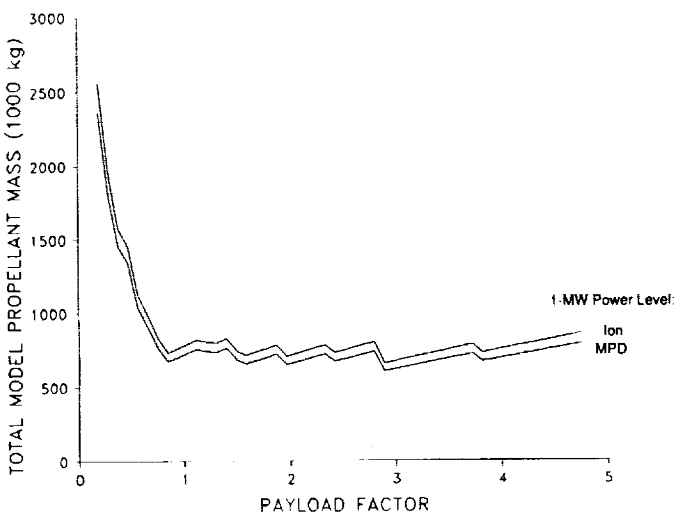

Fig. 12. Propellant mass vs. payload factor. The payload factor is a multiplier for the baseline payload mass. For example, if the payload factor is 2, the total payload mass delivered to LLO and returned to LEO is doubled.

In Fig. 13 the fleet size varies from nine to six MPD OTVs over the payload factor range of 0.8 to 2.4. This variation is caused by the change of the number of OTVs for the differing multiple payload deliveries. In year 15, there are three types of mission: five 22,500-1000 heavy delivery missions, three 19,400-7500 resupply missions, three 14,500-7500 crew rotation missions, and one 2000-0-kg L-2 communications satellite mission. If the payload factor is 1.5, the total number of OTVs is $4 + 2 + 2 + 1 = 9$; similarly, for a payload factor of 2.4, the total number is $3 + 1 + 1 + 1 = 6$.

Large propellant savings are possible with payload remanifesting. A large added payload-mass delivery capability to LLO also results. To achieve this large savings and added mass delivery capability, however, each mission model must have a large number of multiple-flight-per-year missions; in year 18 there are two sets of five heavy delivery and five crew rotation missions. If these multiple sets of missions were eliminated from the mission model, the payload remanifesting would not be effective and the propellant savings would drop significantly.

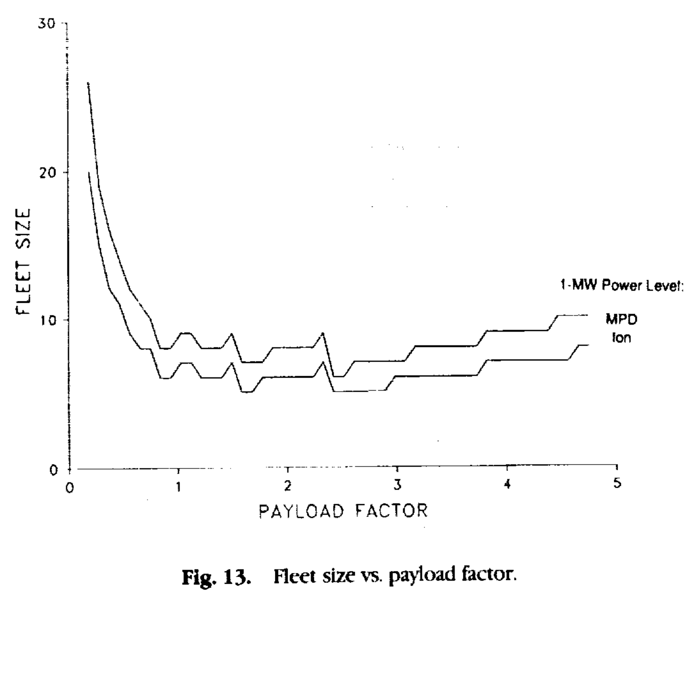

Fig. 13. Fleet size vs. payload factor.

CONCLUSIONS

Both Xe-ion and NH_3-MPD propulsion systems can significantly reduce the LEO launch mass for lunar base development missions. By combining fleets of electric propulsion OTVs and a two-stage O_2/H_2 OTV system, the total propellant mass required to perform a 19-year lunar base transportation model can be reduced by 57-72% (2.7×10^6 kg to 3.3×10^6 kg mass reduction) over an all-chemical propulsion transportation system using aerobraking.

Both solar-electric and nuclear-electric Xe-ion OTVs can enable large propellant mass savings in this transportation system; 18 and 6 OTVs are needed, respectively. Arcjet propulsion systems, using solar arrays or nuclear reactors, are not mass-competitive with chemical propulsion. Nuclear-powered MPD OTVs can also perform the mission model with a minimum nine-OTV fleet size.

The scheduling of the OTV departures to allow rendezvous of the manned chemical OTVs and the electric-propulsion cargo OTVs is required. This scheduling introduces an operational complexity that must be analyzed in more detail.

Payload remanifesting can reduce the total propellant mass required to perform the lunar base mission model. By selecting a heavier payload per OTV and reapportioning the payloads among the resized OTVs, the total transportation system is used more efficiently. This type of optimization is highly dependent upon the traffic model to LLO and LEO.

The mass reduction enabled by electric propulsion translates directly into a large launch-cost reduction. Fewer launch vehicles are required to place the total transportation system mass into LEO. Using Xe or NH_3 propellants in on-orbit storage facilities reduces the total volume of the propellant storage facilities over a cryogenic O_2/H_2 propellant storage depot.

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REFERENCES

Aston G. (1986) Advanced electric propulsion for interplanetary missions. AAS Paper 86-259.

Boeing Aerospace Company (1986) *Orbital Transfer Vehicle Concept Definition and Systems Analysis Study, Phase II Final Review, Technical Summary.* NAS8-36107.

Buden D. and Garrison P. (1984) Space-nuclear power system and the design of the nuclear-electric OTV. AIAA Paper 84-1447.

Carroll W. (1983) *Research on the Use of Space Resources.* JPL Publication 83-36, Jet Propulsion Laboratory, Pasadena.

Eagle Engineering (1984) *Impact of Lunar and Planetary Missions on the Space Station.* Final Report Number 84-85D, Contract No. NAS9-17176.

Edelbaum T. (1961) Propulsion requirements for controllable satellites. *ARS J., 31,* 1079-1089.

General Dynamics Convair Division (1985) *Orbital Transfer Vehicle Concept Definition and Systems Analysis Study, Mid-Term Review, Book 1, Executive Summary, Mission System Requirements.* GDSS-SP-86-011.

Jones R., ed. (1986) *SP-100 Planetary Mission/System Preliminary Design Study—Final Report.* JPL D-2544 (internal document).

Martin Marietta Denver Aerospace (1985) *Orbital Transfer Vehicle Concept Definition and Systems Analysis Study, Volume II, Book 2, Aeroassist, GN&C/Aerothermal Final Review.* NAS8-36108.

Palaszewski B. (1986) Advanced propulsion for coorbiting and polar-orbiting free flyers. AIAA Paper 86-1564.

Palaszewski B. (1987) Geosynchronous Earth orbit base propulsion: Electric propulsion options. AIAA Paper 87-2022.

Palaszewski B. and Krauthamer S. (1986) Arcjet- and resistojet-propulsion power processing. CPIA Conference Proceedings.

Park C. (1987) A survey of aerobraking orbital transfer vehicle design concepts. AIAA Paper 87-0514.

Ride S. (1987) *NASA Leadership and America's Future in Space. A Report to the Administrator.* NASA, Washington, DC. 63 pp.

Robbins H. (1966) An analytical study of the impulsive approximation. *AIAA J., 4,* 1417-1423.

Sercel J. (1987) *Megawatt Nuclear-Electric Propulsion: First-Order System Design and Performance Evaluation.* JPL D-3898 (internal document).

ADVANCED PROPULSION FOR LEO-MOON TRANSPORT: I. A METHOD FOR EVALUATING ADVANCED PROPULSION PERFORMANCE

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We report on a study to evaluate the benefits of advanced propulsion technologies for transporting materials between low Earth orbit and the Moon. A relatively conventional reference transportation system, and several other systems, each of which includes one advanced technology component, are compared in terms of how well they perform a chosen mission objective. The evaluation method is based on a pairwise life-cycle cost comparison of each of the advanced systems with the reference system. Somewhat novel and economically important features of the procedure are the inclusion not only of mass payback ratios based on Earth launch costs, but also of repair and capital acquisition costs, and of adjustments in the latter to reflect the technological maturity of the advanced technologies. The required input information is developed by panels of experts. The overall scope and approach of the study are presented in the introduction. The bulk of the paper describes the evaluation method; the reference system and an advanced transportation system, including a spinning tether in an eccentric Earth orbit, are used to illustrate it.

INTRODUCTION

In the fall of 1986 we initiated an effort to identify and evaluate advanced propulsion concepts for the transportation of materials between low Earth orbit (LEO) and the Moon. We were looking particularly for concepts that would provide a lower-cost alternative to conventional rocketry in supporting scientific work, colonization, and commercial utilization of the Moon, Mars, and perhaps other planets and the asteroids during the twenty-first century.

We identified six tasks to accomplish the aim of the study.

1. Choose a standard mission and a reference configuration as a basis for comparing the performance of advanced configurations. A configuration is here defined as a complete transportation system between LEO and the Moon.
2. Select a small number of the most promising "pure" configurations incorporating a single advanced component or concept.
3. Define criteria by which to evaluate the performance of the configurations.
4. Describe and model each of the configurations to be evaluated.
5. Describe and model, quantitatively insofar as possible, the evaluation criteria.
6. Evaluate all the configurations.

We chose as the objective of the standard mission to carry lunar material ("paydirt") from the lunar surface back to LEO at a specified parametrized annual rate. This objective, although by itself not sufficient to justify the expenditure for a LEO-Moon transportation system, was chosen because it permitted ready and unambiguous comparison of various configurations. We specified that all oxygen for chemical propulsion was to be of lunar origin, and all hydrogen fuel and repair and replacement parts were to be of terrestrial origin. Aerobraking on return to LEO was to be used whenever advantageous; the aerobrake was assumed to be reusable and of terrestrial origin.

The chosen reference case ("Configuration 0") consists of two kinds of vehicles and three stations (*Henley*, 1988). Both vehicles are powered by reaction engines burning terrestrial liquid hydrogen and lunar liquid oxygen. The first kind of vehicle is an orbital transfer vehicle (OTV). Its functions are (1) to carry liquid hydrogen and other terrestrial logistic supplies for lunar activities from LEO to low lunar orbit and (2) to bring back to LEO lunar oxygen for propulsion and lunar material for storage. The OTV carries a reusable aerobrake for the return trip.

The second kind of vehicle is a lunar lander. Its functions are to carry terrestrial logistic supplies from low lunar orbit (LLO) to the lunar surface and to bring excess lunar oxygen and lunar material up to LLO for transfer to the OTV. This vehicle is fitted out with landing gear, and burns a fuel-lean mix to conserve terrestrial liquid hydrogen.

The first station is an "Orbiting Transfer and Staging Facility" (OTSF) in a low Earth circular orbit at 28.5°. Its functions are to store and transfer fuel, payload, spare parts, and repair tools, and to permit docking and berthing of OTVs for repair, refueling, and load transfer. A second, similar facility with comparable functions is in near-equatorial low lunar orbit. Its docks accommodate both OTVs and lunar landers. The third station is a lunar-oxygen production plant, located on the Moon's surface near the equator. The time frame is 2005–2010. It is assumed that a manned lunar base is in existence by then to establish and support this activity, and that a lunar oxygen pilot plant is available for the emplacement and startup of the configurations.

Six configurations, each incorporating one advanced propulsion component, along with appropriate "conventional" components from the reference configuration as required, have been chosen

for detailed evaluation so far. Three of these involve the use of tethers. A seventh configuration, based on solar sails, was eliminated as not suitable for the standard mission in the high-gravity fields prevailing over most of the Earth-Moon trajectory.

Tethers (*Arnold and Thompson,* 1988; *Colombo et al.,* 1974; *Isaacs et al.,*; 1966; *Carroll,* 1985; *Penzo,* 1987) can permit momentum exchange between objects at opposite ends, such as a load and a platform. They become especially attractive if there is two-way traffic, as between LEO and the Moon. In that case, the momentum given up by a platform when loads are picked up and released in one direction can be restored by loads moving in the opposite direction. The three tether configurations are

1. A hanging tether in lunar orbit. In this configuration a very long tether is anchored from a ballasted platform in rather high lunar orbit. It is first deployed toward the lunar surface, so that its tip can rendezvous with a self-propelled lunar load. The tether deployment direction is then changed by 180°, and the load released toward Earth. The procedure is reversed for loads from LEO bound for the lunar surface.

2. A spinning tether in low lunar or Earth orbit. In this concept, a self-propelled load is picked up from below at suborbital velocity, then swung about 180° by the tether before being released. The load thereby gains twice the tangential velocity of the spinning tether. As before, the procedure is reversed for incoming loads.

3. A spinning tether anchored from a massive platform in an eccentric Earth orbit with perigee near LEO ("Configuration 3"). Here an OTV can be picked up in LEO, swung about as above, and released toward the Moon with the same velocity gain. Once again, the platform's momentum is restored upon capture and subsequent release of an OTV traveling in the opposite direction.

Three other configurations incorporating advanced concepts have been examined: laser propulsion, ion-engine propulsion, and mass-driver launch.

In the laser concept (*Kantrowitz,* 1972; R. Glumb, personal communication, 1987) an OTV carries both conventional rocketry and a laser thermal engine. Initially, upon leaving LEO, it is propelled by electromagnetic energy beamed to the vehicle by an Earth-based, high-power infrared laser. The laser beam is focused onto a hydrogen plasma, which is exhausted through a thruster nozzle. The advantage of this concept over a more conventional rocket engine is twofold: The power source (or the oxidant for the hydrogen propellant) need not be carried into space, and the high specific impulse (I_{sp}) derived from this engine results in good fuel economy.

In the ion-engine concept (*Stuhlinger et al.,* 1961) the OTV carries a nuclear electric power source to provide the high-voltage current for ion acceleration. Terrestrial xenon has been assumed as the propellant to be ionized; in practice, lunar argon or oxygen may be more economical. In this configuration, the propellant, as well as the power supply with its massive radiator and radiation shield, must be carried on board (unless solar photovoltaic can substitute for nuclear power), and the thrust is very low, leading to long travel times. Its advantage resides in the very high I_{sp}, leading to manageable propellant loads.

The last configuration incorporates a mass driver (*Chilton et al.,* 1977) for launching packets of lunar material off the Moon's surface. Each packet carries a conventional small propulsion system. Once launched into ballistic orbit, the packets can rendezvous autonomously with an OTSF in low lunar orbit. The launch energy is electrical rather than chemical, and can be provided on the lunar surface either by means of a nuclear power plant or by extensive (but no longer excessively expensive) sheets of amorphous solid-state photovoltaic receptors.

For a more detailed description of these configurations, and of the results of their evaluations, the reader is referred to *Stern* (1989). Some of the results will be stated at the end of this paper.

EVALUATION PROCEDURE

Framework

The evaluation framework comprises two models, a Transportation Model and an Evaluation Model. Two kinds of input are required. Engineering information supplied by technical experts on each configuration serves as input to the Transportation Model. Output of the Transportation Model, along with economic information supplied by evaluation panels, becomes input to the Evaluation Model.

The Transportation Model (*Henley,* 1988) calculates the amount of propellant consumed and the amount of lunar mass delivered to LEO per round trip. From this, one can derive some of the inputs required by the Evaluation Model: the mass payback ratio (MPR), the lunar oxygen plant capacity, and the annual number of round trips required of the OTV and of the lunar lander to satisfy the mission objective. (The MPR is defined as the lunar payload brought down to LEO per tonne of fuel and other supplies that have to be brought up from Earth.) The two models operate independently, and the output of one is fed to the other manually.

The Evaluation Model performs a life-cycle cost analysis of the input data, assuming a venture life and a discount rate. It develops operating costs and capital costs for each advanced configuration, compares these with corresponding figures for the reference configuration, and derives cost-effectiveness measures relative to the reference case from this comparison. Figure 1 shows the flow of information and the relationship between the Transportation Model and the Evaluation Model.

Input data for each configuration come from two sources. Much of the quantitative technical information, such as masses of vehicles and of orbiting or fixed installations, fuel capacities of vehicles, I_{sp} and thrusts of engines, efficiencies and outputs of power sources, and ΔVs supplied by various vehicles or devices, is provided by technical experts or specialists, and becomes input to the Transportation Model. Most of the economic information, whether quantitative or qualitative, is generated by evaluation panels, and is incorporated in the Evaluation Model. This includes estimates of acquisition costs, of technological maturity with its associated development costs and time delays, and of risk of failure and need for repair.

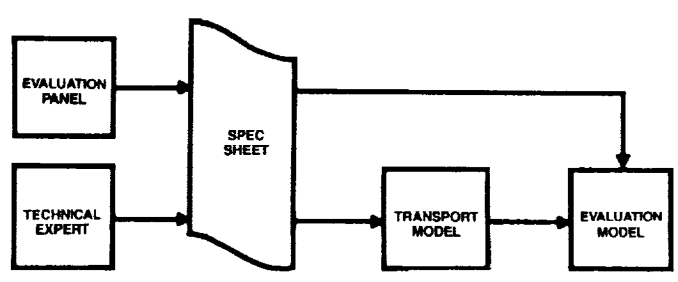

Fig. 1. Information flow in evaluation procedure. This schematic shows the interrelationship between data provided by technical experts and by evaluation panels, and the inputs and outputs of the Transportation Model and the Evaluation Model.

Model Structure

The following ground rules were adopted:

1. Transport from the surface of the Earth to LEO is not considered, but the cost of transport per unit mass between these two nodes is assigned some parametric value $\$\$_0$. All other costs are expressed in terms of this value, insofar as possible.

2. The basic criterion for judging the performance of a given advanced configuration is economic: It is characterized by the payback time or the life-cycle return on investment, which results when the advanced configuration replaces the reference case. The payback time or rate of return is, in most cases, based on a trade-off between savings in operating costs and increased capital costs. The MPR is not a life-cycle measure. It has been retained because it is familiar, and can serve as a coarse sieve to eliminate clearly submarginal schemes.

3. Savings in operating costs are based on improvements in the MPR for an advanced configuration relative to the reference case, corrected for changes in repair and replacement needs.

4. Changes in capital costs take into account transportation as well as acquisition costs of capital installations such as stations and vehicles. Acquisition costs of novel, first-of-their-kind components or subsystems incorporate estimates of their technological maturity. As used in this study, technical maturity is a proxy for the costs of research, development, demonstration, testing, and space qualification associated with the implementation of new technologies.

5. Estimates dealing with repair needs, risk of failure, and technological maturity are quantified with the help of panels of experts. Further details on some of these points are provided below.

As in any financial analysis of a venture, there are two main cost categories in the Evaluation Model: operating costs and capital costs. We shall first deal with these two cost categories by assuming that only transportation costs are important. Then we shall address the complications brought about by inclusion of other cost components, such as repair and maintenance, acquisition and development costs, etc.

To begin with, it may be possible to first weed out totally unsuitable configurations based on operating costs alone, for two reasons. First, a configuration whose operating costs are greater than those of the reference case is almost surely not a viable alternative, since it usually also requires additional capital investments. Second, comparison on the basis of operating costs alone gives an accurate picture of on-going costs, once the start-up investment has been made and becomes a sunk cost. It should be pointed out, however, that even if an advanced "pure" configuration is judged nonviable on this basis, it may still have merit if there are net savings in capital costs—a rare situation. More commonly, it may have merit if its advanced component can be combined symbiotically with other advanced components in a "hybrid" configuration.

The transportation-based operating cost can be derived from the MPR obtained from the Transportation Model. This ratio is defined as

$$\mathrm{MPR} = \frac{\text{payload mass emplaced in LEO}}{\text{mass carried up from Earth to LEO}} \qquad (1)$$

For the reference and advanced configurations chosen in this study, which use lunar-produced liquid oxygen (LLOX), the mass that has to be carried up from Earth to LEO consists mostly of terrestrial hydrogen and of "logistic mass," that is, supplies for operation of the LLOX plant. These have been taken into account in the computation of the MPR carried out in the Transportation Model. Other masses of terrestrial origin for installation of vehicles and equipment and for their maintenance and repair have not been included in the Transportation Model. They will be taken into account in the Evaluation Model, as described below.

From the definition of MPR it is easy to show that the yearly mass savings realized with an MPR > 1 is given by

$$\mathrm{OB} = C_s \cdot (\mathrm{MPR} - 1)/\mathrm{MPR} \qquad (2)$$

where C_s is the annual amount of lunar paydirt to be transported from the Moon to LEO and OB is a yearly operating benefit realized from savings in Earth mass when the paydirt is of lunar rather than terrestrial origin. This equation has the right dependence on MPR: If MPR < 1, there is no benefit, but rather an operating loss associated with using lunar, rather than terrestrial, paydirt. Mass payback ratio = 1 is the break-even point. Once MPR $\gg$ 1, its exact value is of minor importance, since the savings in transportation cost (expressed in mass terms) can never be greater than C_s.

There is a transportation-based capital cost to consider, as well. For equipment to be placed in fixed orbits or space locations, such as LEO, lunar orbit, or the lunar surface, this is the cost of emplacement, expressible in units of mass. For example, for the OTSF in LEO, this cost is just its mass. For the OTSF in low lunar orbit, on the other hand, the mass should be multiplied by a factor >1 to account for the additional propellant load required to accompany the facility.

Transportation-related capital expenditures for vehicles must include an allowance for redundancy. This comes about because of the limited payload capacity and finite turn-around time of each vehicle. For example, delivering an annual lunar payload of 2500 T to LEO in the reference configuration, at about 15 T per round trip, would require approximately 4 vehicles, based on a turn-around time of 8 days. This ignores the relatively narrow biweekly windows available for economical travel between Earth and Moon, which may force a substantial further addition to the fleet.

Capital costs and operating benefits can be combined into a single measure of cost-effectiveness by the well-known device of equating the sum of all future operating benefits, discounted to the present, to the initial investment or capital cost. Two measures derived from this equality are particularly useful. In the first, one assumes a "market" rate of return, *r*, taken at 8% in this paper, and solves for the time, called the payback time, which satisfies the equality. This solution can be expressed in closed form. In the second, one assumes a venture time, fixed at 20 years in this study, and looks for the discount or interest rate, often called the internal rate of return (IRR), which satisfies the equality. This has to be calculated by an iterative procedure, but poses no difficulty for a personal computer. (It should be pointed out that there can be no finite payback time if the annual benefit is less than *r* times the capital cost. By the same token, there can be no IRR if the cumulative benefits over the venture life amount to less than the initial capital cost.)

So far, only transportation costs have been considered. The complications due to other important cost components must now be addressed. These components include the acquisition cost of capital, the R&D costs of developing the technology for an

advanced configuration and bringing it to a state of operational readiness, and the costs of maintenance, repair, and replacement. Their relationship to transportation costs and to the overall measures of cost-effectiveness is represented in Fig. 2.

The acquisition costs of capital for a new configuration can be estimated by experienced space engineers. This is best done by breaking the configuration or system into subsystems and components, many of which are similar to ones already in use or being procured for space applications. The acquisition cost of each component is then estimated in constant-dollar terms. It can be converted to mass units (T) via division by $\$\$_0$ before being added to the transportation cost for that component; conversely, both can be expressed in dollar terms.

The acquisition of a new or advanced component or subsystem, such as the tether-bearing platform and its components in eccentric Earth orbit, or a rocket engine operating at higher-than-conventional oxidizer-to-fuel ratio, poses an additional problem. Clearly, the first embodiment of such a component is much more expensive than the more routine procurement of the fourth or fifth version or copy would be. We have attempted to capture this important cost in a somewhat novel way, summarized here and explained in more detail below.

One can look at this additional cost as a development risk with two consequences: it makes development more expensive than mere acquisition cost, and it entails protracted reduction to practice. The more immature and complex the technology, the greater the cost and the longer the time needed for development. Both cost and time have considerable uncertainty associated with them. In this study, we deal with the cost and time aspects separately. We simplify by assigning their effects to the first embodiment only, rather than distributing them over the first few by means of a "learning curve," as happens in real life.

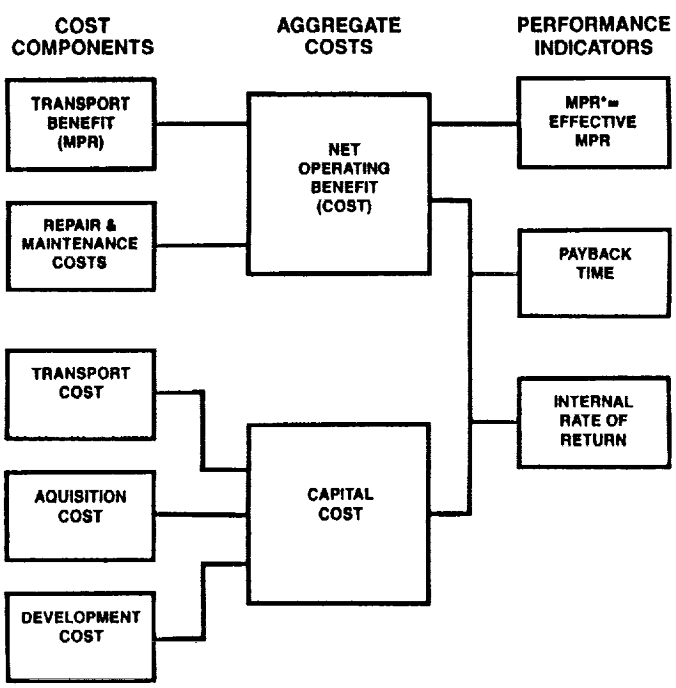

Fig. 2. Interrelationship of cost components. This schematic indicates how the main performance indicators are obtained from various inputs through internal processing in the Evaluation Model.

Briefly, the development cost is estimated by rating the technology readiness at the subsystem or component level; i.e., we list what steps have to be taken to achieve maturity, and evaluate the cost of each step. Development time is arrived at by estimating the time taken to accomplish each step. Delay time is then translated into additional cost (the time value of money) by "discounting" the funds needed for each step forward to the time when operation is to start, with further cost penalties imposed if maturity cannot be expected by the year of initial operation, assumed to be 2005. The effects of technological development and learning are treated deterministically, based on estimates of expected or most probable costs and time delays.

Costs of maintenance, repair, and replacement are aspects of risk of failure in operation, which can be handled as additions to operating costs. After being converted to common (mass or dollar) units, they are summed and subtracted from operating benefits. By equation (2) the revised operating benefit will result in a (generally lower) "effective mass payback ratio" MPR*.

Panel Evaluation Procedure

Three kinds of input were determined by evaluation panels: acquisition costs, technology readiness ratings, and operational risk estimates. Since these inputs play a crucial role in the outcome of the evaluations, they will be described in further detail at this point. Almost all the data were generated at a week's meeting, held in La Jolla, California, July 5-10, 1987.

One panel of from three to five persons was chosen for each configuration. Each panel included one or two technical experts on the particular configuration. The remaining panel members, including the panel chairman, were experts on other facets of space travel, or were technical generalists. Care was taken to balance areas of expertise to include engineering knowledge and some experience with costs, and to preclude advocates from dominating the decisions.

Table 1 lists technology readiness levels that were used as a basis for the ratings. The definitions are those used by NASA's Office of Aeronautics, Exploration, and Technology (OAET).

Each panel was asked, at the outset, to undertake the following:

- determine the level of readiness, L, of the advanced technology of concern to the panel;
- judge the time, Δt, required to advance the level of readiness, one step at a time, all the way to full operational capability; and
- estimate the cost, R, associated with each step, expressed in units of the final (routine) acquisition cost. In this fashion, the question of complexity was finessed.

The results of this preliminary evaluation step are summarized in Fig. 3 for the cost, expressed as an acquisition cost multiplier R, and in Fig. 4 for the time delay Δt, in years. Although there was the expected scatter of estimates in Fig. 3, some common features emerged. None of the technologies was judged to be of level lower than 3. In almost all cases, the cost per step tended

TABLE 1. Technology readiness levels.

Level 1:	Basic principles observed and reported
Level 2:	Conceptual design formulated
Level 3:	Conceptual design tested analytically or experimentally
Level 4:	Critical function/characteristic demonstration
Level 5:	Component/breadboard tested in relevant environment
Level 6:	Prototype/engineering model tested in relevant environment
Level 7:	Engineering model tested in space
Level 8:	Full operational capability

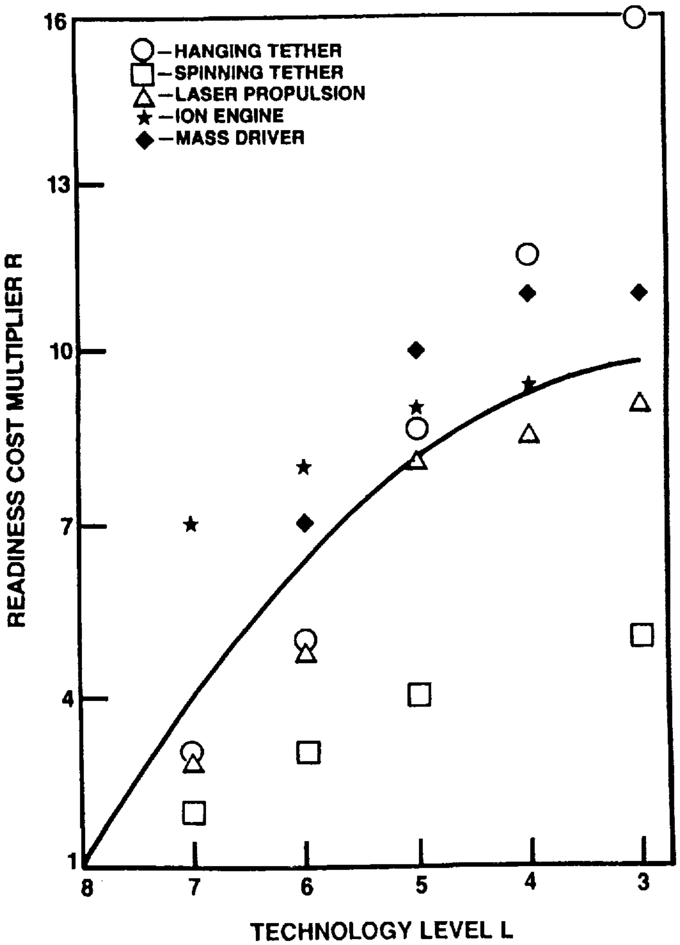

Fig. 3. Technology development cost as function of readiness. The points were obtained from the evaluation panels dealing with five separate technologies. The smooth curve was used as a resulting mean functional relationship.

to increase with increasing level. This is in agreement with the experience that costs escalate as one proceeds from research to development, to prototype laboratory testing, to demonstration. Since the points tended to cluster markedly (except for those of the spinning tether, which were later judged to be too optimistically low), a smooth "eyeball" curve was used as the multiplier for all advanced technologies. The Δt values in Fig. 4 clustered more convincingly about a straight line, which was again used for all technologies. The long delay times for levels 5 and below indicate a high perceived degree of complexity.

Finally, acquisition costs and risks of operation were assigned to each component. Here the judgment of an experienced space engineer on each panel played the key role, since many of the parameters had to be estimated by analogy to present systems and practice. The acquisition cost was intended to reflect the expected "routine" cost of procurement, net of the initial research, development, demonstration, and learning expenditures. Risk of operation was represented in terms of mean expected frequency of replacement or repair, and fraction of total component mass (and dollar value) to be replaced during each repair. For example, it was assumed that in the reference configuration the aerobrake would have to be replaced after 10 missions, but somewhat less frequently in the spinning tether configuration, where it is used to mediate a smaller ΔV.

Model Format

The Evaluation Model was developed on a spreadsheet using the 20/20 (™Access Technology Inc.) software available on UCSD's VAX/VMS operating system. Table 2 displays the input-output section of a run, in this instance, the reference case. Five parameters are inputs from the Transportation Model: MPR, the mass payback ratio based on steady-state payload transportation cost only, here of value 1.31; C_o, the LLOX production required per tonne of payload placed in LEO; C_{10}, the amount of payload put into LEO per OTV round trip; $M_{OTV,F}$, the mass of a fully loaded OTV (as on departure from low lunar orbit toward LEO); and the number of lunar lander trips per OTV trip.

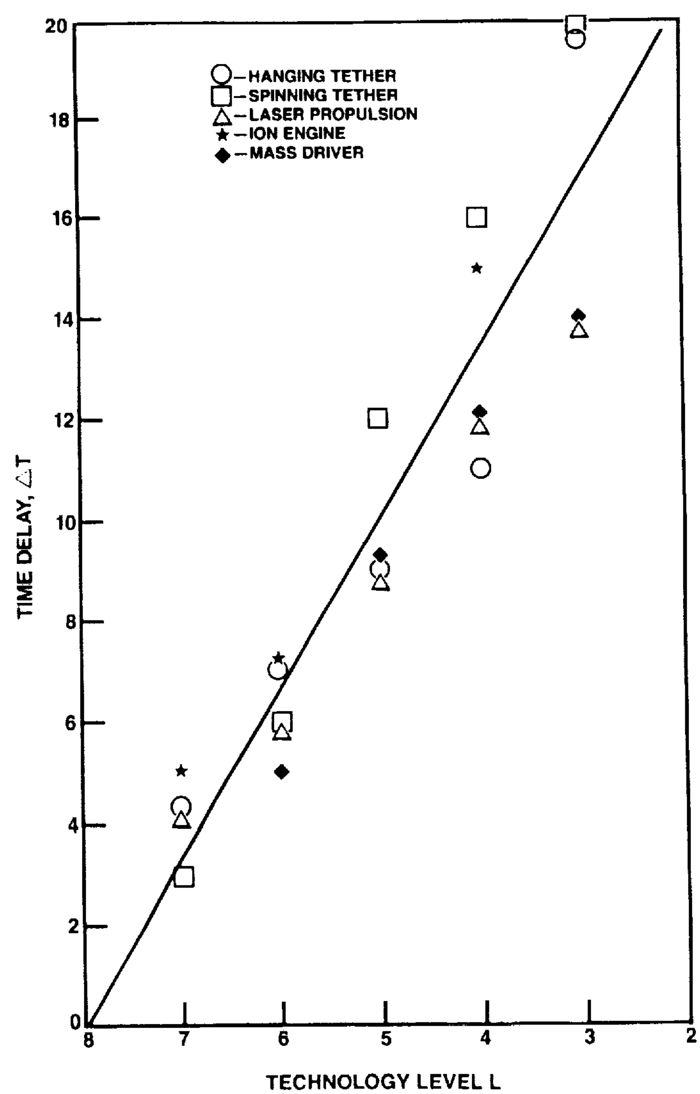

Fig. 4. Technology development time as function of readiness. As in Fig. 3, the points were obtained from the evaluation panels, and the smooth curve represents the adopted functional relationship.

TABLE 2. Input-output section of an evaluation run for the reference configuration.

A	B	C
1		
2	Reference case	Config 0
3		
4		Date: 03-28-88
5		Time: 11:44:40
6		
7	Final results	
8		
9	Operating benefit	-601.398
10	Effective mass payback ratio, MPR*	0.454
11	Incremental capital cost	10755.518
12	Payback time	-11.541
13	Internal rate of return	No Return
14	Corrected LLOX production per LEO T	6.765
15	Corrected LPD into LEO per OTV trip	14.864
16		
17	Parameters from Transportation Model	
18		
19	Mass payback ratio, MPR	1.310
20	LLOX production per LEO T, C_0	6.130
21	LPD into LEO per OTV trip, C_{10}	16.187
22	Mass of loaded OTV, $M_{OTV,F}$	59.236
23	LL trips per OTV trip	7.260
24		
25	Additional parameters	
26		
27	Annual payload mass to LEO, C_s	500.000
28	Cost of 1 T from Earth to LEO, $\$\$_0$	3.000
29	Interest rate, r	0.080
30	Maximum time delay, Δt	13.600
31	F(r)	0.478
32		

This section shows the main inputs, either assumed or obtained from the Transportation Model, and the main aggregate outputs of the Evaluation Model. Three other input parameters can be chosen at will: (1) C_s, the annual payload mass to be emplaced in LEO; (2) $\$\$_0$, the cost (in M$) of bringing 1 T of mass from Earth to LEO; and (3) r, the interest or discount rate. Δt, the maximum time required for implementation of any component of the configuration, is based on the minimum value of L, as supplied by the evaluation panel (see Fig. 4); it is an estimate of how long it takes to implement the configuration. F(r) is a calculated result which, when multiplied by the maximum Δt, approximates the effective time at which all the development investment can be committed as a lump sum to account for the time value of money.

Seven output results are listed: (1) the operating benefit, (2) MPR*, the effective mass payback ratio, corrected to include repair costs, (3) the capital cost for the configuration (in T), (4) the payback time, (5) the internal rate of return, (6) the LLOX production required per T of payload corrected for repair, and (7) the corrected amount of payload put into LEO per OTV round trip.

ILLUSTRATIVE RESULTS

We illustrate the evaluation procedure by presenting results for the reference case and for the spinning tether in eccentric Earth orbit. The illustrations demonstrate how the calculations are performed and what kind of flexibility is available for sensitivity analysis and trade studies.

Reference Configuration

Turning first to a discussion of operating costs, the Transportation Model yields an MPR of 1.31. With C_s = 500 T/yr, equation (2) then leads to an annual transportation operating benefit of 118 T (of mass that need not be launched from Earth). From this operating benefit must be subtracted the three yearly repair cost components: the acquisition cost of the repair and replacement parts, the direct transportation cost of lofting their masses to their assigned destinations, and the indirect (opportunity) cost of transporting them.

Table 2 shows the effects of these corrections. The net operating benefit changes precipitously, from +118 T/yr to -601 T/yr, yielding an MPR* of only 0.454, a negative payback time (i.e., longer than ∞ at an 8% discount rate), and no internal rate of return. Less dramatically, LLOX production required per tonne of delivered payload increases from 6.13 T (C20) to 6.77 T (C14), and load delivered per OTV round trip decreases from 16.19 T (C21) to 14.86 T (C15).

From the complete spreadsheet (found in *Stern*, 1989, and not reproduced here) one learns that repair of the lunar lander alone accounts for over 80% of the total repair cost, based on the repair estimates provided by the evaluation panel for the reference case. These estimates indicate that the five major components of the vehicle must be replaced every 20-30 round trips. Since the lunar lander contributes only about 2 T to the payload for every sortie, 250 sorties per year must be carried out, requiring replacement of the entire vehicle about 10 times annually! Moreover, it can easily be shown that most of this cost (about 90%) is due to acquisition rather than transportation.

Figure 5 examines the repair assumptions. Curve (a) shows how MPR* would change if all costs per repair incident were multiplied by a uniform factor varying from 0 (no repair cost) to 2 (twice as much cost as in the standard case). The economics of the reference case are evidently very sensitive to this component of the operating cost. For comparison, curve (b) shows to what extent the sensitivity of MPR* to repair is reduced if lunar lander repair needs are first scaled down by a factor of 10, before the multiplier on the abscissa is applied.

In sum, much of the operating cost is due to lunar lander repair. It will therefore be necessary to take a closer look at the repair assumptions. This will reveal (1) whether they are realistically based on past experience and (2) whether they could be substantially reduced by additional research and development, leading to the utilization of new materials and/or better design. If neither is feasible, service requirements will severely circumscribe the vehicle's routine operation. This in turn may greatly inhibit the establishment and operation of the lunar base and the beneficial exploitation of the Moon itself.

Turning now to a discussion of capital cost, it should be pointed out that even if the mass payback ratio MPR* approached ∞, giving an annual operating benefit of 500 T for the case of C_s = 500 T/yr, there would be no net return on investment over 20 years for Configuration 0, since, from Table 2, the capital cost is over 10,000 T (location C11 in the table). That fact, combined with the mission objective (chosen to permit ready and meaningful comparison between configurations rather than to represent a realistic national or private-enterprise goal), dictates the form taken by the benefit-cost analysis. That is, long-run payoffs resulting from establishing a LEO-Moon transportation system are taken as a given in this study, and are not quantified in the evaluation procedure.

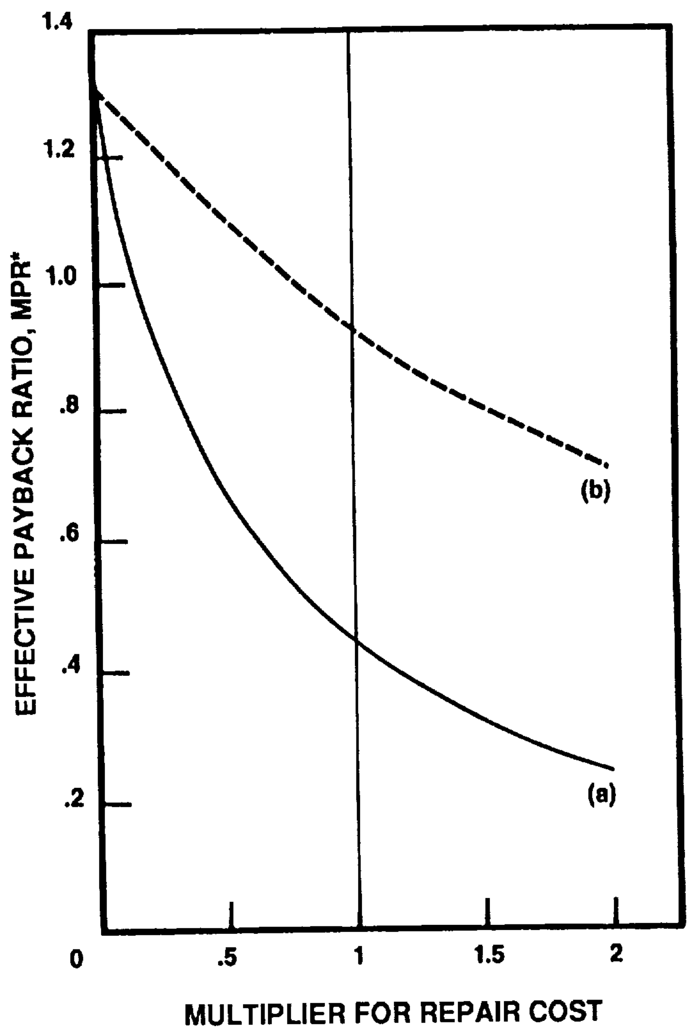

Fig. 5. Impact of repair costs on MPR* for the reference case. Curve **(a)** is obtained by multiplying all repair entries in the spreadsheet by a factor that varies from 0 to 2. Curve **(b)** results when the repair entries for the lunar lander are first reduced tenfold before the factor is applied.

Examination of capital cost components in the complete spreadsheet reveal the following: The stations in LEO and LLO are the major cost items, while the lunar oxygen plant and the two vehicles make only a minor contribution to the total. Moreover, an Evaluation Model in which all technology readiness factors R were set to 1 and all development delays to 0 gives a much-reduced capital cost of ~1870 T. This corresponds to about $5.6 billion (with $\$\$_0$ = M$3), which seems reasonable for a complete LEO-Moon "routine version" transporation system. The difference of some $26 billion should be taken as a first-cut estimate of the cost of bringing the first version of such a system to the implementation stage.

The total capital cost of Configuration 0 is quite insensitive to the assumed value of $\$\$_0$, the dollar cost of transporting 1 T from Earth to LEO, over a range of from $1 million to $10 million. It remains at about $32 billion, with only a slight rise near the high end of the range. This indicates that capital cost is dominated by acquisition cost, not transportation cost. As already mentioned, even for the relatively "state-of-the-art" reference configuration, the development cost dominates the routine acquisition cost by a factor of about 6; the sum of these two components constitutes 90% of the total capital cost, transportation only 10%.

Spinning Tether

Outputs from the Evaluation Model for the spinning tether in eccentric Earth orbit (Configuration 3) are presented in Table 3. In this case, the Transportation-Model-derived MPR (C19) is 3.10, corresponding to an uncorrected annual operating benefit of 338.7 T [see equation (2)]. Repair and replacement costs reduce the benefit by 393.4 T/yr to a net annual loss of 54.7 T, so that the effective mass payback ratio MPR* (C10) becomes 0.90. The degradation of benefit due to repair is almost halved compared to that in the reference case. As already stated in connection with Configuration 0, the low value for MPR* is not in itself very significant in our evaluation.

What is significant is the economic position of Configuration 3 relative to Configuration 0. This is indicated by the operating benefit (C9) of 546.7 T/yr and the incremental capital cost (C11) of 2054 T. Both entries are obtained by taking the difference between corresponding values for the two configurations to be compared. A life-cycle analysis performed by the Evaluation Model indicates that a payback time of 4.6 years or an internal rate of return of 26% can be realized by replacing the reference case by one including a spinning tether in eccentric Earth orbit, even though considerably more development is needed to bring the latter to maturity.

TABLE 3. Input-output section of evaluation spreadsheet for the spinning tether configuration.

A	B	C
1		
2	Spinning tether in EEO	Config 3
3		
4		Date:03-28-88
5		Time:11:49:18
6		
7	Final results	
8		
9	Operating benefit	546.696
10	Effective mass payback ratio, MPR*	0.901
11	Incremental capital cost	2054.477
12	Payback time	4.646
13	Internal rate of return	0.264
14	Corrected LLOX production per LEO T	2.891
15	Corrected LPD into LEO per OTV trip	29.640
16		
17	Parameters from Transportation Model	
18		
19	Mass payback ratio, MPR	3.100
20	LLOX production per LEO T, C_0	2.820
21	LPD into LEO per OTV Trip, C_{10}	30.193
22	Mass of loaded OTV Trip, $M_{OTV,F}$	59,222
23	LL trips per OTV trip	7.260
24		
25	Additional parameters	
26		
27	Annual payload mass to LEO, C_s	500.000
28	Cost of 1 from Earth to LEO, $\$\$_0$	3.000
29	Interest rate, r	0.080
30	Maximum time delay, Δt	17.000
31	F(r)	0.478
32		

The savings for the spinning tether come about because of the reduced fuel load. There are two reasons for this. In the first place, the ΔV that must be supplied by propellant is much reduced, resulting in greater payloads and fewer trips. Equally important, the fewer trips per year lead to smaller repair and replacement needs. For example, lunar lander maintenance requires 604 T/yr in the reference case, but only 301 T/yr in Configuration 3, leading to a mass savings of 303 T/yr, which contributes powerfully to the operating benefit.

CONCLUSIONS

Eight conclusions so far derived from the evaluation procedure described in this paper are enumerated below. The first five flow from the discussion in the paper; for the remainder, the reader is referred to *Stern* (1989).

1. The evaluation method described in this paper permits an objective comparison, based on economic criteria, of the performance of different space systems designed to accomplish a given objective. Here, the method was applied to transporting materials between LEO and the Moon, using either a relatively conventional reference transportation system, or various departures from it incorporating one advanced technology at a time. The method is equally applicable to "hybrid" systems combining several advanced technologies, to transportation systems linking the Earth and other planets or objects in space, or to objectives other than transportation. It is sufficiently flexible and modular to permit extensive "what-if" analyses; it is also helpful in pinpointing high-payoff R&D efforts.

2. Mass payback ratio, as commonly used in space-related studies, is of very limited value as an indicator of good transportation performance, unless reduction of Earth launch mass is the primary objective of the project under consideration, and capital cost is of secondary importance.

3. In our study, the limiting cost for all configurations is their enormous acquisition cost (rather than the launch or transportation cost), especially when research, development, testing, and demonstration costs are taken into account.

4. The cost to repair and replace vehicle and station components must be brought down by almost an order of magnitude if colonization and exploitation of the Moon is to become a reality. This conclusion is independent of configuration, based on those evaluated so far.

5. The spinning tether in eccentric Earth orbit and with the ability to both catch and throw loads or vehicles compares favorably with the reference configuration.

6. Several other advanced configurations, using hanging or spinning tethers, laser propulsion, and mass drivers for lunar launch, also look promising and deserve further investigation.

7. Ion-engine-powered vehicles are somewhat limited for Earth-Moon transport because of their low thrust. Because of their high I_{sp}, however, they may have an important ancillary role to play in "hybrid" configurations. None of the latter have, so far, been evaluated, nor have configurations incorporating nuclear propulsion or solar power.

8. Based on this preliminary effort, it seems likely that one will be able to identify and develop superior hybrid systems combining advanced transportation technologies. These would yield not only high mass payback ratios, but such impressive overall returns as to render obsolete conventional systems based exclusively on chemical propulsion.

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REFERENCES

Arnold J. R. and Thompson W. B. (1992) Advanced propulsion for LEO-Moon transport: II. Tether configurations in the LEO-Moon system. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century*, this volume.

Carroll J. A. (1985) *Guidebook for Analysis of Tether Applications.* NASA CR-178904. 47 pp.

Chilton F., Hibbs B., Kolm H., O'Neill G. K., and Phillips J. (1977) Electromagnetic Mars drivers. *Prog. Aeronaut. Astronaut., 57,* 37-61.

Columbo G., Gaposchkin E. M., and Grossi M. D. (1975) The "Skyhook"—A shuttle-borne tool for low-orbital-altitude research. *Meccanica, 10,* 3-20.

Henley H. W. (1992) Advanced propulsion for LEO-Moon transport: III. Transportation model. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century*, this volume.

Isaacs J. D., Vine A. C., Bradner H., and Backus G. E. (1966) Satellite elongated into a true skyhook. *Science, 151,* 682.

Kantrowitz A. R. (1972) Propulsion to orbit by ground-based lasers. *Astronaut. Aeronaut., 10,* 74-76.

Penzo P. (1987) A low Earth orbit skyhook tether transportation system. AAS 87-436, presented at the AAS/AIAA Astrodynamics Specialist Conference, Kalispell, Montana, August 10-13, 1987.

Stern M. O. (1989) *Advanced Propulsion for LEO-Moon Transport—Final Report.* NASA CR-172084.

Langmuir D., Stuhlinger E., and Sellen J. M. Jr. (1961) *Electrostatic Propulsion*, Vol. 5 of *Progress in Astronautics and Rocketry.* Academic, New York. 579 pp.

ADVANCED PROPULSION FOR LEO-MOON TRANSPORT: II. TETHER CONFIGURATIONS IN THE LEO-MOON SYSTEM

N93-17421

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INTRODUCTION

This brief work discusses a possible application of a tether as a dynamical element in a low Earth orbit (LEO)-Moon transport system, and is a part of the Cal Space study of that transport system (*Stern,* 1989). To be specific, that study concentrated on the downward transport of O_2 from the Moon to LEO, where it is stored for use as a rocket propellant, thus reducing Earth liftoff mass requirements by a factor of about 8. Moreover, in order to display clearly the role of advanced technology, only one novel technology was introduced at a single node in the transport system, the rest being "conventional" rocket transport. For general background in tether applications in space, see *Penzo and Ammann* (1989), *Carroll* (1989), and *Bekey and Penzo* (1986).

Tethers were found useful in several different roles: hanging from platforms in lunar orbits, as supports for elevators, spinning in LEO, or spinning in a tether transport orbit, an elliptical orbit with perigee at ~600 km. Here we will consider only this last use. We will first display the usefulness of the tether, then discuss the nature of the tether system, the apparatus needed to support, deploy, and control it, and end with a discussion of needed developments. Although the authors assume responsibility, this is effectively the outcome of a joint study by the tether team, the enthusiasts—J. Carroll, H. Mayer, and P. Penzo—and the critics, especially B. Waldron, G. Babb, and H. Davis, who have played a major role in clarifying our ideas.

TETHER CHARACTERISTICS

We will characterize tether materials by a simple parameter, the specific tensile strength, i.e., the ratio of tensile strength to density. This may be represented as a characteristic length, the length of itself that a cable could support against the Earth's surface gravity. For steel, for example, this characteristic length is 46 km. Alternatively, the ratio can be expressed as the square of a characteristic velocity, and for steel this characteristic velocity is 0.6 km/sec. If, however, a material with the tensile strength of steel had a density of 1 gm/cm^3 (instead of 7), the corresponding characteristic velocity would be ~1.7 km/sec. There are materials available today (trade names kevlar, spectra) that have characteristic velocities greater than 1 km/sec (Table 1) and our illustrations will be based on this number. Of course, a tether material would have to satisfy many more conditions such as flexibility, resistance to abrasion, and resistance to the hostile space environment; in this study we will ignore such questions (*Jastrzebski,* 1989).

TABLE 1. Material properties.

Fiber	Density g/cm^3	Tensile Strength σ psi ($\times 10^5$)	Char. Length, L km	Critical Velocity, V_c km/sec
Si-Glass	2.54	5.07	190	1.36
Graphite	1.9	3.6	134	1.14
Silica	2.19	8.4	270	1.62
Aramid Polymer (Kevlar)	1.44	4.0	190	1.36
Polythene (Spectra)	0.97	4.3	310	1.75

THE MISSION

We suppose that an oxygen production facility has been established on the lunar surface, and that O_2 must be transported to LEO using lunar oxygen and hydrogen from the Earth's surface as propellant. The success of the scheme is measured by the mass payback ratio, the ratio of the net mass of O_2 delivered to LEO to the mass of H_2 that must be delivered from the Earth's surface to transport the O_2.

The system includes a lunar lander (LL) that transports O_2 from the lunar surface to an orbital transfer vehicle (OTV), which receives the O_2 and delivers the H_2 for the LL (Table 2). The OTV then flies back to LEO, dissipating energy by aerobraking, retaining enough O_2 for a return trip to LEO, delivering excess O_2 and picking up H_2 for the LL and the next return trip.

We will add to this reference system a tether platform (TP) in an elliptic orbit, which uses a spinning tether to add ΔV to the OTV, by recovering some of the orbital momentum of descending OTVs, on the "space elevator" principle (*Isaacs,* 1966).

TABLE 2. Vehicles for tether-assisted transport.

Vehicle	Mission	Gross (tonnes)	Dry	Fuel O:H	I_{sp} sec
Orbital Transfer Vehicle (OTV)	LEO<->LLO	60	14	6:1	470
Lunar Lander (LL)	LLO<->Moon	35	5	11:1	430
Tether Platform (TP)	TTO	400	---	6:1	470

THE LUNAR LANDER

The lunar lander has a dry weight of 5 tonnes and delivers a payload of 11 tonnes to the OTV. The LL uses an oxygen-rich fuel (1:2) with an I_{sp} of 430 sec. With a required ΔV of 2 km/sec, this uses a total of 16.2 tonnes of propellant/roundtrip, made up of 4 tonnes for the downward trip, carrying a payload of 1.5 tonnes of H_2, and 12.2 tonnes for the upward trip, which carries a payload of 15 tonnes of O_2 and uses 5.6 tonnes of H_2. We will use these numbers for the rest of the mission (see Table 3).

TABLE 3. The lunar lander.

Mission	ΔV km/sec	Payload (tonnes)	Total	Propellant O	H	Net O
LLO-Moon	2	1.5	4	3.66	0.33	
Moon-LLO	2	15	12.2	11.15	1.05	11
4 trips				Total H_2	5.5	44

REFERENCE CASE: OTV ALONE

In this case, the OTV leaves LLO, falls back to Earth, loses energy by aerobraking, and is placed in LEO needing a ΔV of ~1 km/sec. The vehicle has a dry weight of 14 tonnes, and uses a hydrogen-rich fuel (O:H = 6:1) with I_{sp} = 470 sec. With a payload of 34 tonnes, this requires 12 tonnes of propellant, 10.3 tonnes of O_2 and 1.7 tonnes of H_2; thus, 33.7 tonnes of O_2 reach LLO.

On the return (ascending) flight the payload is 6 tonnes of H_2, and a ΔV of 4 km/sec is required. This uses 28 tonnes of propellant, 24 tonnes of O_2 and 4 tonnes of H_2; thus, the net O_2 delivered to LLO is 9.7 tonnes and the total H_2 consumed is 11.3 tonnes. The payback ratio is then only 0.85 and there is a net loss (see Table 4).

TABLE 4. Reference case: OTV alone.

Mission	ΔV km/sec	Payload (Tonnes)	Total	Propellant O	H	Net O
LLO-LEO	1	34	12	10.3	1.7	33.7
LEO-LLO	4	6	28	24	4	9.7
Total H_2 5.7 + 5.6 = 11.3 tonnes						
Payback Ratio = 0.85						

THE SPINNING TETHER

Suppose now that the OTV is picked up by a tether of length 100 km, which is rotating prograde with a tip speed of 1 km/sec about a platform in an elliptic orbit with perigee at 600 km (100 km above the LEO circular orbit), and a perigee velocity of 8.3 km/sec (1 km/sec above the LEO velocity), so that the tether tip and the OTV have momentarily the same speed. The OTV is now swung around the platform, and at its maximum has speed of 9.3 km/sec (1 km/sec faster than the platform). Now it needs only 2 km/sec to reach LLO, and for a total load of 20 tonnes (14-tonne vehicle and 6-tonne payload) needs only 10.8 tonnes of fuel, including 9.3 tonnes of O_2, instead of the 24 needed in the reference case. Thus, the O_2 returned as payload to LEO is 24 tonnes, the H_2 used has been reduced to ~8.6 tonnes, and the payback ratio is increased from 0.85 to 2.8.

This must be reduced, however, since in the process of accelerating the loaded OTV, the tether platform has lost momentum 30.8 × 2 = 61.6 ton km/sec. This could be made up with a low-thrust, high specific impulse drive, such as an electric propulsion system, but if made up by the fuel that powers the OTV, needs 13.4 tonnes of fuel, including 11.4 tonnes of O_2 and 2 tonnes of H_2, and the payback ratio is reduced to 1.2.

If the descending OTV of mass 46.7 tonnes is not brought into LEO by aerobraking, but is instead put into an elliptic orbit with perigee at 700 km/sec and a velocity of ~9 km/sec, it can be caught by the tether, brought into LEO and released, giving a change in momentum of the opposite sign to that produced in upward-throw, with a magnitude of 93.4 ton km/sec. To remove the net Δp of 31.8 tonnes km/sec needs only 7 tonnes of fuel, including 6 tonnes of O_2 and 1 tonne of H_2. Then 18.3 tonnes of O_2 are available as payload, and the H_2 needed is 9 tonnes with a payback ratio of 2.1.

There is yet another possibility. Note that the Δp on throwing is about half of that on catching, and opposite in sign, thus if every second descending load is caught by the tether, the orbit of the TP will return almost to its initial value after every second throw, the net Δp being reduced to 1.8 tonnes/km/sec. Moreover, the ΔV experienced by the platform on a simple throw is $\Delta V(TP) = \Delta V(OTV) \times (M(OTV)/M(TP)) \propto 0.1$ km/sec for the masses we have chosen. If the throw and catch occur at perigee, this height is unaltered, and if the tether tip speed can be increased by 0.1 km/sec between throws, the OTV orbits are unchanged. An even better match is effected if two catches are made for every three throws. As displayed in Table 5, this increases the payback ratio to 2.7.

TABLE 5. Tether-assisted transport (optimal case).

Tether Assisted
(1) Tip Rendezvous: Throw + Optimal Catch + Aerobraking (3 Throw, 2 Catch)

Mission	Vehicle	ΔV Tonne km/sec	ΔV	Payload	Total	O	H	Net O_2
LEO-LLO	OTV		2	6	11	9.5	1.5	
LLO-LEO	OTV		1	34	12	10.3	1.8	
ΔV	TP/throw	1	—	—	0.25	0.2	0.05	
Net O_2								23.4
Total H						8.7		
Payback Ratio				2.7				

RENDEZVOUS AT PLATFORM

Babb (personal communication, 1988) observed that the tether tip rendezvous could be particularly difficult and suggested that instead rendezvous should be at the center of mass. This leaves the TTO-LLO section of the orbit unchanged, but requires that the ascending OTV make the transfer from LEO to the elliptical platform orbit, with a ΔV of 1 km/sec. This requires an added fuel consumption of 6.9 tonnes, including 6 tonnes of O_2. If the momentum loss of the tether platform is made up by a high specific impulse drive, the payback ratio is ~2.2. If, on the other hand, momentum loss is made by conventional rocketry, and the downcoming fuel is brought into LEO by aerobraking, the fuel required to make up 29 tonne/km/sec is 6 tonnes including 5.1 tonnes of O_2. The same amount is required to spin the load up to speed after capture; hence, allowing for these requirements,

the net O_2 available as payload is 10 tonnes and the payback ratio is reduced to 1. It may be possible to reduce the fuel needed to spin up the load by using onboard manipulation of the tether tension, but the energy requirements for this might be prohibitive.

THE ROLE OF "UNOBTAINIUM"

In our analysis so far we have made the conservative assumption that tether materials are restricted to the specific tensile strengths available today. However, a LEO lunar transport system is probably at least a decade in the future, and materials science is developing at a rapid pace. As a speculation, let us consider the use of a tether for which v_c is 2 km/sec. The tether could then provide a ΔV of 4 km/sec and the OTV would then need only maneuvering capability of, perhaps, 0.8 km/sec and 3.7 tonnes of fuel, including 3.2 tonnes of O_2. If, moreover, the momentum loss of the tether platform is made up by a high specific impulse drive, the net O_2 payload becomes 33.7 - 3.2 = 30.5 tonnes of O_2. The H_2 requirement is 8.5 tonnes and the payback ratio is 3.47. If the momentum transfer must be made up by rockets, this calls for 20 tonnes of fuel including 17.2 tonnes of O_2, and the payback ratio is reduced to 1.2; however, if the loaded OTV is caught on every fourth return, the net Δp of the TP is 12 tonnes km/sec, or 3 tonnes km/sec per throw. To replace this it needs only 0.65 tonnes of propellant including 0.56 tonnes of O. The net O_2 obtained is 30 tonnes, the total H_2 used, 8.4 tonnes, and the payback ratio is 3.4.

THE TETHER SYSTEM

The tether itself may be a rather simple structure, but the system as a whole, consisting of a ballast platform, the orbital makeup drive, the winches for controlling the tether, the onboard power system, the guidance and control needed to locate the center of mass and keep it in orbit, and the system needed to effect rendezvous, is complex, and in this brief description we will only sketch out the requirements.

The Tether

Tether mass (1.4 M_L). This is determined by the tip speed, which depends on v_c for the tether material. The mass is minimized if the area at any distance from the center of mass is selected so that the tension in that section is a fixed fraction of the breaking stress. It is readily shown that the cross section should have the form

$$a(x) = a_o \exp\left(-\frac{1}{2}\left(\frac{v_t x}{v_c l}\right)^2\right)$$

where l is the tether length and v_t the tip speed. The ratio between the load mass and the tether mass for $v_t = v_c$ is then 1.41. For our conditions in the throw only mode, this calls for a tether mass of 41 tonnes. In the throw and catch mode, the larger mass of the downcoming load (48 tonnes) increases this to 67 tonnes.

Tether length (100 km). This is determined by a compromise between the acceleration to which the load is subject (which decreases with tether length), the time for rendezvous (which increases with tether length), and the probability of damage or destruction by collision with space debris. At 100 km, the expected mean life is ~10 years, and the probability of destruction during the first year is ~10%.

The Tether Platform

Platform mass (~10M_L). On capturing and releasing a load, the platform experiences a change in velocity

$$\Delta V = \frac{\Delta p}{M_T}$$

where M_T is the final mass of the total system: tether, platform, and load on capture, tether and platform on release. The resulting change in the TTO leaves the perigee fixed (if throw and catch both occur at that point), but changes both the semimajor and semiminor axis. If the ballast mass is 10 times the load mass, the change in velocity on catch and throw is 0.2 km/sec, 2.5% of the orbital speed, which is acceptable for catch and throw at perigee, but would otherwise be marginal. For our case this ballast mass is 310 tonnes on throw and 480 tonnes on catch. Note, however, that most of this is ballast, and could be made of spent space units, such as external tanks.

Drive. To make up for momentum loss, a drive of some kind is needed, and since the changes in velocity required are ~0.2 km/sec, the propellant mass needed is

$$\frac{\Delta p}{g I_{sp}}$$

which we have included in our transport models, if conventional rocketry is used. We have observed that a very high I_{sp} drive, electrostatic with I_{sp} ~10^4, would greatly reduce the propellant mass (to 2% of the load mass) to ~0.6 tonnes for the upgoing load—although increasing the onboard power required.

Winch. The winch is needed to deploy and spin up the tether, and to alter tether length when catching a load. When fully deployed and spinning at angular velocity Ω or velocity V, the tether for a 1-T load has an energy E of $0.2 \times 1/2 M_T V^2 = 1.32 \times 10^8$ J/T and an angular momentum of $2E/\Omega = 2Er/V = 2.6 \times 10^{10}$ J sec/T. Note that revving up is a one-time operation, which might be provided by an OTV that draws out the tether and gives it the required angular momentum, or it may be carried out as part of the operation of placing the platform in orbit, if it is feasible to deploy the tether early. Since the orbital makeup thruster is not at the center of mass, it will alternately add to and take away angular momentum from a rotating tether. If the thrust is modulated with respect to the tether phase, the thruster can be used to modify the rotational angular momentum.

In low Earth orbit, angular momentum can be provided by the gravitational gradient. The available energy then is ~0.2 $M_T V_{esc}^2$ $(1/r)^2$, which for the lowest orbit (~200 km) is ~5% of the final required energy. To add angular momentum, the tether must be extended on the downswing and reeled in on the upswing. Initially, there are ~23 minutes to do this, and the initial power needed to reel the tether in is 4.8 kW/T.

If the winch can reel at a speed of 2.5 km/min at low load (800 rpm for a 1-m-diameter reel), then the energy increment is of order 2.5% of the final requirement/cycle, the power needed reaches 160 kW/T near the end of the build-up, and the full angular momentum can be supplied in about 10 days from the gravitational gradient. About 100 kW/T of winch power is needed during capturing maneuvers in which it will be important to

change the tether length by amounts of order 1 km in times of order 1 min. The winch will also be involved in platform orbital correction operations. Since the drive site at the platform is 21.6 km from the center of mass, angular momentum is alternately added and taken away from the system while under drive. Modulating the tether length gives control of the net angular momentum addition.

Finally, the winch is needed to damp out unwanted vibrations of the tether, and to compensate for hysteretic losses in the gravitational gradient. This last operation will require continuous power of order 1.5 kW/T. The winch would have to be rather impressive since it must reel in at a few kilometers/minute under a tension of ~2.4T/T load.

Power Requirements

In addition to the drive, onboard power is needed to drive the winch, at a peak of 160 kW/T load. For our case, with only the upward load using the tether, this requires ~4.8 MW. If the downward load is also handled this becomes 9 MW. This power is needed only in pulses of a few minutes, and is most important during the final stage of spin up or during load capture.

Energy Storage

It would be useful to have some means of storing the energy and angular momentum of the tether, when modest changes in length are made. Since angular momentum is conserved, energy will be stored in the rotation of the platform about the center of mass, and in the speedup of the tether tip. If substantial reel-in is required, for example to avoid a collision, a modulated burn of the platform propellant could get rid of or replace angular momentum to keep the spin at a tolerable value.

Rendezvous Requirements: Tether Tip Vehicle (TTV)

There may be a need for a vehicle at the tether tip to make rendezvous with the incoming loads. We assume that the incoming vehicle can get within 1 km of the rendezvous. The TTV then has about 1 min to detect the incoming load at 100 km and effect a rendezvous. This can be done with an acceleration of 1 m/sec^2 and a maximum speed ~200 m/sec. The vehicle can carry a light line of length ~2 km and of mass ~1 kg, which can be used to draw the tether tip to the incoming load.

The TTV needs detectors and control for target acquisition and guidance, a drive giving ΔV ~200 m/sec and an acceleration of 10 m/sec^2 to make rendezvous with the incoming load, a releasable clamp to attach to the target, a reel that can pull the tether tip ~1 km in about 100 sec, and some scheme for clamping it to the load. In addition to a modest drive ($\Delta P = M_v \times 100$ m/sec) requiring ~3% of the vehicle mass (M_v) in propellant, the TTV needs an onboard power source and a motor yielding ~0.5 kW, enough to reel in the tether mass ~1 kg) in about 1 min. A reel of radius 16 cm, spinning at 1000 rpm under a load of 2 kg, would be adequate. A total TTV mass of 100 kg is probably generous. It might be preferable to mount the reel and the windup motor on the tether tip, thus reducing the TTV mass, but increasing the mass that must be reeled in. Note that the figures given here are per ton of load captured. For our case, the onboard power, the mass of the retrieval line, and the TTV mass must be increased by factors of 30 or 56 in the two cases.

An alternative might be to have the incoming vehicle throw out a line at right angles to the tether so that the lines cross and grapple. Then the TTV vehicle becomes superfluous, although a tether and reel will still be needed. The use of a tether in this mode calls for a "smart" orbiting vehicle, capable of making a rendezvous within less than a minute, within 1 m.

Repair and Maintenance

A serious problem faced by the tether is damage by collisions with debris in space. *Carroll* (1989) has analyzed this using data from *Keller* (1984) and finds that a major role is played by objects a few centimeters in length. If the tether were a single strand, or a single woven cable, then it would have approximately a 10% probability of having a collision and being destroyed each year. One way of compensating for this is to increase the tether mass and build in some redundancy. Moreover, instead of using a single strand, the tether could be composed of a number of strands separated by a few tens of centimeters. Then, although the probability of any given strand being broken is unaltered (and the probability of some collision is increased) a single collision is not fatal. This, by itself, gives no advantage, but if we add some method of repairing damage, either by reeling in the tether for repair, or by sending out some kind of machine that will repair the damaged section without reeling in, then the tether lifetime could be increased by a very large factor. Of course, if the amount of space debris increases in the future due to human activity, the hazards and the need for repair also increase.

CONCLUDING REMARKS

This study strongly suggests that even under very restrictive conditions tethers could play a major role in an Earth-Moon transport system. If a wider view is taken, and extra new technologies or multiple tethers are permitted, their role becomes even more significant. In a mature transport system, in which mass transfer up to the Moon and transfer down to LEO are more nearly equal, the momentum make-up problem could be greatly reduced, and the advantages of tethers become greater.

It must also be clear that the use of tethers depends on the solution of many novel and formidable problems. Can tether materials be developed that will provide not only the required specific tensile strength, but also the flexibility and resistance to abrasion, solar and particle radiation, and to heating in the upper atmosphere? Can reliable technologies for deploying and manipulating tethers, especially multistranded systems, be developed in view of the rates of reeling and the degree of control required? Can the winches, controls, and power sources be produced capable of these complex manipulations that must be carried out without manual intervention? Can the rendezvous problems at the tether tip be mastered? On a more basic level, what are the best orbits for tether missions? We have considered one, but are others more desirable? (Almost certainly, yes!) What are the orbital limitations to tether applications? (In equatorial orbits, a few days/month; in 20° orbits, a few days in every 80 for the orbits we have considered.) What are appropriate manipulation strategies? How should the equipment be designed?

The challenges here to invention, to control engineering, to robotics, and to understanding fundamental mechanical problems, are great enough to call the use of tethers in space an entirely new branch of engineering, and possibly one of greatest importance. Design and laboratory studies, supplemented by space testing, need dedicated resources and an early start; but tethers may eventually justify their most enthusiastic supporters.

REFERENCES

Bekey I. and Penzo P. (1986) Tether propulsion. *Aerosp. Am., 24,* 40–43.

Carroll J. (1989) Guidebook for analysis of tether applications. In *Tethers in Space Handbook,* 2nd edition (P. Penzo and P. Ammann, eds.). NASA Publication NSW-4341.

Isaacs J. D., Vine A. C., Bradner H., and Backus G. E. (1966) Satellite elongated into a true skyhook. *Science, 151,* 682.

Jastrzebski Z. D. (1989) *The Nature and Properties of Engineering Material.* Wiley, New York. 553 pp.

Keller D. J. (1984) *Orbital Debris Environment for Space Station.* JSC Publ. No. 20001, Houston.

Penzo P. and Ammann P., eds. (1989) *Tethers in Space Handbook,* 2nd edition. NASA Publication NSW-4341.

Stern M., ed. (1989) *Advanced Propulsion for LEO-Moon Transportation, Final Progress Report.* NAG9-186, California Space Institute, Univ. of California, San Diego.

ADVANCED PROPULSION FOR LEO-MOON TRANSPORT: III. TRANSPORTATION MODEL

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A simplified computational model of low Earth orbit-Moon transportation systems has been developed to provide insight into the benefits of new transportation technologies. A reference transportation infrastructure, based upon near-term technology developments, is used as a departure point for assessing other, more advanced alternatives. Comparison of the benefits of technology application, measured in terms of a mass payback ratio, suggests that several of the advanced technology alternatives could substantially improve the efficiency of low Earth orbit-Moon transportation.

INTRODUCTION

A computer model has been constructed to assess new technology alternatives as implemented in a reference Earth-Moon transportation infrastructure. This transportation model was developed as part of the Advanced Propulsion for Low Earth Orbit-Moon Transportation study performed for NASA Johnson Space Center by the California Space Institute at the University of California, San Diego (*Stern,* 1989). Input for the transportation model has been developed through interaction with participants in this study to determine the mass payback ratio of transportation system alternatives. This mass payback ratio is only a first measure of merit, and has been used in the study as an input to a separate economic model (*Stern,* 1988) that assesses overall efficiency and cost-effectiveness of these new technology alternatives.

The reference transportation infrastructure employs orbit transfer vehicles (OTVs) for orbit-to-orbit transfer, OTV-derived lunar landers for transportation between the lunar surface and low lunar orbit (LLO), and orbital transfer and staging facilities (OTSFs) in low Earth orbit (LEO) and LLO. Technology needed for the reference infrastructure is already in the planning and early development stages (*Bialla and Ketchum,* 1987).

Several advanced technology alternatives are considered in the transportation model. Tether-assisted transportation, wherein a long tether exchanges momentum between an orbital facility and an OTV or lunar lander, is examined for use from facilities in LEO, eccentric Earth orbit, and LLO. Other advanced technology alternatives considered include lunar-derived aerobrakes, laser propulsion, and ion engines as modifications of the reference OTV, and use of a mass driver to eject material from the Moon's surface into lunar orbit. System parameters for configurations using these technologies were determined through the interaction of a team of academic, government, and industry representatives participating in the Advanced Propulsion for LEO-Moon Transportation study, resulting in representative alternative configurations analyzed in the transportation model.

These alternative systems, which use more advanced technology, are compared with the reference transportation infrastructure in terms of mass payback ratio (MPR), the net mass of lunar material delivered to LEO per unit mass of terrestrial material used in the system (*Frisbee and Jones,* 1983). An MPR greater than one is considered to be necessary for the export of lunar material (such as lunar oxygen) down to LEO, which is preferred over the transport of similar material up from Earth. The reference transportation system can achieve an MPR slightly greater than one (the system can deliver more lunar mass to LEO than the terrestrial mass needed to produce and transport this lunar mass). Mass payback ratios for some of the more advanced system alternatives considered in the following pages are high enough to suggest that these technologies should play a major role in future lunar operations.

REFERENCE TRANSPORTATION INFRASTRUCTURE

The reference infrastructure is based upon recommendations of recent studies at General Dynamics Space Systems Division (*Bialla,* 1986; *Bialla and Henley,* 1987), with minor modifications to optimize the system for utilization of lunar oxygen. Figure 1 provides an overview of this reference infrastructure, illustrating

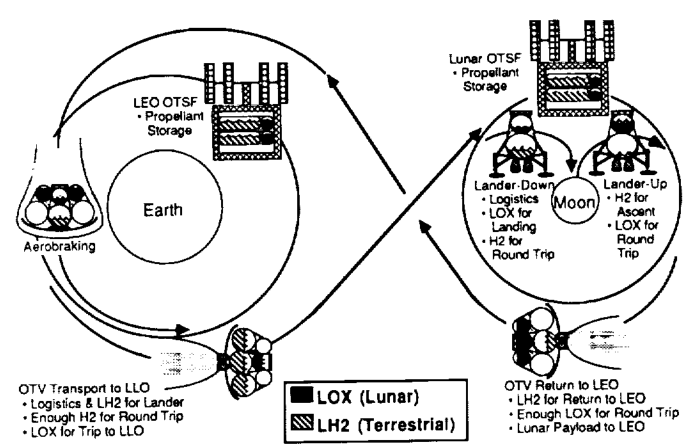

Fig. 1. Reference orbital transfer infrastructure.

the orbit transfer vehicle (OTV), orbital transportation and staging facilities (OTSFs) in LEO and LLO, and an OTV-derived lunar lander.

OTV Concept

The OTV concept chosen for this reference infrastructure is modeled after the modular S-4C concept recommended in recent OTV studies (*Ketchum et al.,* 1988). This space-based, reusable, aerobraked vehicle is illustrated in Fig. 2. The only significant modification of the S-4C for this lunar application is an increase in the aerobrake mass in order to accommodate the large masses of lunar material brought to LEO each time the OTV returns.

The OTV is propelled by two advanced oxygen/hydrogen (O_2/H_2) engines of 22,000 N (5000 lbf) thrust each, with an oxidizer-to-fuel (O_2:H_2) ratio of 6:1 and a specific impulse of 485. This relatively low thrust level minimizes engine mass, but requires a multiple perigee burn trajectory to reduce gravity losses upon departure from LEO. A modification of this OTV engine for lunar lander applications would make use of a significantly higher mixture ratio (well past the stoichiometric ratio of 7.8:1).

The S-4C OTV concept allows variation of the number of tanksets (sets of individual tanks for 0_2, H_2, pressurant, and RCS propellants), with combinations of 1, 3, 4, 5, and 7 tanksets giving the vehicle a wide range of propellant capacity. For the reference OTV, different tankset options have been considered in the analytical model, and the three-tankset configuration has been chosen for the reference OTV. The less efficient one-tankset configuration might be reasonable for use in early, low-mass transport operations required to set up an initial infrastructure, and the most efficient seven-tankset configuration might be preferred for eventual, high-mass transport operations.

The reference OTV uses a fully reusable aerobrake that is sized as a function of the mass brought back to LEO. The aerobrake is specified to be 13% of the total mass entering the Earth's atmosphere, a factor that is typical of previous OTV designs for return from geosynchronous Earth orbit (GEO).

Modular avionics on the OTV allow modification of guidance and control systems with advances in the state of the art. The modular avionics approach also allows easy modification of guidance as required for an OTV-derived lunar lander.

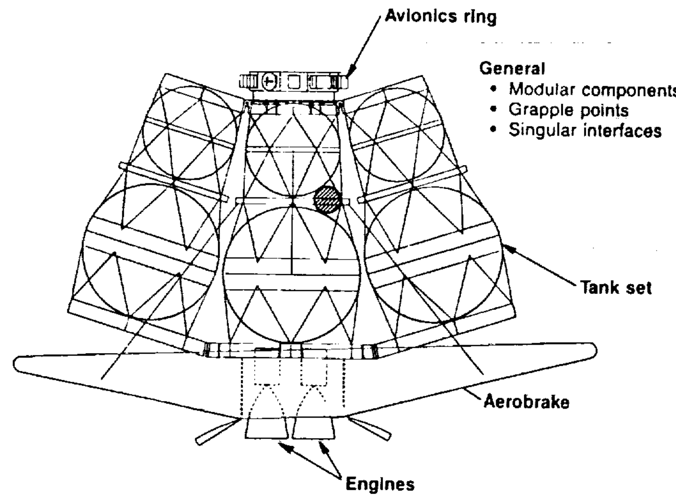

Fig. 2. Reference orbital transfer vehicle.

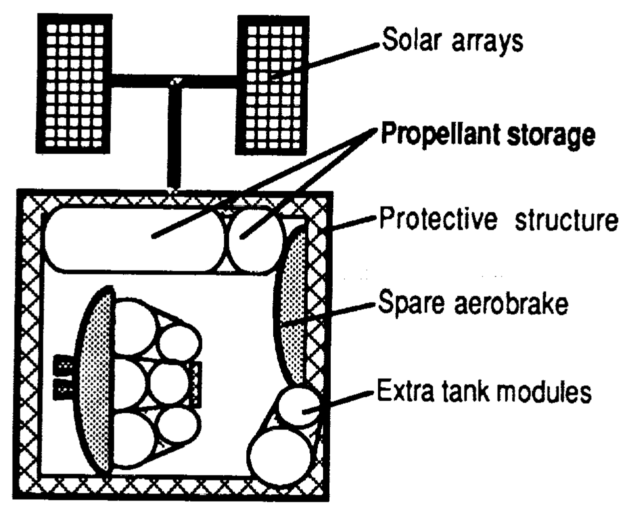

Fig. 3. Representative orbital transfer and staging facility.

Orbital Transportation and Staging Facilities

Two orbital transportation and staging facilities (OTSFs) are used in the reference infrastructure, one in LEO and one in LLO. The OTSF functions include spare vehicle parts storage, meteoroid and debris shelter, and propellant storage. In the transportation model, these facilities are repositories for lunar oxygen and terrestrial hydrogen. With an OTSF present in LLO, the lunar lander can deliver lunar oxygen to LLO while the OTV is in transit between LLO and LEO.

A representative LEO OTSF is illustrated in Fig. 3. Its subsystems are derived from space station hardware and, in this reference case, it co-orbits with the space station at 28.5° inclination and 400-km altitude. Telerobotic operations are expected to be the normal means of maintenance, propellant transfer, and payload processing.

The representative LLO OTSF is similar to the LEO facility in most respects. The lunar facility may use a more advanced solar power system (if derived from evolving space station hardware), and has a larger OTV hangar for multiple vehicles. This facility contains several manned modules, and is expected to evolve with time and eventually serve as a staging base for Mars missions using lunar LOX (*Bialla,* 1986; *Cordell and Wagner,* 1986). More detailed definition of LLO OTSF systems is needed, including design adaptable to later modification by more advanced technology.

OTV-derived Lunar Lander

The reference lunar lander is illustrated in Fig. 4. This configuration is derived from the OTV by substituting landing gear for the aerobrake, and thus has common subsystems and interfaces for propellant handling. More sophisticated avionics packages are substituted for the additional requirements of launch and landing. A single-tankset derivative of the OTV is used for the reference lunar lander, as the thrust from its two engines would be insufficient to lift a larger lander (with full O_2 tanks) from the Moon's surface. The most significant feature of the lander selected

for the reference configuration is the modification of the basic OTV engine for operation at a higher mixture ratio. The purpose of this vehicle is the transport of O_2 from the Moon's surface to LLO, and the return to the surface with logistic supplies and enough H_2 for the next trip up to LLO.

Engine performance as a function of O_2:H_2 ratio (the ratio of O_2 used to H_2 used) follows the trend of the curve in Fig. 5. This curve is based upon the output of a General Dynamics computer program, for one-dimensional equilibrium O_2/H_2 combustion in an engine with a 100-bar (1500 psi) chamber pressure and an area ratio of 400. Higher chamber pressures and area ratios would generally increase the engine's I_{sp}. (Optimal area ratios may actually be lower due to factors such as increased weight and radiative energy losses associated with large engine nozzles.) As the mixture ratio increases beyond the region typical of current O_2/H_2 engines (around 6:1), the I_{sp} (force divided by mass flow rate) decreases. Lunar lander applications can achieve higher MPRs at higher mixture ratios in spite of this decrease in I_{sp}, as the O_2 used is nearly free, while H_2 must be imported from Earth. Oxygen/hydrogen ratios selected for the OTV and the lander were arrived at by trial of various mixture ratio (and corresponding I_{sp}) parameters in the transportation model. The selected O_2:H_2 ratio of 12 for the lunar lander was a compromise; slightly better MPRs would result if the lander engine were operated at a higher O_2:H_2 ratio (>12) for liftoff and at a lower ratio (<12) for landing, but this would require variation in the mixture ratio during flight rather than the somewhat simpler alternative of a constant high mixture ratio. Engine temperatures predicted for this high mixture ratio are actually cooler than those created in conventional 6:1 mixture ratio engines.

Technology Development Requirements

The reference transportation infrastructure in this model presumes fruition of certain technology developments for reusable OTVs, OTV-derived lunar landers, space-based OTV accommodations, and the lunar surface base. Key OTV technology in the reference case includes aerobraking, advanced O_2/H_2 engines,

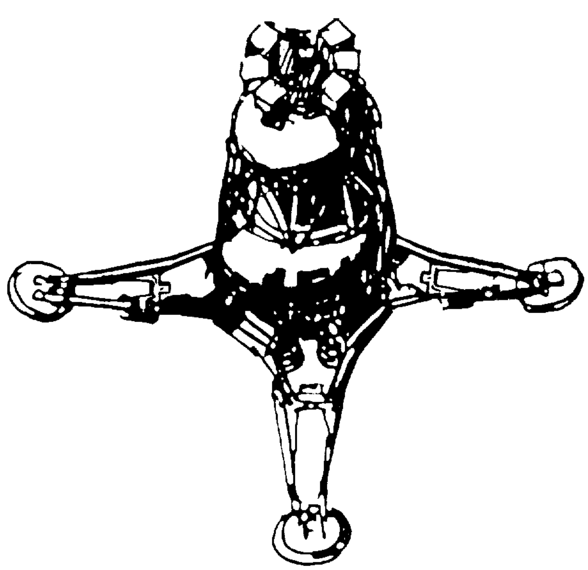

Fig. 4. Reference lunar lander derived from OTV subsystems.

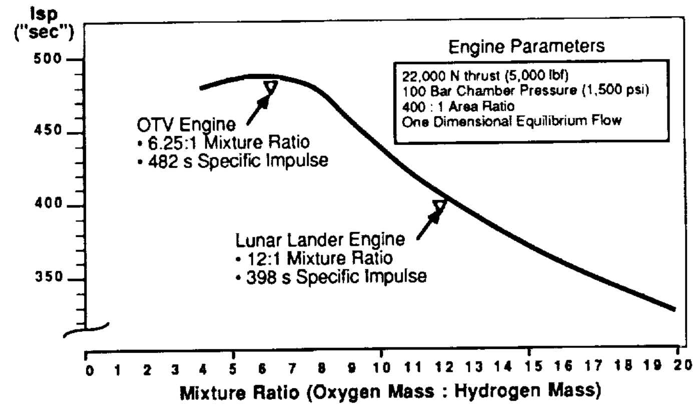

Fig. 5. Engine performance vs. O_2:H_2 mixture ratio.

advanced avionics, and lightweight structures. Technology for space-based OTV servicing at an OTSF includes telerobotic maintenance, zero-g propellant transfer, and automated rendezvous and docking. New technology is also needed for lunar materials processing to produce liquid oxygen propellant for the OTV and lunar lander. In order to use this lunar oxygen most effectively, the lunar lander uses an engine with a high O_2:H_2 ratio.

Modification of a basic OTV engine to operate at a higher mixture ratio for lunar lander applications is considered to be a reasonable evolutionary step for an engine that is still in the early stages of technology development. Engine technology development activities sponsored by Lewis Research Center (such as the use of gaseous oxygen to drive oxygen turbopumps), are relevant to such an increase in O_2:H_2 ratio. Similar high O_2:H_2 ratio and variable O_2:H_2 ratio engines are being studied for Earth-to-orbit applications, where the increase in O_2:H_2 ratio can reduce launch vehicle dry mass (*Martin,* 1987). Small O_2/H_2 engines at the stoichiometric (7.8:1) ratio have been developed and tested for use on satellites (*Stechman and Campbell,* 1973) and on the space station (*Robinson and Rosenthal,* 1986; *Senneff and Richter,* 1986; *Norman et al.,* 1988).

ANALYTICAL MODELING OF TRANSPORTATION INFRASTRUCTURES

An analytical model has been developed to compare advanced technology alternatives against this reference architecture. This model uses Excel spreadsheet software to apply an iterative series of equations to alternative transportation systems. This relatively simple model can easily be modified to consider variations of input parameters, and can be run rapidly on a personal computer.

The analytical model of the lunar transportation infrastructure, which considers separate loops for LEO-LLO and LLO-lunar surface transportation, was illustrated in Fig. 1. The lunar lander: (1) leaves the surface with a full load of O_2 (35,000 lbm) and enough H_2 to reach LLO; (2) transfers excess O_2 to the lunar OTSF (retaining enough to return to the surface) and receives H_2 and logistics mass to make the next round trip and produce the next load of O_2; and (3) returns to the surface to complete this loop. For the reference case, the lander must make approximately seven round trips to the lunar OTSF to transport the O_2 that will be transferred later from the OTSF to fill the three tanksets of the OTV. The OTV loop: (1) leaves LEO with enough

H_2 to make the round trip, enough O_2 to reach LLO, and the payload (hydrogen and logistics mass) required to support the approximately seven lander loops; (2) delivers the payload to LLO and refills oxygen tanks at the lunar OTSF; and (3) returns to LEO with excess O_2. The ratio of this excess O_2 (beyond that required for the next trip up) to H_2 and logistic mass is termed the MPR. This ratio (1.32:1 for the reference infrastructure) is a basis for assessing new technology alternatives to the reference system.

Material on the surface of the Moon is at a higher potential energy level than the same mass in LEO, as illustrated in Fig. 6. If we could construct a "siphon" between the Moon's surface and LEO, mass would flow freely, and if we placed a "turbine" in this mass flow, a tremendous amount of energy would be released. In the reference system, we construct such a "siphon," although it is not very efficient in mass transfer (requiring an input of mass from Earth) or in energy conversion (dissipating energy by aerobraking). Alternative systems that supplement the reference configuration by more advanced technology are generally more efficient in mass transfer and/or energy conversion.

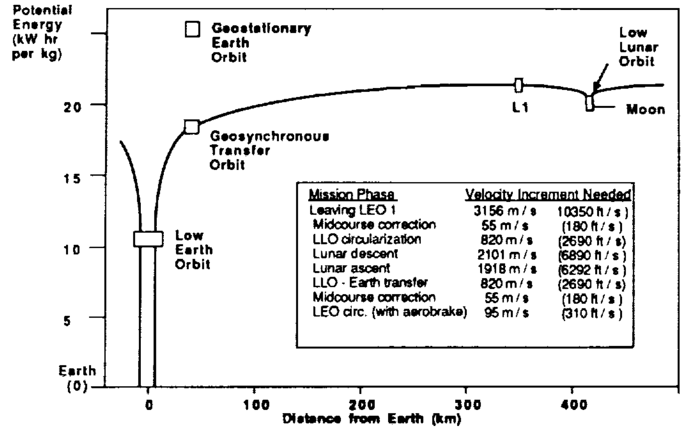

Fig. 6. Potential energy of lunar material.

Velocity increments used in the transportation model are also shown in Fig. 6. For an unmanned OTV, much longer flight times might be reasonable, with attendant reduction in its mission ΔV requirements. The altitude and eccentricity of "low" lunar orbit have not been optimized (with corresponding changes in the individual velocity increments) for the reference or alternative infrastructure, but such an analysis would probably result in greater MPRs. Gravity losses for the lander (which transports more mass upward than downward) could be higher in ascent than in descent, tending to exchange the ΔVs attributed to these mission phases.

Hydrogen is the major component of the OTV's payload from LEO to LLO. For cases in which H_2 use exceeds OTV capacity, additional tankage, weighing 10% of the contained propellant, is presumed to be carried to LLO (and left there). The OTV's H_2 tankage is actually oversized for most mission propellant requirements, and thus, if the logistic mass is H_2, it might be carried directly within OTV tanks. For example, production of O_2 by reduction of ilmenite and subsequent water electrolysis (*Gibson and Knudson,* 1985) would use H_2 as a principal reagent

$$H_2 + FeTiO_3 = H_2O + Fe + TiO_2$$

$$2H_2O = 2H_2 + O_2$$

If all the H_2 used in this reaction is not recovered, H_2 might comprise a substantial portion of the logistics mass required for lunar O_2 production. The transportation model assumes that one unit of terrestrial mass must be delivered to the Moon's surface for every 100 units of lunar mass produced on the Moon (O_2 or other useful lunar products). Spare parts for OTV, OTSF, and O_2 production facility maintenance are not separated from other logistics in this transportation model; however, both their unit cost and transportation cost are included in an economic model (*Stern,* 1988), which uses the output of this transportation model.

This LEO-Moon transportation model describes steady-state operations, assuming that the lunar base, including an O_2 production plant, is already established for reasons other than transport of lunar material to LEO (e.g., scientific exploration). The reference infrastructure would initially transport men and supplies for a manned lunar base, and thus "bootstrapping" of the system (to provide for its own development) is not considered. Expansion of the system for higher O_2 production and transportation rates would require a temporary increase in the flow of mass from Earth, with a return to steady-state operation after system expansion is complete.

TRANSPORTATION MODEL RESULTS

The transportation model has been used both in refining the reference transportation infrastructure and in assessing modifications of this infrastructure with more advanced technology. Results of calculations using the transportation model are portrayed in the following charts, with MPR indicated on the vertical axis. While the scale changes somewhat to accommodate the range of results, the reference transportation system's MPR of 1.31 is indicated on all the charts by a dashed line, and a solid line indicates an MPR of one (the limit for practicality of transport of material down to LEO from the Moon, rather than up from Earth).

Reference Infrastructure Refinement

The significance of both the number of OTV tanksets and the high mixture ratio for the lunar lander is illustrated in Fig. 7. As the number of OTV tanksets increases, the system yields greater MPRs. A large improvement is realized by increasing from one to three tanksets, with far less benefit thereafter. The three-tankset OTV configuration is considered to be most desirable, as it achieves relatively high MPRs, yet keeps the total oxygen load (which the lunar OTSF must store prior to transfer into the OTV) at a reasonable level. When the three-tankset OTV is combined with a 6:1 mixture ratio lunar lander, it obtains an MPR slightly greater than one (1.07); however, the use of the 12:1 lander results in a much greater MPR (1.32). The difference between these MPRs becomes significant when one considers that the net gain per unit mass invested in the 6:1 lander case is only 7%, as compared to a 32% gain in the case of the 12:1 lander. The lower-mixture ratio lander is, in fact, marginal for use with the three-tankset OTV, as unforeseen difficulties could easily turn this small mass profit into a net mass loss. Mass payback ratios for the lower-mixture ratio lander configuration improve somewhat as the number of OTV tanksets increases. However, the MPRs for the 12:1-mixture ratio lander also increase by similar increments. The selected reference system, with three tanksets on the OTV and a 12:1-mixture ratio for the lander, is clearly indicated on Fig. 7 by the bold bar.

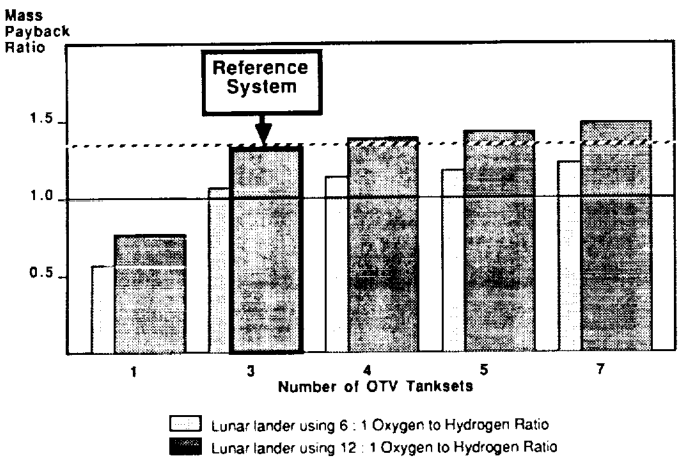

Fig. 7. Sensitivity to number of tanksets and lander O_2:H_2 ratio.

Aerobrake Weight Sensitivity and Potential Production from Lunar Materials

Aerobraking is essential to the success of the reference system, and the mass of the aerobrake is a dominant factor in its MPR. Figure 8 illustrates the sensitivity of MPR to aerobrake mass for the reference OTV, and for alternative configurations that use aerobrakes produced from lunar materials. Aerobrake mass is varied here as a percent of mass entering the Earth's atmosphere. Nominally, 13% of entry weight is used for the reference system's aerobrake, resulting in large aerobrake masses, as the returning OTV's mass (with nearly full oxygen tanks) is relatively large. Multiple aeropass trajectories, with each pass successively lowering perigee, might reduce the aerobrake mass required. If aerobrakes can be produced from lunar materials, substantially larger MPRs may result; the OTV would not have to carry the aerobrake mass from LEO to LLO, but the lander would instead carry the aerobrake mass for the much lower ΔV from the lunar surface to LLO (*Duke et al.,* 1985). If lunar aerobrake manufacture proves to be feasible (for example, using the TiO_2 by-product of ilmenite reduction as a refractory heat shield material), the aerobrake mass could be significantly higher than that of an aerobrake manufactured on Earth, and still be competitive. An expendable lunar aerobrake (discarded at LEO) weighing 25% of the entry mass would still be preferable to the reference system's aerobrake. If the used lunar aerobrake had intrinsic value in LEO (if the mass of the brake discarded at LEO is considered to be part of the payload to LEO), the MPR would continue to increase with increasing aerobrake weight. While the possibility of manufacturing aerobrakes from lunar materials is clearly attractive as a far-term option, the terrestrial aerobrake is retained as a baseline for the reference system.

Tether-assisted Transportation

Alternative systems that use tether-assisted OTV transportation have been emphasized in this Advanced Propulsion for LEO-Moon Transportation study (*Arnold and Thompson,* 1988; *Stern,* 1988). These systems are considered in the model as modifications of a reference transportation facility in LEO or LLO, or as an additional facility in an elliptical Earth orbit (EEO). Tether-assisted transportation alternatives are assumed to compensate for any net imbalance in momentum exchanged toward and away from the Moon through high-I_{sp} propulsion (e.g., ion engines) using propellant from the Moon.

Tether-assisted transportation systems can reduce the ΔV requirements of the vehicles in the reference transportation infrastructure, and thereby increase payload (multiple references). The ΔV supplied by throwing or catching the OTV or lander with a tether is subtracted from the velocity increment needed for a given mission phase. Velocity increments of 500 m/sec (1640 ft/sec) and 1 km/sec (3280 ft/sec) are considered for each tether system alternative. The tether that can throw (release) a vehicle with an initial 500-m/sec velocity, but not catch a similar incoming vehicle, is the least ambitious of the alternatives selected for study, and would be the most reasonable for consideration in "near-term" (early twenty-first century) transportation between LEO and the Moon. Tether-supplied velocity is limited to the maximum velocity increment needed, thus the "1-km/sec" system in LLO would throw an OTV toward Earth at 820 m/sec (2690 ft/sec), the velocity used to escape from LLO. Similarly, 95 m/sec (310 ft/sec) is the maximum increment achievable by system alternatives that catch an OTV for circularization in LEO after aerobraking.

Tether platforms can also provide a means of energy storage (*Arnold and Thompson,* 1988). Consider a platform in EEO with the capability to throw the OTV outward toward the Moon: The OTV uses chemical propulsion to transfer from LEO to EEO, docks with the tether facility, and is thrown by the tether. The momentum given to the mass of the OTV by throwing it at some initial velocity must be compensated by an equal and opposite change in the momentum of the platform in EEO (its mass multiplied by its ΔV). If the platform is heavy relative to the OTV, its resulting velocity change will be small, with little change in its orbital trajectory (a somewhat lower apogee if the OTV is thrown at perigee). Upon returning from LLO, the OTV aerobrakes into EEO, docks with the platform, and is then thrown downward into LEO, at the required remaining ΔV. The momentum of the EEO platform is now changed in the opposite direction

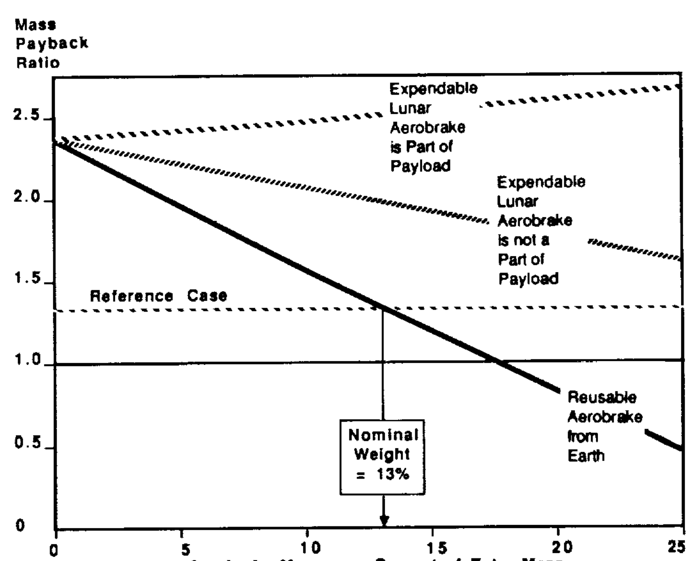

Fig. 8. Reference infrastructure: Aerobrake weight sensitivity.

(returning toward a higher apogee if the OTV is thrown at perigee). Energy transferred to the platform by the action of throwing the OTV toward LLO is thereby returned as the OTV is thrown down into LEO.

Similar momentum transfer could be achieved at a tether platform in LEO, which deorbits mass returning to Earth in exchange for upward boosting of OTVs toward the Moon, or at a platform in LLO, which exchanges momentum gained in the downward boost of lunar landers for the outward boost of OTVs returning to LEO. If platforms can be made to catch vehicles as well as throwing them, further improvements in energy storage can be obtained, with additional increases in MPR. While such transfers of momentum do not fully cancel in practice, the net momentum deficit or surplus is substantially reduced.

In a system with an MPR greater than one, the net momentum imbalance will tend to make the tether platform move toward the Moon as the net lunar mass transported by vehicles moves toward Earth. Momentum could be balanced by several methods, including (1) sending additional mass from Earth toward the Moon; (2) throwing vehicles at a lower velocity toward Earth than the velocity at which they are thrown toward the Moon; (3) conversion of orbital energy into other forms (e.g., into electrical energy) via an electrodynamic, conducting tether cutting through geomagnetic field lines; or (4) consumption of propellants at the affected platform.

Platforms equipped for tether-assisted transportation are presumed to use the fourth method noted, with low thrust, high I_{sp} propulsion to cancel any net momentum imbalance. The propellant for such momentum makeup is considered to be a lunar product and, for the purposes of the transportation model, is included as a part of the lunar O_2 produced and transported. Argon in lunar regolith is easily released by heating (*Kirsten and Horn,* 1974), and could be a reasonable propellant choice in place of O_2. An I_{sp} of 5000 sec is presumed for momentum makeup, consistant with the value used for ion engine OTV propulsion discussed later. As the net momentum deficit or surplus is generally small, MPRs are not very sensitive to this selection of advanced propulsion for the facilities equipped for tether-assisted transportation.

Figure 9 contrasts the MPR achieved through tether-assisted transportation from a single facility in LEO, EEO, or LLO. Each case considers two velocity increments supplied in a system that (1) only throws vehicles and (2) both throws and catches vehicles. While any of these alternatives is clearly better than the reference case, several interesting observations can be made through comparison of the alternatives with each other. The LEO tether facility gains little by adding the ability to catch due to the small velocity needed for circularization of the OTV in its low perigee orbit after aerobraking. (Tether-assisted transportation of mass between Earth and LEO has not been considered for the LEO OTSF due to the groundrules of the present study, but would tend to increase its effective MPRs.) The EEO tether facility, in contrast, would benefit considerably from the ability to catch vehicles in addition to throwing them. The increased MPRs for the EEO facility, however, must be traded against the increased operational complexities of such a system. Tether-assisted transportation from the LLO OTSF results in the largest MPRs for any single facility location, as the facility is used to reduce propulsive velocity requirements for the lunar lander as well as the OTV. Here the MPRs achieved by throwing alone equal or exceed those that would be obtained by combined throwing and catching from LEO or EEO facilities. The improvement in MPR that would result from an LLO facility that could catch as well as throw is also far more significant than that for an LEO or EEO facility.

At a high enough velocity, catching and throwing the OTV with a tether may be preferable to aerobraking (*Eder,* 1987). Figure 10 plots the MPR achieved with and without the use of an aerobrake vs. velocity supplied by tether for the case of a tether facility in EEO that can both throw OTVs and catch them. As calculated using the transportation model, the aerobrake becomes a detriment, rather than an asset, if the tether facility can impart a velocity of approximately 1.4 km/sec both in throwing and catching. At low tether-supplied velocities (below 0.7 km/sec), this type of system would be less effective than the reference infrastructure.

Laser Propulsion, Ion Engine, and Mass Driver Systems

Other modifications of the reference infrastructure with new technology could also increase MPR substantially. Figure 11 compares laser OTV propulsion, ion engine OTV propulsion, and a lunar mass driver as modifications to the reference system.

The laser propulsion case, as defined by R. Glumb of TRW, uses a laser to heat H_2 propellant for departure of the OTV from LEO.

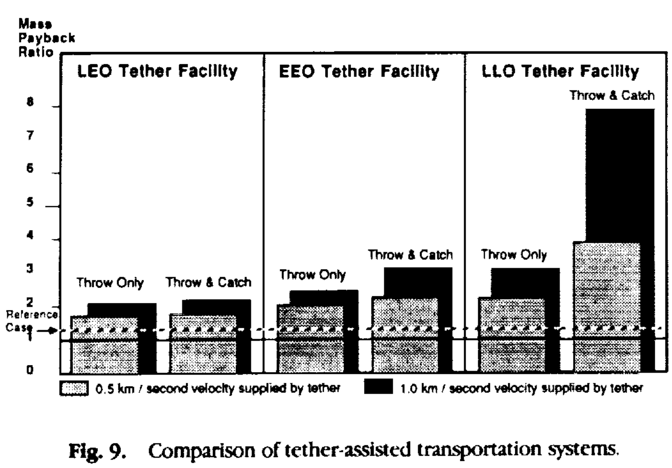

Fig. 9. Comparison of tether-assisted transportation systems.

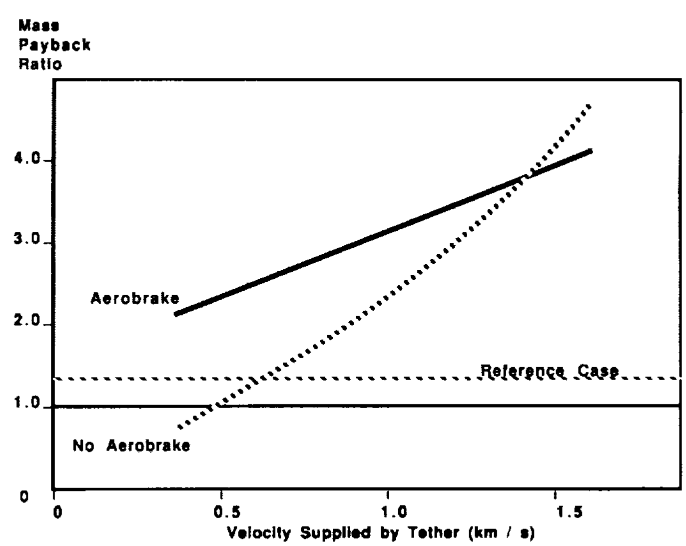

Fig. 10. EEO tether system: Aerobrake vs. no aerobrake.

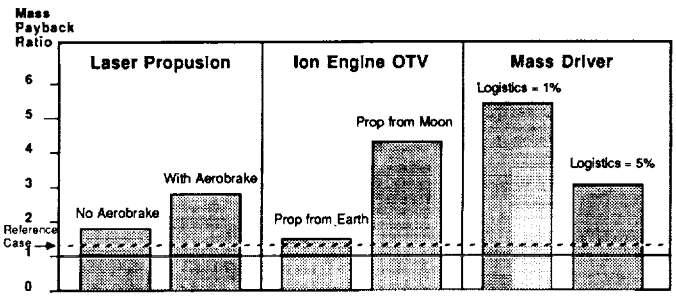

Fig. 11. Laser propulsion, ion engine, and mass driver systems.

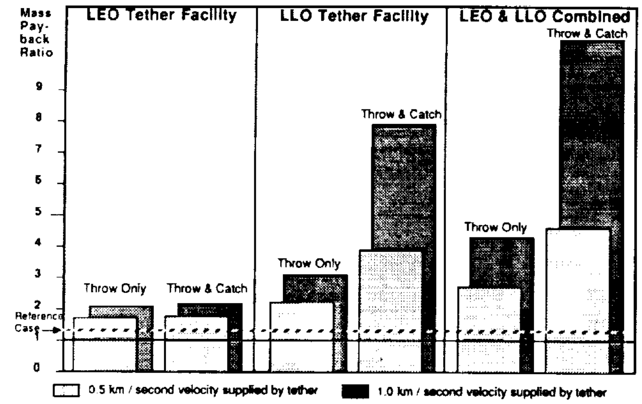

Fig. 12. Combined tether-assisted transportation in Earth and lunar orbits.

The propulsion system of the reference OTV is retained for use in the vicinity of LLO. This alternative results in a relatively high MPR if the aerobrake is retained, but a somewhat lower MPR if the aerobrake is relinquished in favor of carrying additional H_2 for laser propulsion in return to LEO.

An OTV equipped with an ion engine, as defined by Ralph Lovberg of UCSD's Physics Department, also achieves a very high MPR, provided that its propellant is supplied from the Moon. This vehicle has a large mass, no aerobrake, and low-thrust ion engines. The low thrust of the vehicle substantially increases the effective mission ΔV, as well as the mission duration. Use of an aerobrake in conjunction with ion engine propulsion was not considered, due to the presumption that a large power supply would be needed. Nuclear power safety implications or large, fragile solar cells could prohibit aerobraking. (For the purposes of the transportation model, OTV transportation reached LEO rather than being limited to a higher, "nuclear safe" altitude, which would have required a separate vehicle for intermediate transportation to LEO). If aerobraking were feasible, the mission duration and ΔV requirements for ion engine propulsion could be reduced substantially, with a corresponding increase in MPR.

A mass driver situated on the Moon would also result in a high MPR. Two cases are considered here through the transportation model, with logistics mass taken down to the Moon by the lander equaling nominal (1%) and increased (5%) fractions of lunar O_2 produced. An increase in logistics mass may be warranted, as the mass driver (as defined by Hu Davis of Davis Aerospace) launches O_2 payloads with apogee kick motors attached for self-circularization in LLO, and these motors are presumed to be imported from Earth. Propellant required for the collection of O_2 payloads in LLO would also result in an effective increase in logistic mass requirements.

Combined Tether Systems in LEO and LLO

Combined systems, where hanging or spinning tethers are used at two tether facilities in LEO and LLO, have been selected for investigation by the working groups involved in the Advanced Propulsion for LEO-Moon Transportation study. Hanging and spinning tether facilities are identical as evaluated in the transportation model. Results for this case would apply equally well to the use of swinging tethers, which may be another reasonable alternative.

Figure 12 illustrates the LEO and LLO systems alone (as they were shown in Fig. 11) and the combined system of tether-assisted transportation from both LEO and LLO. The MPR improves substantially through the combination of two similar or identical systems in LEO and LLO. The development cost of two such facilities should be a relatively small increase over that for a single facility to be placed in either LEO or LLO.

CONCLUSIONS AND RECOMMENDATIONS

The results produced by this LEO-Moon transportation model suggest that advanced technology can significantly improve the potential for lunar resource utilization in LEO. The reference LEO-Moon transportation infrastructure, using aerobraking OTVs, lunar oxygen, and high-mixture-ratio lunar lander engines, can deliver slightly more lunar mass to LEO than the mass of propellants and logistics needed from Earth for transportation and lunar oxygen production. New technologies of tethered momentum transfer, lunar material aerobrakes, laser OTV propulsion, ion engine OTV propulsion, and lunar mass driver use all have been seen to increase the efficiency of the reference system in bringing lunar mass to LEO.

In order to reap the benefits of such advanced technology, continuing research and development is needed. High-mixture-ratio lunar lander engines are important for efficient use of lunar oxygen, and deserve consideration in ongoing technology development activities. Conceptual design studies of LEO-Moon transportation systems should consider modifications over time as new technologies mature. Further investigation of advanced technology is necessary in the near term as an input to preliminary design for early LEO-Moon transportation systems. Continued consideration of such advanced systems is recommended to provide the groundwork for their eventual implementation in transportation between the Earth and Moon, as well as in regions beyond cislunar space.

Acknowledgments. Credit is due in several respects for assistance in development of this transportation model. Many of the participants in the Advanced Propulsion for LEO-Moon Transportation study made useful contributions in defining the reference configuration and advanced alternatives to be studied. R. Ford's performance calculations for high O_2:H_2 ratio propulsion were an important input to the reference lunar lander definition. W. Thompson provided guidance and insight into potential operations using tethers for momentum transfer. H. Bonesteel, M. Felix, A. Schneider, and M. Stern are thanked for their comments on earlier drafts of this paper. The research reported here fulfilled part of the requirements for a Master of Science degree in aerospace engineering at the University of California, San Diego. General Dynamics Space Systems Division assisted in related tuition expenses, computer resources, and arrangements for presentation and publication of this paper.

REFERENCES

Arnold J. R. and Thompson W. B. (1988) Advanced propulsion for LEO-Moon transportation: II. Tether configurations for the LEO-Moon system (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 10. Lunar and Planetary Institute, Houston

Bialla P. H. (1986) The diversity of roles for orbital transfer systems. 37th Congress of the International Astronautical Federation, Paper No. IAF 86-116, Innsbruck, Austria.

Bialla P. H. and Henley M. W. (1987) The space-based OTV and the establishment of the next launch site. 38th Congress of the International Astronautical Federation, Paper No. IAF87-196, England.

Bialla P. H. and Ketchum W. J. (1987) Key technologies for orbital transfer vehicles. *Proceedings of the 24th Space Congress*, Cocoa Beach, Florida.

Bock E. (1979) *Lunar Resources Utilization for Space Construction, Final Report*. Contract NAS9-15560.

Cordell B. M. and Wagner S. L. (1986) Strategies, synergisms, and systems for the mannned exploration of Mars. 37th Congress of the International Astronautical Federation, Paper No. 14F-86-320, Innsbruck Austria.

Cordell B. M., Varnado C. L., and McClain C. (1988) Implications of the NASA lunar initiative for a typical space transportation architecture (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 59. Lunar and Planetary Institute, Houston.

Duke M. B., Mendell W. W., and Roberts B. B., (1985) Strategies for a permanent lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 57-68. Lunar and Planetary Institute, Houston.

Eder D. (1987) Lunar base logistics. Paper presented at the Manufacturing in Space Conference, Princeton, New Jersey.

Frisbee R. H. and Jones R. M. (1983) An analysis of propulsion options for transport of lunar materials to earth orbit. 19th Joint Propulsion Conference, Paper No. AIAA-83-1344, Seattle, Washington.

Gibson M. A. and Knudsen C. W. (1985) Lunar oxygen production from ilmenite. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 543-550. Lunar and Planetary Institute, Houston.

Ketchum W. J., Maloney J. W., and Pea L. (1988) Servicing and maintaining a lunar transportation system in LEO (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 135. Lunar and Planetary Institute, Houston.

Kirsten T. and Horn P. (1974) ^{39}Ar-^{40}Ar dating of basalts and rock breccias from Apollo 17 and the malvern achrondrite. In *Soviet-American Conference on Cosmochemistry of the Moon and Planets*. NASA SP-370.

Loftus J. P. and Brashear W. L. (1985) Beyond low Earth orbit—An overview of orbit to orbit stages. 36th Congress of the International Astronautical Federation, Paper No. 1AF-85-141, Stockholm, Sweden.

Martin J. A. (1987) Space transportation main engines for single-stage vehicles. 23rd Joint Propulsion Conference, Paper No. AIAA-87-1941, San Diego, California.

Norman A. M., Briley C. L., and Evans S. (1988) Space station propulsion system test bed. 39th International Astronautical Federation Congress, Bangalore, India.

Robinson P. J. and Rosenthal S. E. (1986) A proven 25 lbf H_2/O_2 thruster for space station auxiliary propulsion. 22nd Joint Propulsion Conference, Paper No. A1AA-86-1560, Huntsville, Alabama.

Senneff J. M. and Richter G. P. (1986) A long-life 50 lbf H_2/O_2 thruster for space station auxiliary propulsion. 22nd Joint Propulsion Conference, Paper No. A1AA-86-1404, Huntsville, Alabama.

Stechman R. C. Jr. and Campbell J. G. (1973) Water electrolysis for Satellite Propulsion. AFRPL-TR-72-132.

Stern M. O. (1988) Advanced propulsion for LEO-Moon transportation: I. Overview (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 233. Lunar and Planetary Institute, Houston.

Stern M. O. (1989) *Advanced Propulsion for LEO-Moon Transportation, Final Progress Report*. NAG9-186, California Space Institute, Univ. of California, San Diego.

ENABLING LUNAR AND SPACE MISSIONS BY LASER POWER TRANSMISSION

N93-17423

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Applications are proposed for laser power transmission on the Moon. A solar-pumped laser in lunar orbit would beam power to the lunar surface for conversion into either electricity or propulsion needs. For example, lunar rovers could be much more flexible and lighter than rovers using other primary power sources. Also, laser power could be absorbed by lunar soil to create a hard glassy surface for dust-free roadways and launch pads. Laser power could also be used to power small lunar rockets or orbital transfer vehicles, and finally, photovoltaic laser converters could power remote excavation vehicles and human habitats. Laser power transmission is shown to be a highly flexible, enabling primary power source for lunar missions.

INTRODUCTION

Lunar bases and major space activities in the next century will substantially increase the demand for power. Present power technology such as solar photovoltaics, solar dynamics, and nuclear reactors can meet the near-term power requirements, but these technologies all have major disadvantages when scaled to high power levels with the additional demand of powering a very diverse set of remote lunar missions. One potential power scenario, not previously considered for lunar missions, is laser power transmission to multiple lunar users from a lunar orbiting solar-pumped laser (*De Young et al.*, 1987). Such a system allows a maximum of flexibility, since the primary power source is not located on the lunar surface. The laser beam could service a variety of users at diverse locations simultaneously as long as each user was in line of sight with the orbiting power station. The lunar receiver could convert the received laser power into electricity or propulsion with potentially greater than 50% efficiency in either case (*Walker and Heinbockel*, 1987a,b; *Jones*, 1981). This power concept would use relatively inexpensive and lightweight photovoltaics as the laser-to-electricity converter. The goal of laser power transmission is to make the receiver primary power system a minor component with respect to mass and complexity, and thus more emphasis can be placed on the science, materials processing, or other mission requirements at the lunar receiver.

This paper outlines in broad perspective a solar-pumped laser power system and several potential missions that such a power system would enable. The intent here is to stimulate further technical discussion of such a concept and to define both its advantages and disadvantages. We believe that this concept has potential for radical new missions not thought possible using more conventional primary power sources.

SOLAR LASER POWER STATION

A schematic diagram of a 1-MW solar-pumped laser is shown in Fig. 1. This is one of several potential solar-pumped laser concepts that could produce a laser beam of sufficient continuous wave (CW) power to meet the power needs of future space missions. In this specific design (*De Young et al.*, 1987), a 395-m-diameter solar collector focuses sunlight onto a laser cavity containing t-C_4F_9I. This molecule photodissociates and creates excited I, which lases at 1.315 μm. The laser beam is sent to a 28-m-diameter transmission mirror where the laser energy is beamed to the receiver of a lunar mission.

The power station would orbit the Moon at an altitude of approximately three lunar radii (5200 km). Since the lunar synchronous orbit is too high to be useful, multiple laser stations are necessary to cover the entire lunar surface. Also, it is not possible to avoid going into the Moon's shadow, as is possible in sun synchronous Earth orbits; thus, again, multiple stations are necessary. A receiver on the lunar surface could receive 6 hours of continuous power transmission before the laser station went below the horizon. A single laser station would then return

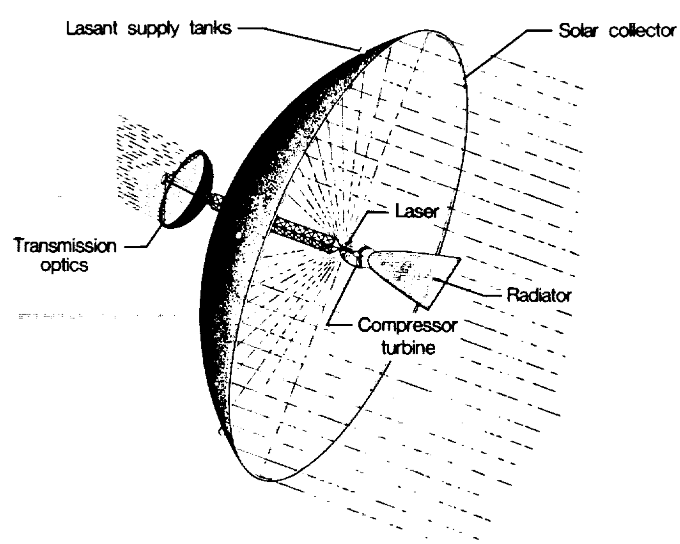

Fig. 1. An iodine solar pumped laser using t-C_4F_9I. From one to five of these laser stations would orbit the Moon and transmit power to the surface.

14 hours later. With multiple stations, this dead time (8 hours) could be substantially reduced, and with 5 stations, every lunar spot could be continuously covered with laser power.

The above description of the laser-power station is only an example. The exact nature of the power station will depend on the number of missions, the power requirement per mission, mission location, and mission duty cycle. The major advantages to such a system are that the prime power source (laser) is not taken to the lunar surface, thus saving mass, and, as a result of orbiting the surface, can supply power to diverse locations.

LUNAR APPLICATIONS OF LASER POWER

Research is continuing to define potentially more efficient solar laser systems. High-efficiency solar lasers would convert a significant fraction of the absorbed solar spectrum into laser light. Laser wavelengths should be shorter, i.e., nearer the visible spectrum, to minimize the size of the laser transmission optics. Pointing and tracking, as well as safety issues, also need to be addressed in a lunar laser power station.

A variety of missions are now proposed that give an indication of the diversity and flexibility inherent in laser power transmission. Each mission uses a simple photovoltaic laser-to-electric converter to power electric motors, storage batteries when necessary, and science or industrial mission requirements. Also, laser power could be used for direct lunar launch and orbit raising requirements.

Lunar Rover Power

Figure 2 shows a remote rover involved in lunar regolith science missions being powered by a laser beam. The rover uses a simple lightweight photovoltaic laser-to-electric converter of approximately 1-m diameter. Calculations have indicated that the laser-to-electric converter would have a mass of 20 kg for a 74 KW_e system. The radiator for this system is the dominant mass component at 240 kg for the 48% efficient converter. With such a system, more emphasis could be placed on mission requirements and less on the primary power system. The rover could be lightweight and very maneuverable and could maintain its mission for a considerably longer time than if powered by battery or fuel cells. It also does not have the radiation hazards associated with nuclear reactor power sources. The laser beam power level could be varied depending on real time power requirements, whereas with totally onboard power systems, the rover primary power system must be designed to the maximum power requirements, adding unnecessary weight to the rover for some missions.

Rover missions could be designed that demand power significantly greater than that available from radioisotope thermoelectric generators (RTGs), fuel cells, or batteries for enabling new mission scenarios.

Lunar Laser Propulsion

As materials processing, manufacturing, and habitats expand, transportation from the lunar surface to orbit will become a key activity and potentially a constraint on further lunar base expansion. Laser power transmission could have a dramatic impact on lunar transportation. In Fig. 3, a laser-powered rocket or orbital transfer vehicle (OTV) is shown. A laser beam is intercepted and used to heat hydrogen to approximately 15,000 K, producing a specific impulse (I_{sp}) of 1500 to 3000 sec (*Jones and Keeker,* 1981; *Krier and Glumb,* 1986). Such a rocket would not need to carry oxidizer, thus substantially reducing weight.

The thrust, F_T, of a laser propulsion engine is given by

$$F_T = 2\ P_{jet}/g\ I_{sp} \qquad (1)$$

where P_{jet} is the engine efficiency times the incident laser power; g, the acceleration due to gravity, is 9.8 m/sec^2; and I_{sp} is typically 1500 sec. For direct launch from the lunar surface, the thrust must be greater than the vehicle weight times g/6 where g/6 is the Earth acceleration of gravity divided by 6 for lunar environment. A calculation of the payload, m, that can be delivered to a very

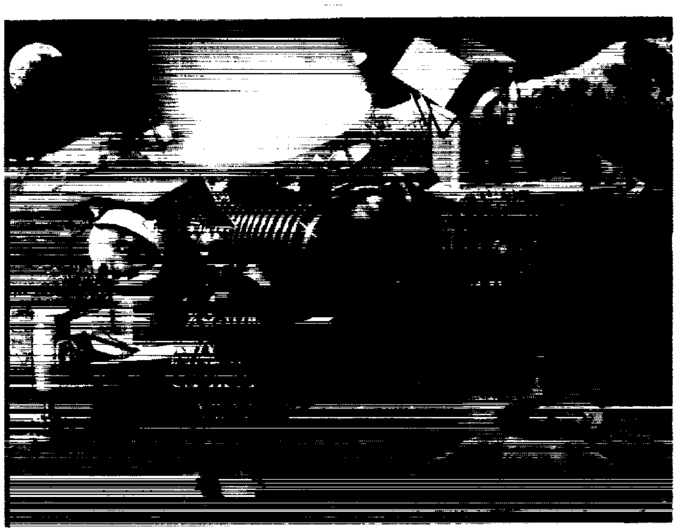

Fig. 2. A laser photovoltaic-powered lunar rover.

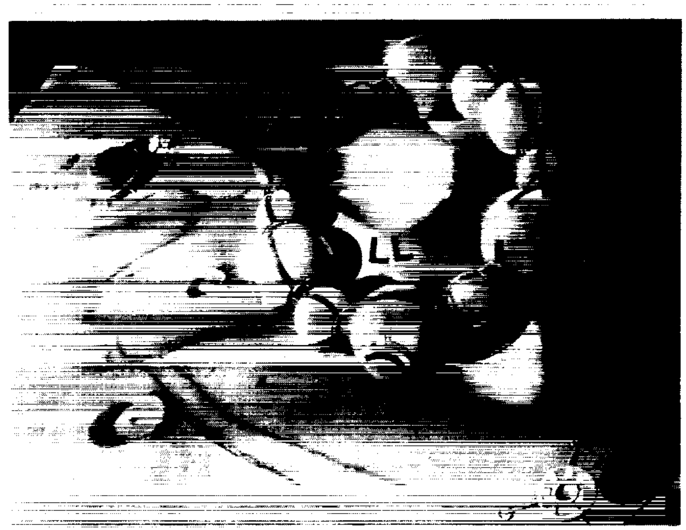

Fig. 3. A laser propulsion vehicle lifting high-value cargo from the lunar surface.

low lunar orbit (just above the surface, assuming that the engine mass in kilograms is equal to the engine thrust in N divided by 8, gives (*Frisbee*, 1984)

$$\frac{m\,(\text{payload})}{p\,(\text{laser})} = 25\,\frac{\text{kg}}{\text{MW}} \tag{2}$$

when 50% of the laser power, p, is converted into jet exhaust power by an engine with a I_{sp} of 1500 sec. If a 10-MW laser is available, then 250 kg of payload could be launched to low lunar orbit. From there, laser power would be used to change the orbit and inclination as required.

Figure 4 shows the relationship between the ratio of payload mass to laser power (kg/MW) and laser power (MW) for launching payloads from the lunar surface to low lunar orbit. The laser propulsion engine is assumed to be 50% efficient, that is, 50% of the incoming laser power is converted into kinetic energy of the H_2 propellent. Propulsion is shown for engine mass equal to $(F_T)^{1.15}/8$, $F_T/8$, and zero (*Frisbee et al.*, 1984).

Also shown is the time required to go from the lunar surface to lunar orbit for the zero engine mass case. Looking at the ideal case of engine mass equal to zero, we see the relationship between I_{sp}, launch time to low lunar orbit, and the amount of H_2 fuel per kilogram of payload expended. Going from an I_{sp} of 2000 sec to an I_{sp} of 1000 sec increases the engine thrust but also expends more H_2 fuel per kilogram of payload. Note that the launch time to low lunar orbit is nearly the same for all three cases. Thus, higher I_{sp} laser propulsion engines are desirable, since they more efficiently use the valuable H_2 fuel.

Lunar Surface Modification

Another laser application would use the beam directly to modify the lunar landscape. In the previous 1984 NASA/NAS Lunar Base Symposium, the desirability and potential applications of molten regolith were addressed in several presentations (*Rowley and Neudecker*, 1985; *Khalili*, 1985). Large numbers of rocket launches from the lunar surface can create dust clouds that can travel considerable distances, deposit material on solar panels, radiators, and other structures, and degrade their performance. Thus, some method for hardening the lunar surface is needed. In Fig. 5, a beam from the orbiting lunar power station is absorbed

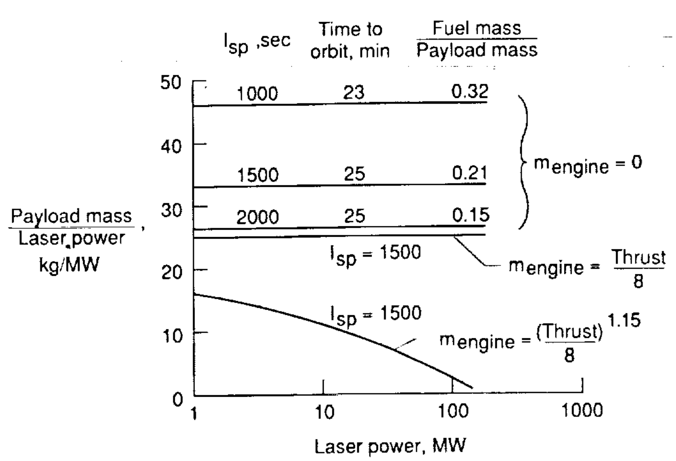

Fig. 4. Surface-to-low lunar orbit laser propulsion. Conversion efficiency = 50%.

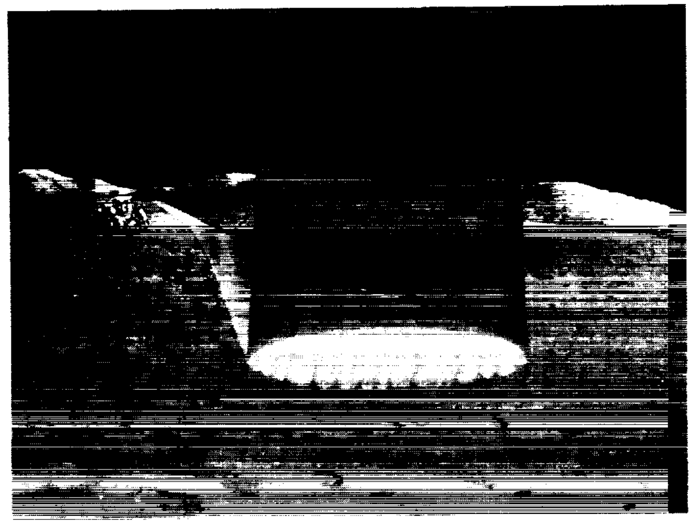

Fig. 5. A high-intensity laser beam heating the lunar surface to the melting point. After cooling, a hard glassy surface is created for roadways and launch pads.

by the lunar surface. The surface is heated to the melting temperature and then cooled, resulting in a thin, solid glassy surface. Such surfaces are dust free and thus could be used as "paved" surfaces for launch pads, roadways, and building sites. Also, glassmaking for construction and containers could be additional benefits.

A first-order analysis was made of lunar surfaces heated by laser beam absorption in order to indicate the thickness of the glass surface, assuming a laser input power of 1 MW. To characterize the time rate of change of surface temperature (T) as a function of depth (z), the following equation was used

$$\frac{\partial T}{\partial t} = \frac{1}{\rho C_p}\frac{\partial}{\partial z}\left(k\,\frac{\partial T}{\partial z}\right) \tag{3}$$

where ρ is the regolith density (2 g/cm^3), C_p is the specific heat (1.26 J/g-K), and k is the thermal conductivity (0.021 W/sec-cm-K). Nominal values of the thermal parameters k and C_p have been chosen based on lunar sample studies (*Stimpson and Lucas*, 1972), and for these preliminary calculations they are held constant. The initial condition is $T(z,0) = 350$ K, which is representative of lunar surface daylight temperatures. The boundary conditions are

$$k\left(\frac{\partial T}{\partial z}\right)_{z=0} = -q + \epsilon\sigma\,[\,T(0,t)]^4 \tag{4}$$

and

$$T_{z\to\infty} \rightarrow T(z,0) = 350\ \text{K} \tag{5}$$

where

$$q = 0.8P/\pi R^2 \tag{6}$$

It is assumed that 80% of the incident laser power, P (assumed to be 1-MW), is absorbed by the lunar soil in a circular area of radius R. The symbol ϵ is the emissivity (0.8), and σ is the Stefan-

Boltzmann constant. The maximum surface temperature is defined when the absorbed surface energy is equal to the reradiation energy or

$$T_{lim} = (q/\epsilon\sigma)^{1/4} \tag{7}$$

Figure 6 shows the results of calculations of surface temperature as a function of surface depth for absorbed energy of 800 kW in either a 1- or 0.75-m-diameter spot. Laser irradiation times extend from 10 to 40 min. T_{melt} is the typical melting temperature of the lunar soil (*Lin,* 1985); thus, temperatures above T_{melt} result in a hard glassy surface. For a 1-m spot size in Fig. 6 and a 10-min irradiation time, a glassy surface of approximately 2-cm depth can be achieved. This may be sufficient depth for roadways and launch pads, depending on vehicle mass. By increasing the energy density with a 0.75-m spot diameter, the irradiation time can be reduced for a given depth z. For larger lunar surface areas and greater depths, much higher laser powers are required.

Power for Lunar Base

Significant electrical power will be needed at the lunar base, on the order of 1 MW_e, to accomplish a variety of missions, including an oxygen production plant, mining and refining lunar materials, habitat life support systems, and transportation. Nuclear reactors with Sterling engines have been proposed to meet the power need. Such technology requires a fixed permanent lunar location with significant radiation protection problems. An alternative approach would be to put the solar- or reactor-driven laser in lunar orbit and beam the power to a variety of lunar locations.

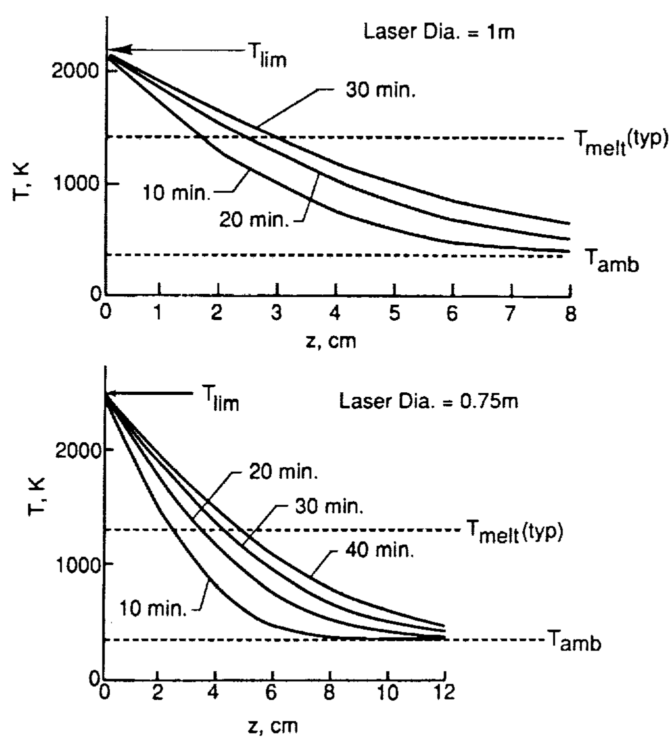

Fig. 6. Surface temperature as a function of surface depth.

In Fig. 7 a remote lunar site is being excavated by equipment powered by a laser beam. Since the vehicle power system is compact and lightweight, the excavation equipment could be highly mobile and remotely controlled. Temporary human habitats could also receive their power through laser transmission. The figure also shows a glass-brick-fabrication vehicle producing construction materials for future buildings.

A long term presence on the lunar surface will require extensive power for human habitats. Figure 8 shows such a major habitat, which houses approximately 100 people and is powered by a laser photovoltaic converter. The converter is a flat plate design mounted to track the lunar orbiting laser station. Two photovoltaic converters are needed to have continuous power as one laser station goes below the horizon and another comes above the opposite horizon. If multiple laser stations are not available, then energy storage must be added to the habitat.

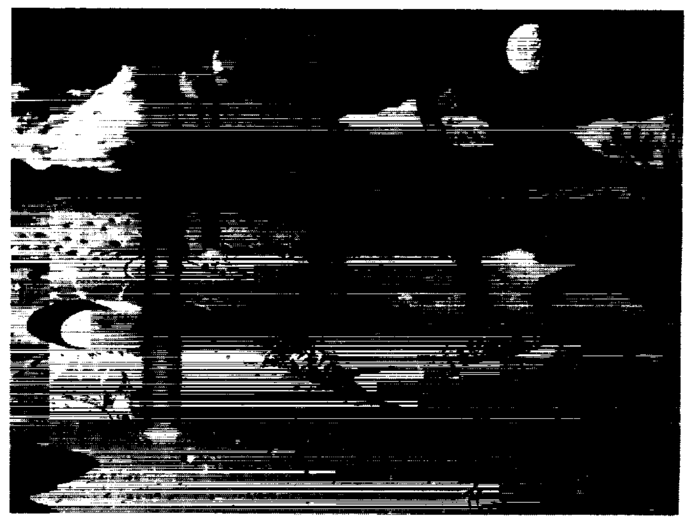

Fig. 7. Multiple laser beams powering excavation vehicles at a remote lunar site.

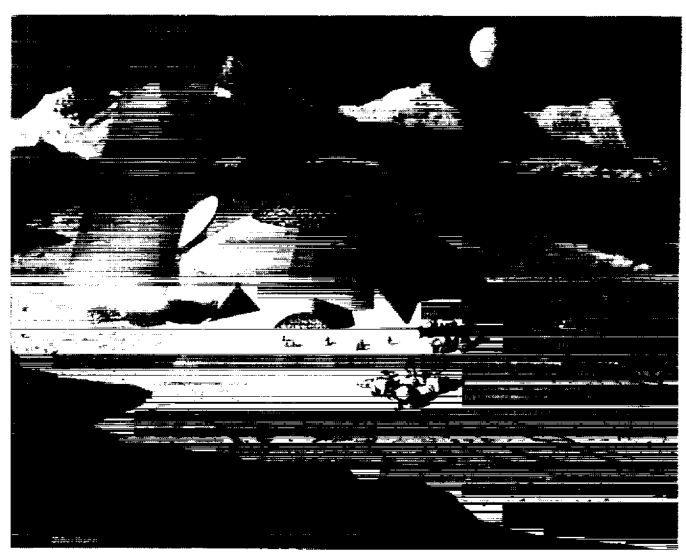

Fig. 8. A lunar human habitat powered by laser photovoltaics.

CONCLUSION

Laser power transmission to the lunar surface could power a wide variety of missions including prospecting rovers, surface-to-orbit propulsion, lunar surface hardening, excavation equipment, and human habitats. Using simple high power photovoltaic converters would permit more emphasis to be placed on the mission rather than its primary power requirements. Future studies may reveal other missions enabled by laser power transmission.

REFERENCES

De Young R. J., Walker G. H., Williams M. D., Schuster G. L., and Conway E. J. (1987) *Preliminary Design and Cost of a 1-Megawatt Solar-Pumped Iodide Laser Space-to-Space Transmission Station.* NASA TM-4002. NASA Langley Research Center, Hampton, Virginia.

Frisbee R. H., Horvath J. C., and Sercel J. C. (1984) *Space-Based Laser Propulsion for Orbital Transfer.* JPL D-1919. Jet Propulsion Laboratory, Pasadena.

Jones L. W. and Keeker D. R. (1981) *The NASA Laser-Propulsion Project: An Update.* AIAA-81-1248.

Krier H. and Glumb R. J. (1986) Studies of processes in laser-sustained plasmas for application to rocket propulsion. *Isr. Annu. Conf. Aviation and Astro.*, pp. 38-48. Haifa, Israel. A87-35001.

Lin T. D. (1985) Concrete for lunar base construction. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 381-390. Lunar and Planetary Institute, Houston.

Rowley J. C. and Neudecker J. W. (1985) In-site rock melting applied to lunar base construction and for exploration drilling and coring on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 465-477. Lunar and Planetary Institute, Houston.

Stimpson L. and Lucas J. W. (1972) Lunar thermal aspects from Surveyor data. *Therm. Charact. Moon, 28.*

Walker G. H. and Heinbockel J. H. (1987a) Parametric study of laser photovoltaic energy converters. *Solar Cells, 22,* 55-67.

Walker G. H. and Heinbockel J. H. (1987b) *Mathematical Modeling of a Photovoltaic-Laser Energy Converter for Iodine Laser Radiation.* NASA TM-100482. NASA Langley Research Center, Hampton, Virginia.

LUNAR ^{3}He, FUSION PROPULSION, AND SPACE DEVELOPMENT

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The recent identification of a substantial lunar resource of the fusion energy fuel ^{3}He may provide the first terrestrial market for a lunar commodity and, therefore, a major impetus to lunar development. The impact of this resource—when burned in D-^{3}He fusion reactors for space power and propulsion—may be even more significant as an enabling technology for safe, efficient exploration and development of space. One possible reactor configuration among several options, the tandem mirror, illustrates the potential advantages of fusion propulsion. The most important advantage is the ability to provide either fast, piloted vessels or high-payload-fraction cargo vessels due to a range of specific impulses from 50 sec to 1,000,000 sec at thrust-to-weight ratios from 0.1 to 5×10^{-5}. Fusion power research has made steady, impressive progress. It is plausible, and even probable, that fusion rockets similar to the designs presented here will be available in the early part of the twenty-first century, enabling a major expansion of human presence into the solar system.

INTRODUCTION

Recently, a connection between the Moon and future terrestrial energy needs was recognized: the lunar resource of the isotope helium-3 (^{3}He) can provide a clean and safe source of energy on Earth for centuries (*Wittenberg et al.,* 1986). Measurements of lunar regolith samples from the Apollo and Luna programs show significant quantities of ^{3}He (*Cameron,* 1991). The burning of ^{3}He with deuterium (D) as a fusion fuel has been known for many years to be attractive, but no significant terrestrial source has been found (*Miley,* 1976; *Dawson,* 1981; *McNally,* 1982). The present paper examines the implications of lunar ^{3}He for space development in the context of one possible fusion propulsion system and the capabilities it would provide.

The lunar ^{3}He resource is estimated to be $\sim 10^9$ kg (*Wittenberg et al.,* 1986; *Kulcinski et al.,* 1991). The presumed source of this ^{3}He is the solar wind; ^{3}He has been deposited on the lunar surface over the past 4 b.y. and spread a few meters deep into the regolith by meteorite bombardment. To put this resource into perspective, 10^9 kg of ^{3}He burned with D would provide 2000 years of present world energy consumption or, using the fusion rocket design discussed in this paper, would allow 10,000,000 one-way trips to Mars of 90-day travel time with 12,000-Mg (metric tonne) payloads.

Fusion reactors for space propulsion were first investigated in the 1950s, and the first D-^{3}He version was published in 1962 (*Englert,* 1962). Many of the concepts proposed in the early work remain valid. However, since that time, a great deal of progress has been made in understanding both the science and the technology of fusion energy. In particular, configurations have evolved and the sophistication of experimental, theoretical, and numerical tools has increased dramatically (*Post,* 1987).

After a brief examination of fusion fuel cycles, concentrating on their use in space, one potential fusion propulsion system will be described. The capabilities of such systems for increasing payload fractions or decreasing flight times will be assessed. The timeframe for fusion power development will be compared with that needed for a major human expansion into space, and the implications of the availability of D-^{3}He fusion propulsion on space development will be discussed. Finally, conclusions will be drawn.

FUSION FUEL CYCLES FOR SPACE APPLICATIONS

The main consideration in choosing a fusion fuel for space applications is the achievable specific power in terms of kilowatts of thrust per kilogram of total rocket mass. Therefore, the selection criteria are heavily weighted toward reactions producing a high fraction of power in charged particles—which may be converted to electricity at very high net efficiency (*Santarius,* 1987; *Santarius et al.,* 1987, 1988) or may be channeled by a magnetic field to provide direct thrust. Consequently, less heat must be rejected and radiator mass is reduced. A low fraction of energy in neutrons also allows substantial reduction in the mass of biological and magnet shielding.

Fusion fuel cycle physics has been extensively studied, and good summaries are available (*McNally,* 1982; *Dawson,* 1981). The most important fusion fuel cycles are based on the primary reactions given in Table 1. Of particular interest are the D-^{3}He fuel cycle, which produces 95% to 99% of its energy (including side reactions) in charged particles; the D-T cycle, which burns at the lowest temperature; and the D-D cycle, whose fuel is most plentiful on Earth. The "catalyzed" D-D cycle, in which the D-D fusion products T and ^{3}He are both subsequently burned, produces about the same energy fraction in neutrons as D-D, but achieves a power density comparable to D-^{3}He. Secondary and tertiary reactions with fusion products make the analysis of the ^{6}Li cycles difficult. However, detailed analyses (*McNally,* 1982) of the ^{6}Li cycles indicate that their power density is lower than the first three fuel cycles and that significant quantities of neutrons are produced by side reactions. The p-^{11}B reaction, although it gives no neutrons, is marginal for ignition, and would therefore produce almost all its power as thermal (bremsstrahlung)

TABLE 1. Primary reactions for the most important fusion fuel cycles (side reactions also occur, as do secondary and tertiary reactions with fusion products).

D + ^{3}He	→ p (14.68 MeV) + ^{4}He (3.67 MeV)	
D + T	→ n (14.07 MeV) + ^{4}He (3.52 MeV)	
D + D	→ n (2.45 MeV) + ^{4}He (0.82 MeV)	(50%)
	→ p (3.02 MeV) + T (1.01 MeV)	(50%)
^{3}He + ^{4}He	→ 2p + ^{4}He	(12.86 MeV)
p + ^{11}B	→ 3 ^{4}He	(8.7 MeV)
p + ^{6}Li	→ ^{3}He (2.3 MeV) + ^{4}He (1.7 MeV)	
D + ^{6}Li	→ five primary reactions, D-D reactions, ^{6}Li-^{6}Li reactions, and secondary (fusion-product) channels	

radiation. The ^{3}He-^{3}He reaction, although also neutron-free, has a very low cross section.

Figure 1 shows the approximate distribution of fusion power among charged particles, neutrons, and surface heat for the eventual energy loss of D-^{3}He, D-T, and catalyzed D-D plasmas, which differs from and is more relevant than the initial distribution of energy among reaction products. The D-^{3}He fuel cycle shows a clear advantage. This is diminished somewhat by a lower plasma power density (see Fig. 2), but the benefits of an efficient direct-thrust system over a thermal cycle for conversion of fusion energy to electricity and a further cycle to power ion thrusters, along with the reduction in shield mass, will be shown to lead to better performance from a D-^{3}He fusion propulsion system than from a D-T system.

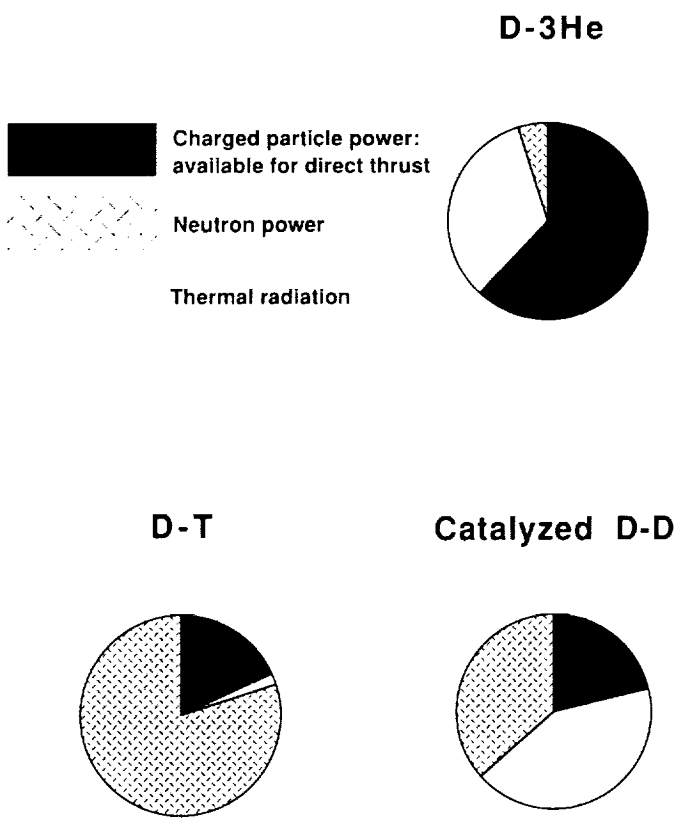

Fig. 1. Approximate distribution of energy loss among charged particles available for direct thrust, neutrons, and thermal radiation that appears as surface heat.

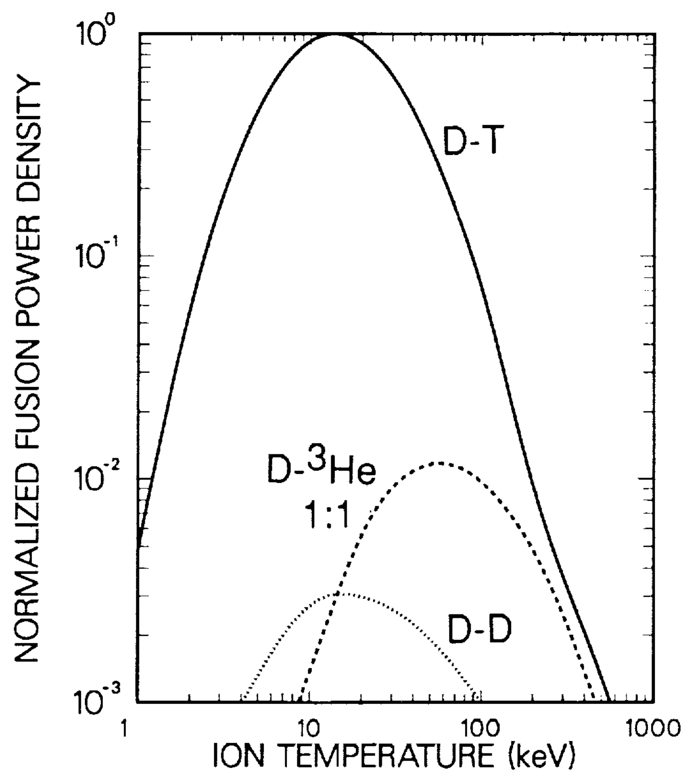

Fig. 2. Plasma power density for the major fusion fuel cycles.

ONE POTENTIAL FUSION PROPULSION SYSTEM DESIGN

Two key choices underpin a fusion rocket design: the fuel cycle and the configuration. Some of the earliest work on fusion propulsion, at NASA Lewis Research Center (*Englert,* 1962) and at Aerojet-General Nucleonics (*Hilton et al.,* 1964), applied essentially the same reasoning as in the present paper to identify linear fusion reactors burning D-^{3}He fuel as attractive options. In the intervening years, not only has the lunar ^{3}He resource been recognized, but fusion power research has undergone considerable evolution and, in particular, linear systems have progressed from the single-cell magnetic mirrors of the early 1960s to tandem mirrors (*Dimov et al.,* 1976; *Fowler and Logan,* 1977) and to thermal barrier tandem mirrors (*Baldwin and Logan,* 1979). This progression provides better confinement for the magnetic "bottle" at the cost of a more complicated containment scheme (see Fig. 3). Although a linear device will be used to illustrate D-^{3}He fusion propulsion's attractiveness here, toroidal devices also merit attention and some work on their design for space is extant (*Roth et al.,* 1972; *Borowski,* 1987).

A linear D-^{3}He fusion rocket has been designed by extrapolating from conceptual designs of D-^{3}He fusion reactors for power in orbit (*Santarius et al.,* 1988, 1989) and on Earth (*Santarius et al.,* 1987). The high efficiency of direct thrust and the reduced shield mass lead to a specific power value of ~1.2 kW/kg, based on the configuration shown in Fig. 3 and the parameters summarized in Table 2. Thrust is produced by driving one end cell more vigorously to increase axial confinement on that end, thereby unbalancing the end loss of plasma. All these coils are solenoids, and magnetohydrodynamic (MHD) stability is pre-

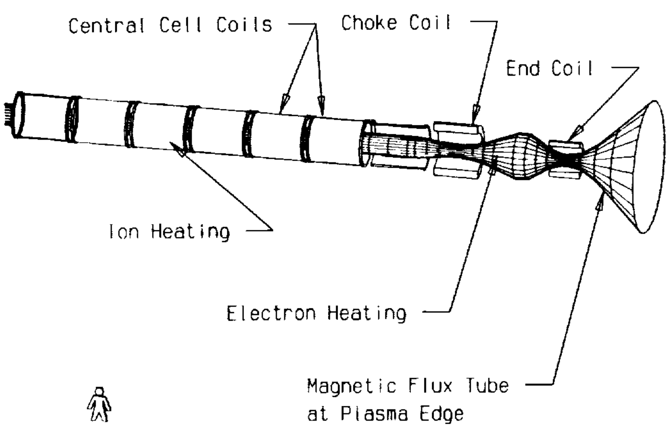

Fig. 3. Basic configuration for a thermal barrier tandem mirror reactor.

TABLE 2. D-^{3}He tandem mirror fusion propulsion system design parameters.

Parameter	Value
Thrust power per unit power system mass	1.2 kW/kg
Fusion power	1959 MW
Input power	115 MW
Thrust power	1500 MW
Thermal power (bremsstrahlung and synchrotron radiation, neutrons, plasma not usable for thrust)	574 MW
Neutron wall loading	0.17 MW/m^2
Total mass	1250 Mg (tonnes)
Total length	113 m
Central cell outer radius	1.0 m
Central cell on-axis magnetic field	6.4 T
Electron density	1.0×10^{21} m^{-3}
Helium-3 to deuterium density ratio	1
Electron temperature	87 keV
Ion temperature	105 keV
Fuel ion confinement time	6 sec
Ion confining electrostatic potential	270 kV

sumed to be provided by 25 MW of ion cyclotron range of frequencies power in the central cell. This is one method of several proposed to allow axisymmetric magnetic mirror machines to achieve MHD stability at high beta (ratio of plasma pressure to magnetic field pressure), and it has been demonstrated experimentally at low density and temperature (*Breun et al.,* 1986). The magnet shield material is LiH, and the magnets in the central cell are made of NbTi superconductor. Higher-field magnets are required for the end cells: on each side are one 12-T (on-axis) Nb_3Sn magnet and one 24-T magnet whose field is generated by 16 T from Nb_3Sn superconductor and 8 T from a normal-conducting Cu insert that requires 8 MW of power.

An important aspect of fusion propulsion is the flexibility inherent in the abilitv to tailor the thrust program to a wide variety of missions. This flexibility stems from three main operating modes: direct exhaust, mass-augmented exhaust, and thermal exhaust. These modes are shown schematically in Fig. 4. Typical burning plasma temperatures are 40-100 keV (500-1200 million K), so that exhausting the plasma directly would lead to extremely high specific impulses (exhaust velocity divided by standard Earth surface gravity) of about 10^6 sec. Lower specific impulses are also available, ranging continuously from about 10^5 sec to about 200 sec at thrust-to-weight ratios ranging from about 3×10^{-4} to 0.03, as shown in Fig. 5. The midrange is reached by adding a low-field magnet onto the end of the device and injecting matter, which is ionized by the end-loss plasma

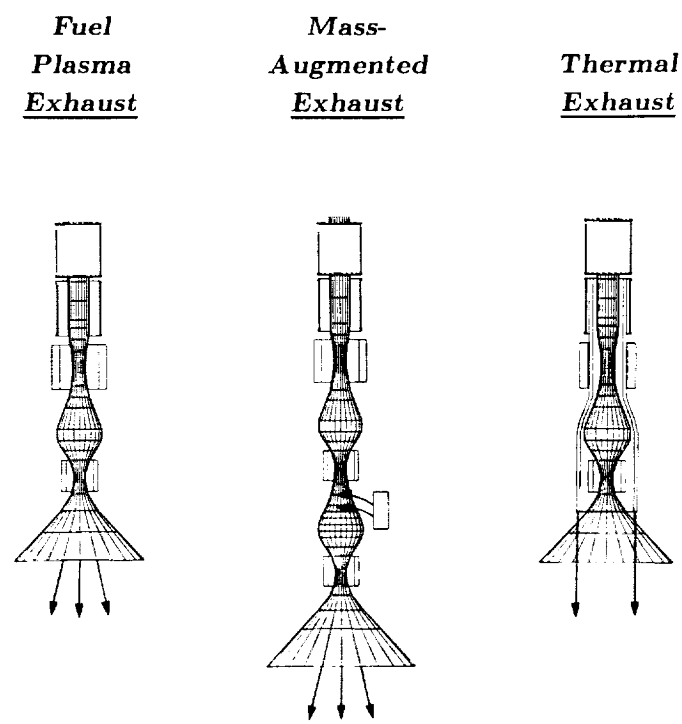

Fig. 4. Thrust mode options for a linear fusion propulsion system.

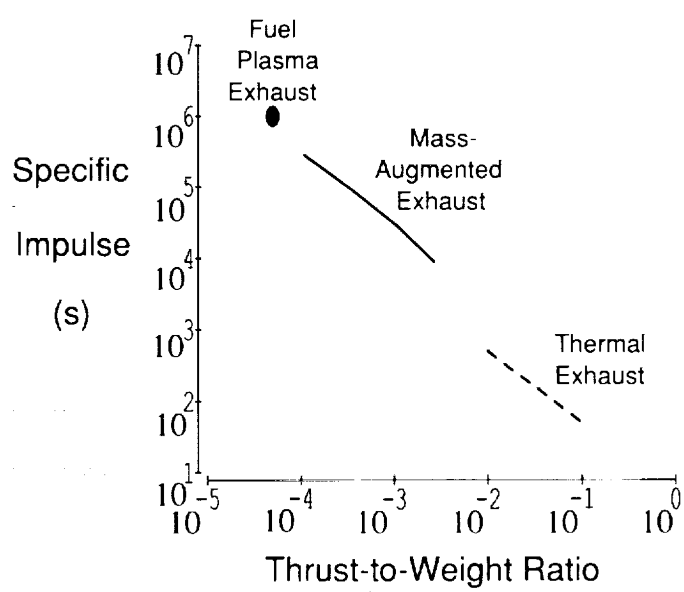

Fig. 5. Range of specific impulses and thrusts available from the fusion propulsion system discussed in this paper.

energy. The new cell would have a higher field on the rocket side than on the space side, creating a magnetic mirror in which ions reflect a few times off the magnetic field axial gradients (mirrors) before they collisionally scatter into the mirror "loss cone" and produce thrust. This process, which derives from the well-verified basic principle (adiabatic confinement) of magnetic mirrors, lowers the exhaust plasma temperature and increases the thrust. Higher thrust can be achieved by heating a gas with thermal (bremsstrahlung and synchrotron) radiation in a blanket surrounding the plasma and then exhausting the gas. Parameters typical of chemical systems, limited by materials considerations to about 1600 K, are available from this mode.

CAPABILITIES OF FUSION PROPULSION

The benefits of high specific impulse and continuous thrust, even at low thrust-to-weight ratios, have been known since the early 1950s, and detailed discussions of trajectory optimization are summarized in the classic references by *Ebricke* (1962) and *Stublinger* (1964). Although more total energy is required compared to chemical systems, much less fuel mass is needed and trip times can be shortened or payload mass fractions (payload mass/initial rocket mass) can be increased. The fusion propulsion system of the previous section, which produces power at ~1.2 kW/kg, can thus provide either fast human transport or large-payload-ratio cargo vessels. Using *Stublinger's* (1964) simile, these are like sports cars or trucks.

Fusion propulsion's capabilities are best illustrated by comparison with the primary chemical propulsion mode: minimum-energy, elliptical trajectories (Hohmann orbits). The calculations are based on *Stublinger* (1964) and are optimized assuming an acceleration of constant magnitude, but optimized direction. For a 1-kW/kg system and a 90-day, one-way, Earth-Mars mission, that assumption requires tuning the specific impulse over a range of 10,000 sec to 200,000 sec, which Fig. 5 shows to be attainable with the mass-augmented exhaust mode. Figure 6 shows the sports car mode and gives flight time for the same payload fraction, while Fig. 7 gives payload fraction for the same flight time—the truck mode. These figures show that fusion propulsion performs approximately as well as chemical systems even for low Earth orbit (LEO)/Moon missions, and far surpasses chemical propulsion performance for missions to Mars or Jupiter. For Earth-Mars missions, the trade-off between payload fraction and trip time is plotted in Fig. 8 (based on *Stublinger*, 1964).

Deuterium-helium-3 fuel possesses an extremely high energy density (19 MW-yr/kg), surpassed only by matter/antimatter, and is the highest energy density fuel presently known of those that

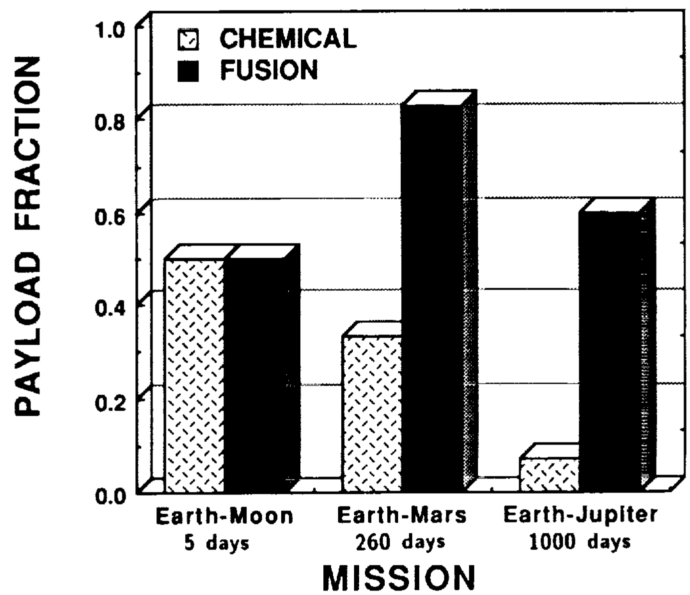

Fig. 7. Payload fraction for the same flight time (truck mode).

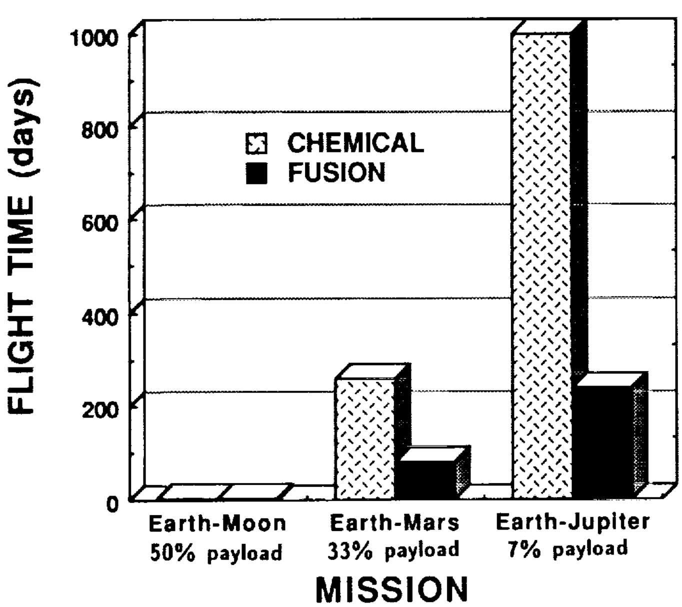

Fig. 6. Flight time for the same payload fraction (sports car mode).

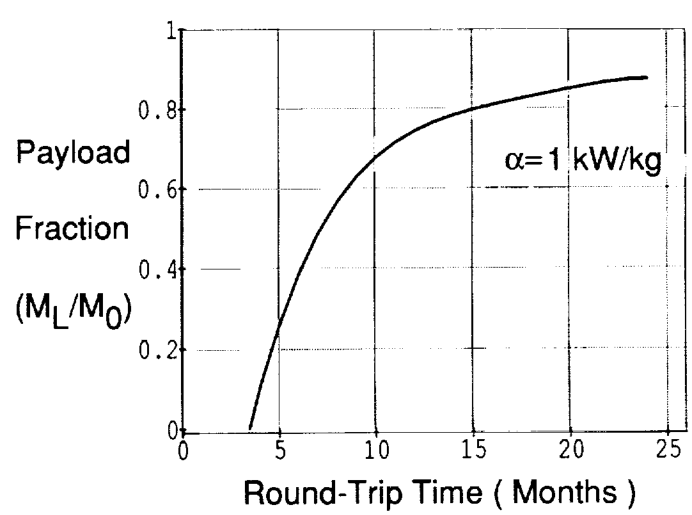

Fig. 8. Payload fraction vs. round-trip flight time for an Earth-Mars mission.

release more energy than is required to procure them. Once a fusion rocket is constructed in orbit, much of its mass will be reusable. A chemical rocket, with most of its mass in fuel/propellant, will require much more mass to be placed in orbit for each mission than will a fusion rocket, which uses negligible fuel mass and considerably less propellant mass. Mass requirements for an Earth-Mars round trip are compared in Table 3. Transporting 12,000 Mg between Earth and Mars would require orbiting an extra 47,000 Mg for chemical rockets and 3000 Mg for D-^{3}He fusion rockets.

Few constraints exist on the type of matter used as propellant in the mass-augmented mode of a fusion system; local sources such as regolith could probably be used because plasmas are hot enough to ionize almost all matter. Fusion's advantage would then be increased, since propellant for the return trip would not need to be carried. The high energy density of D-^{3}He also enhances the flexibility of a fusion propulsion system, since a reserve of fuel could easily be carried without a substantial rocket mass increase.

TABLE 3. Masses required for fusion and chemical transport between Earth and Mars, assuming a nine-month trip time each way.

	Chemical	D-^{3}He Fusion
Payload (each way)	11,800 Mg	11,800 Mg
Propellant	47,200 Mg	2,000 Mg
Fusion reactor	—	1,000 Mg
D-^{3}He fuel burned	—	0.08 Mg
Nonpayload mass orbited	47,200 Mg	3,000 Mg

FUSION POWER DEVELOPMENT TIMEFRAME

A key question in discussing space applications of fusion energy is whether fusion could be developed on the timescale required for a major human thrust into the solar system. Fusion progress over the past 30 years is illustrated in Fig. 9, where experimentally achieved values of the product of the three most important fusion physics parameters (plasma temperature, electron density, and energy confinement time) are plotted vs. time. The requirement for an ignited plasma, whose energy losses are sustained by the fusion power it produces, is also shown. Although the next step is by no means a trivial one and other important issues exist besides these three parameters, the six orders of magnitude already overcome suggest that the remaining hurdles can at least plausibly be surpassed on the timescale required by present space development plans (*National Commission on Space,* 1986).

The present terrestrial fusion research program, however, is focused mainly on the D-T fuel cycle because it is easier to ignite than is D-^{3}He. This is shown in Fig. 10, where curves are given for ignition of D-T and D-^{3}He against losses due to the finite plasma energy confinement time and bremsstrahlung radiation. Experimentally attained values of plasma temperature vs. the confinement parameter $n\tau_E$ are also plotted. The physics requirements on temperature and energy confinement are each about a factor of 4 higher for D-^{3}He than for D-T. Another difficulty in the context of this paper is that budget considerations have focused the present Department of Energy development plan for terrestrial fusion reactors on the tokamak—a toroidal system (*U.S. Congress OTA,* 1987). However, substantial progress on linear systems and other toroidal configurations had been made (*Callen et al.,* 1986) and a small effort remains, so a strong foundation exists.

Fortunately, the development of D-^{3}He fusion power promises to be much easier than the previous paragraph suggests. The key consideration is that, although the physics development for D-^{3}He fusion will be more difficult than for D-T, the reactor technology development will be faster and easier. The demonstration of D-^{3}He physics, suggested by *Atzeni and Coppi* (1980) and by *Emmert et al.* (1989) as possible even in next-generation D-T experimental test facilities, could quickly lead to a prototype, power-producing, D-^{3}He reactor. Sufficient ^{3}He exists on Earth for this purpose (*Wittenberg et al.,* 1986). Specifically, materials are already known that have been demonstrated to withstand the lower neutron fluence of D-^{3}He reactors, whereas materials

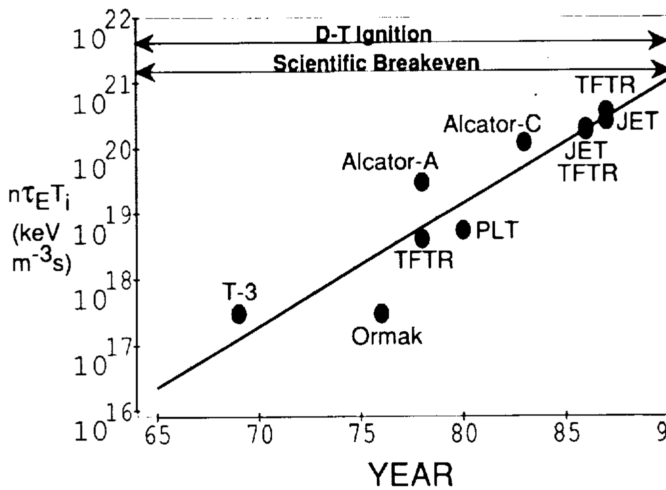

Fig. 9. Experimentally achieved parameter progress in fusion research.

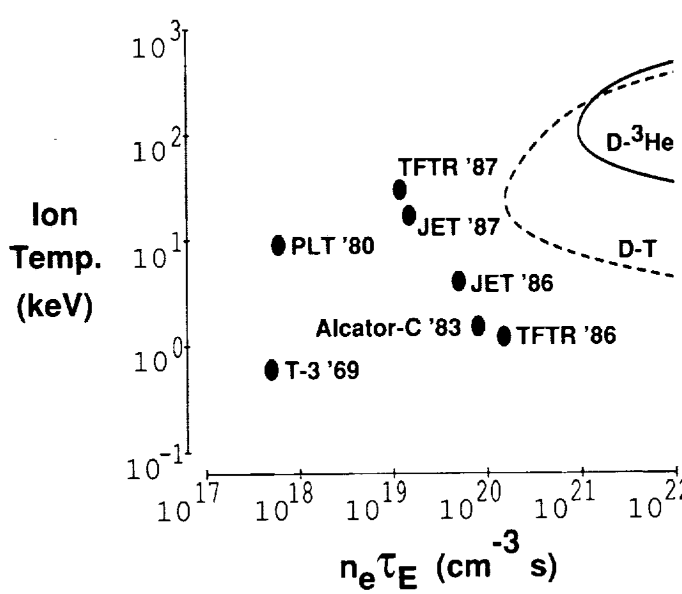

Fig. 10. Plasma ignition requirements for D-T and D-^{3}He plasmas.

C-2

suitable for the high neutron fluence of D-T reactors remain to be identified and would require an additional test device (or separate demonstration program). Also, the breeding of T fuel in a "blanket" surrounding the plasma requires considerable development and testing. There appear to be only a few areas where D-^{3}He propulsion systems could not rely on developed materials and technology. These include fueling, plasma current drive, and high-heat-flux materials. All these issues will be similar for D-^{3}He and D-T; they will, therefore, be addressed within the present D-T fusion program.

IMPLICATIONS FOR SPACE DEVELOPMENT

The development of terrestrial D-^{3}He fusion power will have an enormous impact on Earth's energy future and on lunar development. In space, D-^{3}He fusion will be an enabling technology for a large-scale human presence beyond Earth orbit, and the eventual impact may be even greater than on Earth. The high performance and flexibility of fusion propulsion will greatly expand the options available in building a major space infrastructure as the need for such systems begins to gain prominence early in the twenty-first century.

A fleet of fusion rockets could provide much of the "Bridge Between Worlds" of the *National Commission on Space* (1986). Figure 11 illustrates some potential space applications of fusion propulsion and power. It also shows the use of important by-products of ^{3}He mining, the other released gases such as CO_2 and N_2 for life support (*Bula et al.,* 1991). These rockets would vary only modestly in design, but would operate in the optimal thrust mode for a given mission, carrying humans quickly or cargo efficiently throughout the solar system. Although D-^{3}He fusion would provide high performance for large-scale operations beyond Earth orbit, present designs are inherently low thrust-to-weight systems, and alternatives would be required for surface-to-orbit operations except on asteroids and small moons. The specific D-^{3}He fusion system discussed in this paper remains attractive down to powers of ~100 MW, but other fusion configurations or nonfusion sources would be needed at low power.

Noteworthy for operations in the outer solar system is that D-^{3}He fuel is more abundant than any fuel except the proton-proton fuel of stars. Assuming a primordial composition, the gas giant planet mass fractions are approximately 10^{-5} ^{3}He and 3×10^{-7} D (*Weinberg,* 1972). Unfortunately, it appears that the probability of finding fossil fuels in the solar system beyond Earth is

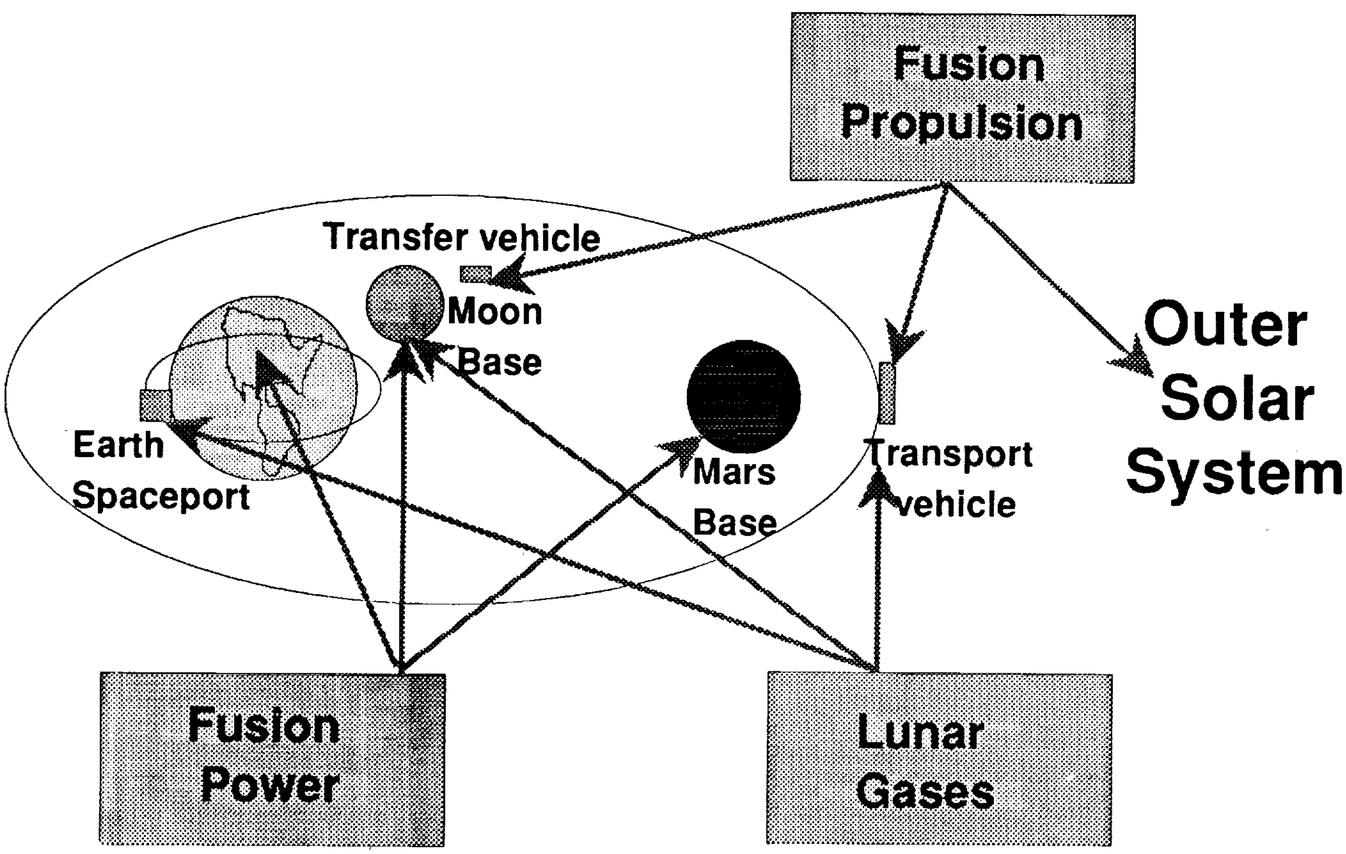

Fig. 11. The potential impact of D-^{3}He fusion on the "bridge between worlds" discussed by the *National Commission on Space* (1986).

very small, and the processing of fissile fuel, even if it exists in relative abundance, will require a massive and complex technology. On the timescale that a small percentage of the lunar surface can supply ^{3}He—a few hundred years—it is reasonable to anticipate development of the technology required to access the enormous quantities of D and ^{3}He in the gas giants.

Fusion propulsion, therefore, will dominate future transportation throughout the solar system. For missions beyond the Moon, where chemical systems quickly become inefficient in both payload fraction and trip time, fusion represents a key enabling technology.

CONCLUSIONS

The main conclusions of this analysis of the space applications of D-^{3}He fusion power are

1. Deuterium-helium-3 fusion will provide safe, efficient propulsion, offering a wide range of options—from fast, pilot missions to slower, cargo transport.
2. Linear systems most obviously provide an efficient means of producing direct thrust, but numerous options are likely to develop, and toroidal configurations also appear promising. The linear rocket design presented in this paper would provide a specific power of ~1.2 kW/kg.
3. The D-^{3}He fusion fuel cycle possesses distinct advantages over other candidate fusion fuel cycles, fission, and chemical systems for space applications.
4. Fusion power using D-^{3}He can be developed on a timeframe consistent with space development needs.
5. D-^{3}He fusion propulsion will enable a major expansion of human presence into the solar system.

REFERENCES

Atzeni S. and Coppi B. (1980) Ignition experiments for neutronless fusion reactions. *Comments Plasma Phys. Controlled Fusion, 6,* 77.

Baldwin D. E. and Logan B. G. (1979) Improved tandem mirror fusion reactor. *Phys. Rev. Lett., 43,* 1318.

Borowski S. K. (1987) A comparison of fusion/antiproton propulsion systems for interplanetary travel. In *Proc. 23rd Joint Propulsion Conf.,* Paper No. AIAA-87-1814.

Breun R. A., Brooker P., Brouchous D. A., Browning J., Butz G., et al. (1986) Stabilization of MHD modes in an axisymmetric magnetic mirror by applied RF waves and initial results of Phaedrus-B. In *Plasma Physics and Controlled Nuclear Fusion Research 1986, Vol. 2* (Proc. 11th Intl. Conf., Kyoto), p. 263. IAEA, Vienna.

Bula R. J., Wittenberg L. J., Tibbitts T. W., and Kulcinski G. L. (1992) Potential of derived lunar volatiles for life support. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Callen J. D., Santarius J. F., Baldwin D. E., Hazeltine R. D., Linford R. K., et al. (1986) *TPA Plasma Science Status Report.* Univ. of Wisconsin, Madison. 441 pp.

Cameron E. N. (1992) Helium mining on the Moon: Site selection and evaluation. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Dawson J. D. (1981) Advanced Fusion Reactor. In *Fusion, Vol. 1* (E. Teller, ed.), p. 453. Academic, New York.

Dimov G. I., Zakaidakov V. V., and Kishinevsky M. E. (1976) Thermonuclear confinement with twin mirror system. *Fiz. Plasmy, 2,* 597.

Ehricke K. A. (1962) *Space Flight, Vol. II: Dynamics.* Van Nostrand, Princeton. 1210 pp.

Emmert G. A., El-Guebaly L. A., Kulcinski G. L., Santarius J. F., et al. (1988) Possibilities for breakeven and ignition of D-^{3}He fusion fuel in a near-term tokamak. *Nucl. Fusion, 29,* 1427.

Englert G. W. (1962) Towards thermonuclear rocket propulsion. *New Sci., 16,* #307, p. 16 (October 4).

Fowler T. K. and Logan B. G. (1977) The tandem mirror reactor. *Comments Plasma Phys. Controlled Fusion, 2,* 167.

Hilton J. L., Luce J. S., and Thompson A. S. (1964) Hypothetical fusion propulsion vehicle. *J. Spacecr., 1,* 276.

Kulcinski G. L., Cameron E. N., Santarius J. F., Sviatoslavsky I. N., Wittenberg L. J., and Schmitt H. H. (1992) Fusion energy from the Moon for the 21st century. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

McNally J. R. Jr. (1982) Physics of fusion fuel cycles. *Nucl. Technol./Fusion, 2,* 9.

Miley G. H. (1976) *Fusion Energy Conversion.* American Nuclear Society, La Grange Park, Illinois. 454 pp.

National Commission on Space (1986) *Pioneering the Space Frontier.* Bantam, New York. 211 pp.

Post R. F. (1987) The magnetic mirror approach to fusion. *Nucl. Fusion, 27,* 1579.

Roth J. R., Rayle W. D., and Reinmann J. J. (1972) Fusion power for space propulsion. *New Sci., 54,* #792, p. 125 (April 20).

Santarius J. F. (1987) Very high efficiency fusion reactor concept. *Nucl. Fusion, 27,* 167.

Santarius J. F., Attaya H., Corradini M. L., et al. (1987) Ra: A high efficiency, D-^{3}He, tandem mirror fusion reactor. In *Proc. 12th Symposium on Fusion Engineering,* p. 252. IEEE Catalog No. 87CH2507-2.

Santarius J. F., Kulcinski G. L., et al. (1988) SOAR: Space orbiting advanced fusion power reactor. In *Space Nuclear Power Systems 1987* (M. S. El-Genk and M. D. Hoover, eds.), pp. 167-176. Orbit, Malabar, Florida.

Santarius J. F., Kulcinski G. L., et al. (1989) Critical issues for SOAR: The space orbiting advanced fusion power reactor. In *Space Nuclear Power Systems 1987* (M. S. El-Genk and M. D. Hoover, eds.), pp. 161-167. Orbit, Malabar, Florida.

Stuhlinger E. (1964) *Ion Propulsion for Space Flight.* McGraw-Hill, New York. 373 pp.

U.S. Congress, Office of Technology Assessment (1987) *Starpower: The U.S. and the International Quest for Fusion Energy.* U.S. Government Printing Office, Washington, DC. 236 pp.

Weinberg S. (1972) *Gravitation and Cosmology.* Wiley, New York. 657 pp.

Wittenberg L. J., Santarius J. F., and Kulcinski G. L. (1986) Lunar source of ^{3}He for commercial fusion power. *Fusion Technol., 10,* 167.

THE TRANSPORTATION DEPOT— AN ORBITING VEHICLE SUPPORT FACILITY

N93-17425

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This paper describes the details of an effort to produce conceptual designs for an orbiting platform, called a transportation depot, to handle assembly and processing of lunar, martian, and related vehicles. High-level requirements for such a facility were established, and several concepts were developed to meet those requirements. By showing that the critical rigid-body momentum characteristics of each concept are similar to those of the dual-keel space station, some insight was gained about the controllability and utility of this type of facility. Finally, several general observations were made that highlight the advantages and disadvantages of particular design features.

INTRODUCTION

Events of the last few years, including the Challenger disaster, reorganization of the space station program, and the usual funding problems for NASA, have shown the need for new goals in the national space program. Establishment of a permanently manned lunar base and a manned mission to Mars are the two goals most often cited as ways to reinvigorate this country's space program and reestablish our leadership in space-based operations.

Nearly every scenario that has been proposed for lunar bases and missions to Mars make extensive use of low-Earth-orbit (LEO) facilities, such as the space station, for assembly and maintenance of vehicles, storing of propellant, and temporary crew billeting. For both the Mars and lunar missions, however, the level and duration of support that is necessary poses potential problems for currently planned space station science and materials processing activities. There is, in fact, a fundamental conflict between activities that require a quiescent environment, such as microgravity research or high-precision astronomical measurements, and those activities, such as vehicle processing, that produce potentially large dynamic disturbances. This is not to say, of course, that the two types of activities are hopelessly incompatible. It simply means that if the two types of activities are present on the same facility, either the science and microgravity activities will be forced to deal with a less than ideal environment or the vehicle support activities will have to be curtailed to avoid disturbing those more sensitive users. The obvious solution is to separate as many conflicting activities as possible by either moving the sensitive users to a coorbiting facility, or developing a facility specifically for the needs of lunar and Mars mission support.

In recent months the space station office at NASA Langley has sponsored wide-ranging lunar base and Mars mission system analysis studies. These studies have shown, among many other things, that the current space station design is capable of supporting vehicle processing, but that the necessary modifications would adversely impact both the astronomical viewing and the microgravity environment to the point where it would be highly desirable to separate those sensitive users from the vehicle support facilities. This paper describes results of attempts to develop requirements and preliminary concepts for an LEO facility, called a transportation depot, to support assembly and maintenance of vehicles for lunar and Mars missions. Future studies will refine the concepts, develop growth scenarios, and perhaps consider the implications of the opposite alternative—developing a separate science and microgravity research facility.

VEHICLE ACCOMMODATION OPTIONS

The lunar and Mars missions envisioned for the next century are unlike any other planned space activity in that the mass and size of the vehicles, propellant, and support facilities are orders of magnitude greater than anything previously proposed for attachment to the space station. Furthermore, the length of time over which space station support is needed will make it impossible for other users to avoid dealing with the disturbances that are produced. For example, space station support for the lunar base described in *Weidman et al.* (1987) begins in the middle of the next decade and continues well into the next century, with a typical on-orbit mass of vehicles, propellant, and support equipment totaling nearly 300,000 kg (660,000 lbm). In contrast, the entire dual-keel station mass is only 209,000 kg (460,000 lbm). Similarly, the Mars mission described in *Cirillo et al.* (1988) entails a decade or more of support with, at one point, an on-orbit mass of 1,112,000 kg (2,448,000 lbm) over and above that of the station. As shown in the studies referenced, the magnitude and duration of support for lunar and Mars missions will make it very difficult to produce conditions acceptable for the needs of science and materials-processing users. It is clear, then, that despite the ability of the current station design to accommodate such missions, in the interests of satisfying the needs of as many users as possible, it is necessary to explore other options.

Table 1 shows a matrix of options that have, in some form or other, been considered in attempts to determine the most appropriate scheme for developing a usable LEO infrastructure. They represent various combinations of locations in LEO for potentially conflicting activities. The following definitions describe

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TABLE 1. Low Earth orbit infrastructure options.

Option	Location in LEO: Vehicle Support Facilities	Propellant Storage	Vehicle Support Crew	Sensitive Users
Station based	Station	Station	Station	Station
Station based w/PTF	Station	PTF	Station	Station
Mantended transportation depot	Depot	Depot	Station	Station
Mantended depot w/PTF	Depot	PTF	Station	Station
Manned transportation depot	Depot	Depot	Depot	Station
Manned depot w/PTF	Depot	PTF	Depot	Station
Science emphasis	Station	Station	Station	Science Platform

the nomenclature used for these activities and the various facilities on which they might be located.

1. Vehicle support facilities are the hangar, tools, robotics, etc. needed to assemble, refurbish, and check out vehicles.

2. Propellant storage refers to facilities such as tanks, pumps, utilities, and robotics support for storing and handling propellant.

3. The vehicle support crew is the crew needed to assemble, refurbish, and check out Mars and lunar vehicles.

4. Sensitive users are experiments or processes that would be greatly affected by field-of-view blockage or by disturbances to the microgravity environment.

5. A propellant tank farm (PTF) is a coorbiting facility for storage and transfer of propellant.

6. "Mantended" means that a crew transfers to a facility for a given work shift and then returns to permanent quarters on a different facility.

7. A transportation depot is a coorbiting facility designed specifically to meet the needs of vehicle preparation and maintenance.

8. A science platform is a coorbiting facility designed to meet the field of view and microgravity requirements that cannot be met in the vicinity of vehicle support activities.

Each option shown in Table 1 has advantages and disadvantages associated with how well contrasting requirements are met. The first option is, in a sense, the default condition, where all activities are kept on the station. As discussed, this scenario is feasible but has adverse impacts on sensitive users. The second option has the advantage of separating crew and instruments from the potential danger and contamination of propellant, but does not alleviate blockage of field of view or disturbances to the microgravity environment.

The third option shown in Table 1 features all vehicle and propellant operations relocated away from the station onto a transportation depot. Options 4 through 6 extend this idea further by separating propellant and permanently manning the depot. These options have the advantage of maintaining a quiescent environment at the station, but disadvantages exist as well. Clearly, if the deposit is mantended, a scheme must be developed to transfer crew to and from the facility on a routine basis, involving added risk and potential loss of usable work time. For the permanently manned case, the potential risk to crew from propellant mishandling is the same as for the first option described, but moving propellant off the depot adds the complexity and expense of a third facility.

The last option shown on Table 1 brings the discussion full circle to maintaining the station as a base for vehicle operations while moving sensitive users to a separate facility. The obvious question here is, if science users were relocated, would it make sense to retain the dual-keel station configuration for vehicle support? The answer is almost certainly no. Rather, it would make sense to redesign the station for the specific purpose of supporting lunar and Mars missions, i.e., develop a transportation depot.

The above discussion is not meant to show that any one of the options is the definitive answer to optimizing the LEO infrastructure for the next century. It is meant, rather to establish the concept of a transportation depot as a viable means of supporting lunar and Mars missions, while maintaining a suitable environment for users with more stringent requirements. The remainder of this paper focuses on the development and analysis of various transportation depot designs as represented by the third option shown in Table 1. This option, the man-tended transportation depot, can be upgraded to a permanently manned configuration, and many of its critical features remain essentially the same whether or not propellant tanks are attached. Thus, it represents a good basis for study.

APPROACH

The work described in this paper proceeded along two lines. First, a list of high-level design requirements was established, and three depot concepts were developed and evaluated against those requirements. Second, a quantitative analysis was performed that determined mass properties and flight mode attitudes for each concept. In this way the feasibility of each concept was evaluated and, more importantly, some generalizations were made about how to improve future designs.

HIGH-LEVEL DESIGN REQUIREMENTS

Design requirements were established from which the three depot concepts were developed. First, it was found necessary that the design provide (1) volume to accommodate vehicles and support equipment; (2) docking facilities to accommodate the OMV and shuttle; and (3) a pressurized command center for controlling/watching EVA and robotic activities. The depot should have expansion capability and provide room for propellant tanks and support equipment. Also, the robotic and EVA should have access to the vehicle and propellant tanks. The facility must provide for simple vehicle separation. The vehicle should separate from the depot along the velocity vector or negative radius vector, and there must be room to avoid any collisions. Orbital decay parameters must not interfere with separation. The vehicle, EVA crew, and propellant must be protected from micrometeoroid impact and as much volume as possible must be enclosed to provide for containment of debris. The EVA crew and propellant

must have thermal protection, and solar dynamic power, GN&C, C&T, and RCS systems must be provided. Finally, the design must assure controllability of all phases of vehicle assembly while minimizing control system size and complexity.

TRANSPORTATION DEPOT CONCEPTS

Three concepts for a man-tended transportation depot were developed based on the design features listed above. An attempt was made to develop concepts that differed in their overall approach, yet still incorporated the desirable features. For example, to assure sufficient access to the vehicle throughout all stages of assembly, a good deal of surrounding truss structure was included in each concept; the differences lay in how much of the vehicle can be enclosed, and how easily robotic arms can get to the center of the vehicle while attached to the structure.

A word should also be said about the rationale behind the total size of the depot structures and the sizing of the propellant tanks that are attached. As will be seen, each concept was made large enough to accommodate a fully assembled piloted Mars vehicle stack as described in *Cirillo et al.* (1988). This meant that approximately 45,000 cm of volume was provided just for the vehicle. A typical lunar vehicle stack is somewhat smaller and so would be accommodated as well. The propellant most often proposed for lunar and Mars vehicles is a mixture of liquid oxygen (LOX) and liquid hydrogen (LH_2). It was decided that the depot should accommodate the maximum amount of LOX/LH_2 needed for the Mars Sprint mission described in *Cirillo et al.* (1988), and so nearly 800,000 kg of propellant can be stored on the depot concepts developed here. This is, of course, more than adequate for lunar mission support, since each lunar sortie requires only about 91,000 kg (200,000 lbm) of propellant. However, since lunar sorties occur six or seven times a year in most scenarios, the amount of propellant needed for multiple sorties or possible rescue missions would likely be two or three times that needed for a single sortie.

Liquid oxygen/liquid hydrogen propellant is generally proposed for lunar and Mars vehicles because of its relatively high specific impulse and because it is hoped that by mining O on the lunar surface, the overall cost of propulsion can be reduced. The great disadvantage is that storing such massive amounts of propellant on the depot applies significant demands on its control system. For example, every time a lunar or Mars vehicle leaves from the depot, a change in the total mass of the system of hundreds of thousands of kilograms results. In order to minimize the complexity of the control system, then, it is desirable to minimize the mass property changes by keeping the propellant distributed around the structure and loading it into the vehicle in a way that keeps the location of the center of mass (CM) relatively constant. Thus, all three depot concepts have three LOX tanks and seven LH_2 tanks distributed in various ways around the truss structure, rather than a single large tank for each. Also, one tank was included on each concept for the hydrazine propellant used by the orbital maneuvering vehicle (OMV). Suffice it to say, then, that while LOX/LH_2 propellant can be accommodated on the depot, development of alternate (i.e., less massive) propellants would help reduce the complexity of the control system and provide greater flexibility in design.

The mass balance of the depot is also affected by the configuration of the lunar or Mars vehicle. Since nearly all scenarios show multistage vehicles that leave LEO together but return at different times, the implication is that at different times during the mission there may be any number of pieces of the complete vehicle stack attached within the depot. This raises the question of how best to attach the vehicles within the depot structure to maintain a stable connection, provide adequate EVA and robotic access, maintain mass balance, and yet not hinder the eventual egress of the vehicle. This is particularly important for lunar vehicles, because the frequency of their arrival and departure makes simplicity and flexibility essential. Two schemes have been proposed. First, a series of deployable/retractable truss structures could be developed and distributed along the length of the vehicle to provide attachment points, as well as a convenient scaffolding for EVA and robotic access. Second, stiff cable could extend between the vehicle and the surrounding truss to provide stability and still allow simple movement of vehicles into, out of, and within the enclosed volume.

Finally, it should be noted that wherever possible, current space station hardware designs were used for the depot concepts; the truss bays are 5 m square, the solar dynamic power, alpha joints, and RCS systems are the same as those found on the station, and the command center and docking ports were derived directly from station nodes and modules. This was meant not only to explore the flexibility of those designs, but also to point out that the experience gained building the station is directly applicable to assembly and maintenance of the depot.

The following sections give brief descriptions of the three depot concepts developed in this study.

The Open-Box Concept

The open-box concept shown in Fig. 1 is the first of the three concepts developed. It features truss sections arranged into a rectangular box that completely encloses the vehicle during all stages of assembly. The box is open on the front, rear, and top faces, but blocked by a cross piece on each side and bottom. Robotic access to the vehicle is via the cross pieces, while the vehicle and associated hardware enter or leave via the front, rear, or top. It is probable that the entire box would be enclosed with impact and thermal protection and debris-containing material that would be drawn back to provide space for vehicle egress. The command center is placed at the top of the box overlooking the vehicle, and the attached docking port and airlock extend out into the flight path. This placement allows adequate viewing of the vehicle, with room for docking the shuttle or OMV and, at the same time, separates the crew from the propellant.

The open box is 12 truss bays long, 9 bays high, and 9 bays wide. Since each truss bay is 5 m square, the outside dimensions of the open box are $60 \times 45 \times 45$ m, the inside dimensions are $50 \times 35 \times 35$ m and the total inside volume 61,250 cu m (2,163,000 cu ft).

Figure 1 shows a standard right-handed body axis system where the positive X axis extends nominally in the direction of the velocity vector, and the positive Z axis extends toward Earth. For convenience in modeling, the origin was placed at approximately the centroid of the starboard face. The concepts described and shown in the next sections have different origin locations but, of course, maintain the same right-handed orientation.

The Prism Concept

Figure 2 shows the prism concept, which, like the open box, features truss sections that completely enclose the vehicle and can be covered with thermal and impact protection or debris

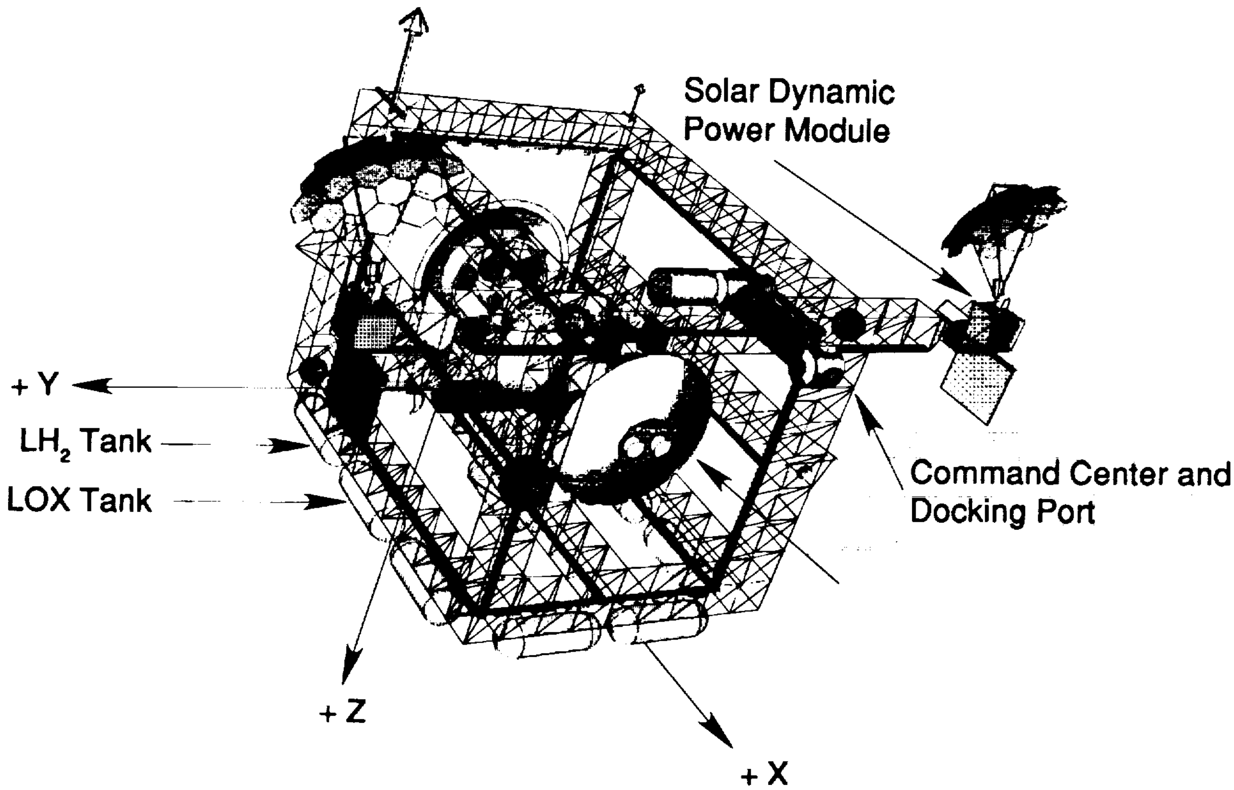

Fig. 1. The open-box concept.

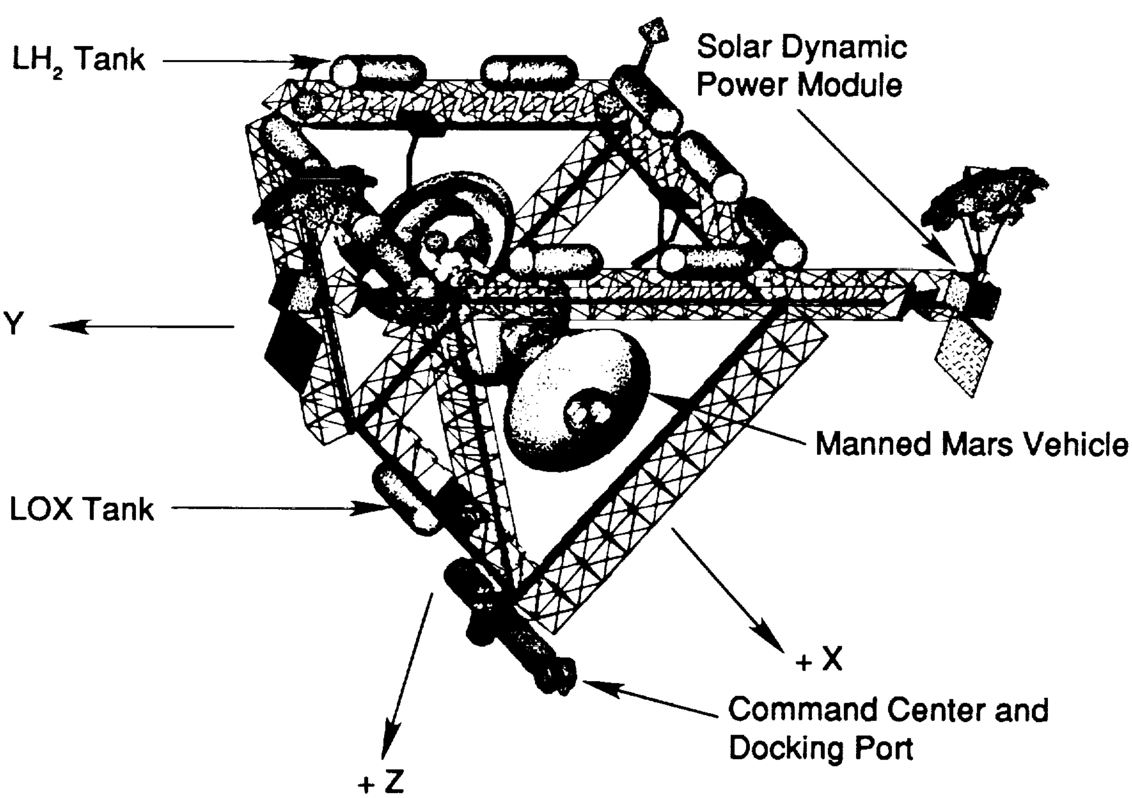

Fig. 2. The prism concept.

containment material. In this concept, however, the vehicle is enclosed by an equilateral triangular prism rather than a rectangular box. The advantages of this design are that the command center is placed with a slightly better view of the vehicle, and the triangular structure allows hardware to enter or leave through all five faces. A disadvantage is that the robotic arms have farther to reach to the center of the structure than on the open-box concept.

The triangular section of the prism is 11 bays (55 m) on each side, and the structure as a whole is 11 bays long. Thus, its inside volume is approximately 59,000 cu m (2,080,000 cu ft), slightly less than that of the open box. Another important feature is that the propellant tanks are distributed around the top of the prism to keep propellant away from the command center, except that to maintain mass balance, one LOX tank was located on the lower apex. As shown on the figure, the origin of the body fixed-coordinate system is in the center of the volume.

Readers who are familiar with the history of NASA's space station program will immediately recognize the similarity between the prism concept and an earlier proposed station configuration called the Delta (*Woodcock* 1986). Early in the program it was thought that the control problems associated with structural flexibility were greater than those associated with rigid body dynamics and the Delta was developed to provide a very stiff platform to reduce the flexible body demands on the control system. However, later studies showed that by increasing the size of the truss bays, the overall flexibility could be reduced to the point that control of rigid body momentum buildup was the dominant issue. This, among other things, eventually led to the adoption of the dual keel as the baseline station configuration.

There are, fortunately, some significant differences between the prism and the Delta, so it should not be thought that the same configuration is being recycled. Most notably, the Delta was to fly "inertially" (always oriented the same with respect to the sun), while the prism, like all the concepts described here, flies "Earth pointing" (with the same apex always pointing toward the Earth). This orientation is possible because the prism uses solar dynamic collectors for power generation, while the Delta used photovoltaic (PV) arrays attached to one side of the structure. Since the Delta was thus forced to fly with the PV side always facing the sun, it was also forced to deal with a constantly changing orientation with respect to Earth, and the corresponding changes in the aerodynamic and gravity gradient forces made it difficult to control.

The Open-Platform Concept

The open-platform concept shown on Fig. 3 was derived somewhat from the dual-keel space station configuration, though obviously the inner transverse boom was removed and the keels were rearranged to provide access to the vehicle. In this concept the major advantage is that the command center and docking port

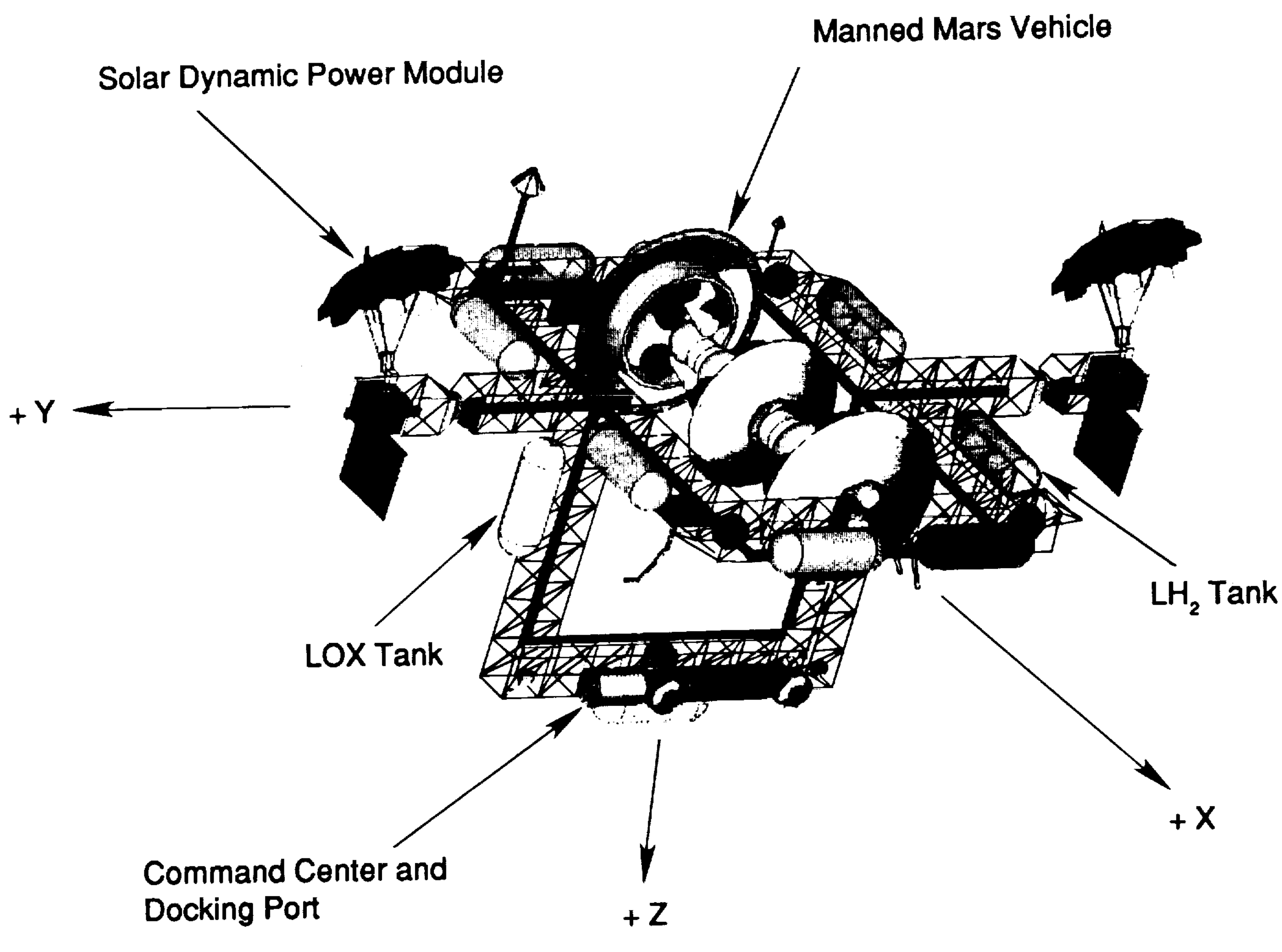

Fig. 3. The open-platform concept.

are placed such that maximum visual access to the vehicle is obtained. Disadvantages are that robotic access is reduced, and the vehicle cannot be completely enclosed for thermal and impact protection or debris containment.

The rectangle, or platform, that surrounds the vehicle is 12 bays long (60 m) and 9 bays wide (45 m). The lower keels, which connect the platform with the lower boom, are 8 bays long (40 m). These dimensions provide adequate room for both the Mars vehicle shown and the lunar vehicles that are generally proposed. The origin of the body fixed-coordinate system is shown in the center of the platform.

The velocity vector of the open platform is such that for the shuttle or OMV to dock at the docking port it must travel under the lunar or Mars vehicle and the surrounding structure. This is certainly possible, but for safety reasons it is not the most desirable scheme. Also, for balancing reasons, the open-platform concept has one LOX tank on the lower boom near the command center.

Table 2 lists the estimated mass of each component of the three depot concepts. The only difference in the total mass of each concept lies in their amount of truss structure and utility trays. As stated above, all components are copies or derivatives of current space station components, except in the case of the propellant tanks, where the masses were derived from previous studies of cryogenic propellant storage. Table 3 shows mass properties derived from the analytical models of the three concepts. The total mass of each concept shown in Table 2 corresponds approximately to the second configuration of each concept in Table 3, which includes tanks but no propellant.

TABLE 2. Component mass summary.

Component Name	Component Mass (kg)		
	Open Box	Prism	Open Platform
Airlock	2,014	2,014	2,014
Alpha joints	1,200	1,200	1,200
CMGs	1,567	1,567	1,567
Cupola	1,455	1,455	1,455
Docking adapters	1,000	1,000	1,000
Nodes (2)	9,091	9,091	9,091
Command center	31,523	31,523	31,523
MSC/transporter	4,909	4,909	4,909
RCS clusters	1,025	1,025	1,025
RCS propellant and tanks	6,364	6,364	6,364
SD power modules (2)	14,078	14,078	14,078
TDRSS and antenna	586	586	586
Tele. servicer (2)	2,381	2,381	2,381
11 propellant tanks	68,924	68,924	68,924
Attached hardware	12,980	12,980	12,980
2 depot radiators	3,670	3,670	3,670
Logistics	8,285	8,285	8,285
Truss	9,875	7,163	4,799
Utility trays	18,008	13,062	8,750
Total	198,900	191,300	184,600

ANALYSIS OF TRANSPORTATION DEPOT CONCEPTS

The following sections give the details of the analyses that were performed on the three transportation depot concepts described above. The first section describes a high-level systems analysis that evaluated each concept against the requirements that were established previously. The second describes the determination of the flight attitude and corresponding momentum buildup for each concept.

Systems Analysis

Each of the concepts described above was developed based on the established design requirements, but as the discussion shows, they do not perform equally well in all areas. Table 4 and the discussion that follows show how each of the concepts has been evaluated with respect to those requirements. The list of requirements that appears above was rearranged to produce nine important areas for the evaluation of the overall effectiveness of each concept. Each area was weighted according to its importance, and numerical rankings were given for each concept. The ranking received by each concept in each area was multiplied by that area's weighting factor to produce a score. Finally, the scores for each concept in each area were totaled to produce a numerical comparison of the effectiveness of each concept.

Naturally, the determination of the weighting factors and rankings was a highly subjective process. However, by outlining the rationale behind each, it is hoped that the total scores can be accepted as a valid comparison and, more important, provide a basis for development of more refined concepts. As more knowledge is gained about the characteristics of this type of structure, more precise evaluation techniques can be applied. In that sense, then, Table 4 provides a concise form for expressing the advantages and disadvantages of each of the concepts developed here.

Effectiveness of docking port/command center.

The placement of the command center and docking port is vital to the overall effectiveness of the depot, but because of the potential for telerobotic technology and the general flexibility of EVA, a less than optimum location can be tolerated. Thus, this area was given a weight of 4. The location of the command center on the open box was given a ranking of 2, since it may be blocked somewhat by the vehicle aeroshells. The prism and open platform command centers are in better locations, but as previously mentioned, docking to the open platform may entail some difficulty from passing below the vehicle.

Capability for expansion.

Capability for expansion is important for any space-based facility, but particularly one that hopes to provide a base for expanded human exploration. It is hoped, however, that the initial capabilities of the depot design will be sufficient for meeting the needs of lunar and Mars mission support. Thus, this area was rated an importance of 3. Each of the depot concepts has room for expansion, but the open box is capable of expanding in every direction. The prism can expand in each direction as well, but the added volume may be less useful due to the skewed nature of the triangular shape. The open platform could easily be modified into a structure resembling a box and then have greater capability for expansion.

Access to vehicle.

Without access to all parts of the vehicle during all phases of its construction and refurbishment, the depot would not be meeting its most basic goal. This area, then, has the highest importance. Access to the vehicle by EVA is essentially the same

TABLE 3. Analytical model mass properties.

	Center of Mass (M)			Inertia (K_g-$M^2 \times 10^7$)					
Configuration (total mass)	X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
Phase II dual-keel space station (267,000 kg)	-3.5	-0.9	3.2	31.0	9.1	25.6	1.76	-1.54	-1.87
Open box (118,000 kg)	-3.1	-19.5	-12.2	8.0	8.8	9.7	0.02	-2.9	-0.08
Open box w/tanks (200,000 kg)	-12.0	-19.7	3.2	16.9	19.2	15.6	0.07	-6.7	-0.18
Open box w/propellant (1,030,000 kg)	-22.5	-20.0	20.7	46.4	39.5	49.7	0.33	-11.3	-0.32
Open box w/vehicle (1,190,000 kg)	-22.7	-20.0	17.9	53.1	49.9	53.9	0.33	-10.9	-0.32
Prism (107,000 kg)	-1.0	0.6	-61.1	22.3	18.6	9.1	0.1	0.78	-0.65
Prism w/tanks (190,000 kg)	-3.7	1.1	-45.1	37.9	31.3	15.2	0.15	-1.28	-0.57
Prism w/propellant (1,020,000 kg)	-7.0	1.6	-26.4	143.2	102.8	70.1	0.8	-13.2	-2.1
Prism w/vehicle (1,180,000 kg	-6.3	1.4	-25.0	145.4	108.9	74.8	0.69	-12.5	-2.4
Open platform (99,900 kg)	0.5	-1.8	20.8	7.9	3.9	6.5	-0.06	0.38	-2.2
Open platform w/tanks (183,000 kg)	-0.6	-0.9	18.5	13.2	7.5	11.5	-0.24	0.26	-5.5
Open platform w/propellant (1,010,000 kg)	-1.7	-0.26	17.6	65.2	36.6	56.4	-0.37	-0.50	-0.12
Open platform w/vehicle (1,180,000 kg)	-1.1	-0.2	15.1	70.2	45.5	60.9	-0.39	-1.6	-0.25

TABLE 4. Analysis of concepts.

		Open Box		Prism		Open Platform	
	Weight	Rank	Score	Rank	Score	Rank	Score
Effectiveness of docking port/command center	4	2	8	3	12	4	16
Capability for expansion	3	5	15	4	12	3	9
Access to vehicle	5	5	25	4	20	3	15
Access to propellant	4	4	16	4	16	3	12
Safety of propellant tank location	5	4	20	3	15	3	15
Impact protection	4	5	20	5	20	3	12
Debris containment	4	5	20	5	20	2	8
Thermal/radiation protection	3	5	15	5	15	3	9
Ease of vehicle separation	5	3	15	3	15	4	20
Total Score		154		145		116	

Weight: 5 = very important; 1 = not important.
Rank: 5 = very good; 1 = poor.
Score = Weight × Rank.

for all concepts. Robots have considerably farther to reach on the prism than on the open box, and they may have difficulty reaching around the vehicle on the open platform.

Access on the prism could be improved considerably by building cross pieces between the front and back faces, similar to those on the open box. This would allow robots to travel much nearer the vehicle, but would close off two of the faces through which hardware can enter or leave the enclosed volume. Also, if the vehicles were attached to the structure by deployable/retractable truss sections, as mentioned previously, robotic access on both the prism and the open platform could be enhanced.

Access to propellant.

Since a great deal of the total vehicle support activity consists of replacing or refilling propellant tanks and maintaining storage and pumping apparatus, routine access to propellant areas is essential. Also, in the case of a spill, access must be simple and rapid to allow repairs to begin as quickly as possible. This area was weighted a 4 because, like the command center location, a less effective propellant handling scheme can be partially overcome by automation and expanded EVA.

All three concepts have adequate robotic access to propellant areas, but because the open box and the prism allow EVA access from inside the enclosed volume, they were ranked slightly higher. Clearly, the thermal, radiation, and impact protection provided by the enclosed volume should be taken advantage of wherever possible.

Safety of propellant tank location.

To maintain mass balance, the propellant tanks must be distributed around the depot structure. However, it is also important to have propellant near the vehicle to aid the fueling process, and because pumping it over great distances might increase the chances of a mishap. Of course, to ensure a safe haven in an emergency, as well as an uncontaminated base for repair operations, it is desirable to maintain as much distance as possible between the tanks and the command center. Since this is both an operations and a safety issue, it was weighted as high as possible. As stated before, the platform and prism have one LOX tank near the command center, and so they were rated slightly lower than the open-box concept.

Impact protection.

The importance of this area was rated at 4 out of 5 because the probability of impact damage increases with surface area, which Mars and lunar vehicles have in abundance, and with mission duration. Thus, over the decades of support for lunar and Mars missions, the potential for impact damage to unprotected vehicles and crew will be considerable. Clearly, the open box and the prism have a great advantage over the open platform in being able to cover the entire vehicle with impact protection.

Debris containment.

One of the many things that humans must learn to do in space is contain the loose material that results from routine operations, such as assembling or repairing spacecraft. This is on the same order of importance for the depot as providing impact protection, since every piece of orbiting debris is potentially damaging to the vehicle and crew. Again, the open-box and prism concepts have the advantage of completely enclosing the entire work and storage area, whereas on the open-platform concept the area can be, at best, only partially enclosed.

Thermal/radiation protection.

Both Mars and lunar mission support will require significant amounts of EVA for assembly and maintenance of vehicles. This makes it essential that external thermal and radiation protection be provided to augment that afforded by the EVA suit. Furthermore, due to the long duration of exposure to space that the vehicles will undergo, reduction of radiation and thermal cycling effects will enhance the reliability of the hardware. As before, the open-box and prism concepts have the advantage of enclosing the vehicle, work, and storage areas.

Ease of vehicle separation.

Separating the vehicle from the depot will be a complicated process regardless of the configuration of the depot. The disentangling of plumbing, utilities, and checkout equipment, the retracting of support structure or cables, and the danger of collision, make simplicity and reliability essential. Furthermore, should a failure happen such that the vehicle and depot were not able to be separated for a significant length of time, the impact on the mission would be severe. Thus, this process has been rated of equal importance with propellant safety and vehicle access.

There is a fundamental conflict between the two concepts that enclose the entire vehicle (open box and prism) and the one that does not (open platform). Enclosing the vehicle carries the advantages of enhanced protection, but at the same time complicates the separation process. In this area, then, the open platform is superior to the other two concepts due to the sparseness of the truss that surrounds the vehicle. It is vital, however, to consider the manner in which the vehicle separates from the depot. With the open-box and prism concepts, the vehicle may exit the enclosed volume through the front, back, or top faces, depending on orbital and plume impingement requirements. With these concepts, therefore, it is only necessary to assure that sufficient margin for clearance is allowed when sizing the vehicle and depot. With the open platform, the vehicle must separate vertically away from the depot, but only half the vehicle needs to clear the surrounding truss. The operational flexibility of the open platform is less than the other two concepts, but the danger of collision is reduced as well, and so it was rated slightly higher.

FLIGHT ATTITUDE ANALYSIS

A necessary step toward showing the viability of a large space structure design such as those described in this paper is to determine whether its configuration is controllable enough to provide a stable environment for on-orbit operations. A complete analysis would develop an attitude control/momentum management scheme for each stage of its assembly, as well as for as many operational configurations as possible, and couple flexible body effects with a control system design. *Heck et al.* (1985), *Woo et al.* (1986), *Robertson and Heck* (1987), and *Sutter et al.* (1987) describe how this level of analysis is being performed for the current space station design. For the purpose of this study, however, such an in-depth analysis would be premature. Instead, it was decided that by determining the key flight attitude characteristics for each transportation depot concept and comparing them to the corresponding values for the space station, the reasonableness of each design could be shown and indications could be made for future refinements. The following sections give a brief discussion of momentum management schemes, show how the depot attitude control requirements differ from those of the station, and give results of the analysis.

Approach and Modeling

An orbiting spacecraft is subject to a variety of environmental and operational effects that disturb its flight attitude. Environmental disturbances include forces due to aerodynamic drag, the difference in gravitational force due to the mass distribution (called gravity gradient forces), and forces due to solar radiation pressure. *NASA* (1969, 1971, 1986) details these effects. Operations such as relocating payloads, berthing and docking, and articulation of solar dynamic collectors, produce disturbances to the attitude of the spacecraft, as well as changes to its physical configuration and mass properties. Of course, changes in the physical characteristics of the spacecraft alter the aerodynamic, radiation pressure, and gravity gradient effects as well. The net result of these disturbances is a buildup of angular momentum that must somehow be dissipated to maintain the desired flight attitude.

If it were possible to instantaneously measure or predict environmental effects and calculate operational effects, it would be posssible to continuously correct the attitude of the spacecraft to maintain a net angular momentum of zero. Much like flying a plane or driving a car, whenever a disturbance was sensed, a corresponding correction would be made to maintain the desired attitude or direction. Unfortunately, due to imprecise knowledge of the aerodynamic and solar environment, and the difficulty with sensing disturbances, such a complete knowledge of the current state of the spacecraft is not achievable. Furthermore, continuous correction of attitude would require either constant use of expendable fuel, or some means to continuously change the aerodynamic and gravity gradient characteristics of the system. Such continuous correction would also be likely to disturb normal operations.

A more practical scheme is to allow some moderate amount of momentum to build up over a period of time and correct the attitude only when operational requirements allow. Unfortunately, if angular momentum is simply allowed to build up, the attitude of the spacecraft will change significantly over a period of time, changing viewing angles and seriously complicating C&T and protection systems. It is necessary, then, to provide a mechanism by which momentum can be stored within the spacecraft without

disturbing its attitude or continuously making corrections. Of course, all the momentum that is stored must eventually be released in some way, and so periodic corrections must be made.

The scheme for maintaining momentum buildup in the current space station design involves the use of control moment gyros (CMGs) as a method of storing momentum. As the station passes through each orbit, a certain amount of angular momentum is built up and countered with torques produced by the CMGs. When the momentum buildup reaches a level near the CMGs torque-producing capacity, the station's reaction control system (RCS) jets are fired in a way that releases the built-up angular momentum and allows the CMGs to return to a lower level of torque. This process is called "desaturation" of the CMGs.

The momentum buildup is divided into two distinct components called "cyclic" and "secular." Cyclic momentum results from environmental forces that grow and then dissipate through an orbit such that the net buildup is approximately zero. This type of momentum buildup is important because, even though the net value is negligible, the peak value is generally so large that it greatly affects the size of momentum storage devices needed to maintain attitude. Secular momentum results from forces that vary in magnitude such that the net resulting momentum is nonzero. Values of secular momentum are generally lower than those for cyclic, but it is the secular component that must be dissipated periodically to avoid exceeding the capacity of the momentum storage system. The capacity of the momentum storage device needed, then, is determined by the maximum value of momentum buildup (the maximum sum of cyclic and secular momentum) experienced in an orbit. *Robertson and Heck* (1987) give a good discussion, as well as numerous examples, of how the secular and cyclic components combine to produce the total spacecraft momentum.

Along with the use of CMGs to store momentum, the space station makes use of the fact that since environmental forces are highly dependent on the flight attitude, it is generally possible to maintain an attitude that reduces their magnitude. Clearly, by maintaining an attitude that minimizes the magnitude of the environmental forces, the CMGs would need to store less angular momentum and be desaturated less often. This minimum torque attitude is expressed as three ordered Euler angles, called torque equilibrium angles (TEAs), which represent successive yaw, pitch, and roll rotations about the body axes. Before the rotations are executed, the body axes correspond to the local vertical local horizontal (LVLH) coordinate axes defined by the positive X axis along the veolocity vector, positive Z toward Earth along the nadir vector, and positive Y in the orbit plane to form a right-handed system.

A major advantage of the transportation depot concept is that by separating the vehicle processing activities from the sensitive users, the depot is released from requirements for astronomical and Earth viewing angles. Thus, while the space station must maintain a pitch angle of $\pm 5°$ for viewing reasons, the depot is constraihed only by communications requirements and operational needs, such as vehicle separation. Also, since the depot need be only a moderately quiescent environment, the constraints on attitude correction are less stringent as well. On the other hand, the mass property changes mentioned several times above make it imperative that a flexible momentum management scheme with sufficient CMG capacity be provided.

The flight attitude analysis consisted of calculating TEAs and corresponding momentum buildups for 4 different configurations of each depot concept (12 configurations in all), where each configuration represents a different operational state. The first configuration was simply the depot without any propellant, tanks, or vehicle on board. Empty tanks were added for the second configuration, propellant was loaded into the tanks for the third, and finally a complete vehicle was included. In this way, it was possible to evaluate the effects of increasing and repositioning the total mass of the three depot concepts.

Solid models of each configuration were developed using the GEOMOD program developed by Structural Dynamics Research Corporation (SDRC), and the geometry and mass properties were then passed to the ARCD program described in *Heck et al.* (1985) and *Robertson and Heck* (1987). The ARCD program computes forces and moments needed to maintain a given attitude and calculates the momentum buildup about each axis in a single orbit. The calculations in ARCD include environmental effects as well as the effects due to articulating mechanisms such as solar dynamic collectors. The two software packages, GEOMOD and ARCD, are integrated under a single operating environment at NASA Langley called IDEAS[2] (*SDRC*, 1985), which also includes other SDRC- and NASA-developed software for structural, thermal, and controls analysis.

Flight Attitude Analysis Results

Table 5 contains the basic results of the flight attitude analysis of the depot concepts in each of the four configurations described above. The first three entries in Table 5 are the attitude angles for each configuration that minimize the amount of momentum built up in an orbit. These are the TEAs described above, where positive values for the angles ϕ, θ, and ψ represent positive rotations about the X, Y, and Z axes, respectively. Thus, if one were looking along the flight path, a positive ϕ would be characterized by a clockwise roll, θ by an upward pitch, and ψ by a left-to-right yaw.

The fourth entry in Table 5 is the resultant magnitude of the X, Y, and Z axis secular momentum buildups that correspond to the given TEAs. The fifth is the value of momentum buildup for the given attitude, which represents the greatest resultant of the cyclic and secular components in a single orbit. These two values are given because it is the secular component that needs to be dissipated periodically, while the maximum value provides a good measure of the size of momentum storage device needed. In a broader sense, then, the maximum value expresses the overall difficulty of maintaining the given attitude and thus is a convenient means for comparing concepts.

Before discussing the results in detail, a few things should be mentioned. First, the dominant momentum value generally occurs about the pitch axis, but in some cases the value about the roll or yaw axis is larger. This is why the resultant magnitude is given, with the implication that it is necessary to provide significant momentum storage and dissipation along all three axes. Second, these values are very sensitive to small changes in attitude. For example, the maximum momentum buildup for the open box and the prism increase by two orders of magnitude with only a 5° change in θ. This implies that to provide margin for maneuvering and reboosting, a significantly larger storage device would be needed. The open platform is slightly less sensitive than the open box and prism, but still would require a greater capacity than indicated by the maximum value shown. Also shown in Table 5 are results for the dual-keel space station design. These values are subject to the same argument, but since the station will not experience the same degree of mass property changes, it will not need to change its attitude as drastically or as often as the depot.

TABLE 5. Flight attitude characteristics.

Configuration	Torque Equilibrium Angles (Degrees) ϕ (X)	θ (Y)	ψ (Z)	Momentum Buildup (Nt-M-Sec) Secular	Maximum
Phase II dual-keel space station	0.2	3.0	0.6	1,500	3,050
Open box	1.3	-36.6	-2.6	467	5,800
Open box w/tanks	-1.45	-48.1	-2.0	674	4,740
Open box w/propellant	0.1	-41.4	-9.2	2,442	8,400
Open box w/vehicle	0.0	-44.4	-4.4	2,335	3,270
Prism	-4.0	-6.4	-4.0	3,441	10,870
Prism w/tanks	-2.1	2.2	0.9	1,841	6,130
Prism w/propellant	-3.8	10.1	0.1	1,440	3,360
Prism w/vehicle	-3.9	9.9	0.0	1,075	2,900
Open platform	5.0	-0.4	-1.0	1,750	6,470
Open platform w/tanks	4.9	1.8	-0.9	2,019	9,860
Open platform w/propellant	0.3	4.0	-0.7	442	2,880
Open platform w/vehicle	0.6	10.9	-0.7	616	5,380

Finally, it is interesting to note what the momentum buildup would be if the depot were to fly at a zero attitude (ϕ, θ, and ψ equal to zero). For the open box without tanks, propellant, or vehicle, the maximum value is 588,800 Nt-M-sec. The prism in the same configuration would have a maximum value of 313,800 Nt-M-sec, while the open platform would build up 128,400 Nt-M-sec. Clearly, a great deal is gained by flying at a mimimum torque attitude, but the precision required to maintain it, and the penalty for straying away from it must also be considered.

With the above in mind, it is immediately clear from Table 5 that the prism and the open platform exhibit much more favorable TEAs than the open box, while the momentum values are comparable. It would be desirable in future studies to investigate the performance of the box with impact or thermal material covering various faces. Blocking off the top or bottom, for instance, would likely alter the aerodynamic drag profiles in such a way as to reduce the pitch TEA.

Another interesting result that does not appear on Table 5 is that, because of its symmetry, the box has a minimum torque attitude at a high positive pitch angle. For example, the box without tanks, propellant, or vehicle can fly at a pitch angle of 52.3° with a maximum momentum value of 6500 Nt-M-sec. This, again, is because by not covering the top or sides, minimum frontal area of the configuration exists when a front edge blocks one of the rear edges. Thus, the minimum aerodynamic drag profile exists when the box is tilted so that a front-to-back diagonal aligns approximately with the velocity vector.

The prism and open platform are less symmetrical and so do not exhibit an alternate minimum torque attitude. It is important to note as well that the prism does not experience the same control difficulties as the Delta space station configuration described previously. In fact, the momentum requirements of the prism are not significantly more demanding than those of the other depot concepts or the dual-keel station, and it also maintains moderate TEAs.

The above analysis of depot concepts increases the feeling that this type of facility is feasible. By showing that the controllability requirements of a few very different configurations are comparable to those of the current station design, some confidence is gained that future designs will avoid significant difficulty.

RECOMMENDED FURTHER ANALYSIS

Throughout the high-level analysis described in this paper, several studies were identified by which the transportation depot concepts can be refined. The authors hope to continue the development and analysis of concepts to (1) verify controllability of the depot during various stages of vehicle assembly and for various amounts of stored propellant; (2) determine reactions to dynamic disturbances (modal excitation, structural loads, control systems interactions, etc.); (3) develop depot assembly timelines; (4) manifest depot hardware on ELV or shuttle; (5) determine resource requirements (power, logistics, etc.); and (6) develop growth scenarios for depot concepts. It is also hoped that time can be found to investigate the characteristics of other large-scale coorbiting facilities such as a science platform, a propellant tank farm, and an artificial-/variable-gravity facility.

GENERAL OBSERVATIONS

The following discussion summarizes several of the generalizations drawn from the analysis of the transportation depot concepts. As mentioned, not all the desirable design features listed as high-level requirements are compatible, and so it is important to identify conflicts and potential solutions wherever possible.

Large Enclosed Volumes

By enclosing the vehicle work areas, it is possible to provide better thermal, radiation, and impact protection, better debris containment, and generally better robotic and EVA access. However, it is important not to block viewing from the command center or somehow cut off the ability to enter or leave the volume in an emergency. Also, surrounding the vehicle with a great deal of truss or equipment makes for more complex separation of the vehicle from the depot.

Command Center Location

It is important that the command center be near the middle of the vehicle or high over one end to avoid blockage of view by the vehicle aeroshells. Also, by locating the command center

inside an enclosed volume or very near a partially enclosed volume, it is possible to take advantage of extra thermal, radiation, and impact protection. In fact, the two concepts with large enclosable volumes, the open box and the prism, could both have the command center moved inside the volume while still attached to the docking port.

Propellant Tank Location

For both lunar base and Mars mission support, the largest mass on orbit at any given time is propellant. Because the location of the propellant mass greatly affects the depot's flight attitude (TEAs) and creates potential problems for the control system, it is vital that proper distribution of tanks be maintained. Furthermore, distributing tanks around the structure would reduce the effect of a localized spill, since only limited amounts of propellant would be released. However, this may require that propellant be pumped over great distances, thereby increasing the complexity of the pumping system and the probability that a spill would occur.

On-orbit Assembly of Depot

Assembly and integration experience gained on the space station is directly applicable to the assembly of the depot. In particular, techniques and timelines for the assembly of truss, integration of power modules and utilities, GN&C and C&T systems, robotic support, EVA/IVA procedures, maintenance and failure prediction, and the transfer and mating of payloads all will have been dealt with in detail during the assembly and test phase of the space station. Finally, assembly and checkout of the depot has a significant advantage in that the crew needed for the effort can be based on-station and transferred to the assembly area via OMVs.

CONCLUDING REMARKS

The transportation depot and other similar coorbiting facilities clearly represent the second generation of space stations and, as such, assume successful completion of currently planned facilities. In particular, the space station, the shuttle, heavy lift launch systems, and OMVs all play a part in the overall infrastructure needed to provide proper support for expanded human presence in space. The above discussions have established the viability of the transportation depot concept, but whether this type of facility is ever built is another matter altogether.

Ultimately, the decision to locate vehicle processing activities on a coorbiting facility rather than on the station requires a prioritization of national goals and, most likely, a compromise between conflicting requirements for science and exploration. The resolution of such conflicts is obviously well beyond the scope of this paper. However, it is clear that without a broad-based commitment from government, industry, and the public, ambitious projects such as lunar and planetary bases, and the facilities required for their success, can never become a reality. It is hoped that as NASA's Space Station Program and Office of Exploration proceed, such long-term goals continue to be articulated as being vital for the continued growth of the national space program.

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REFERENCES

Cirillo W. M., Ayers J. K., Kaszubowski M. J., Llewellyn C. P., Merideth B., and Weidman D. J. (1988) *Manned Mars Mission Accommodation—Sprint Mission.* NASA TM-100598.

Heck M. L., DeRyder L. J., and Orr L. H. (1985) *Articulated Space Station Controllability Assessment.* AIAA Paper 85-0024.

NASA (1969) *Spacecraft Gravitational Torques.* NASA SP-8024.

NASA (1971) *Spacecraft Aerodynamic Torques.* NASA SP-8058.

NASA (1986) *Natural Environment Definition for Design.* NASA, JSC30000, Appendix 3.1, Version 3.0.

Robertson B. P. and Heck M. L. (1987) *Spacecraft Attitude Control Momentum Requirements Analysis.* NASA CR-278219.

SDRC (1985) *IDEAS² User's Manual.* Structural Dynamics Research Corp., Cincinnati.

Sutter T. R., Cooper P. A., Young J. W., and McCutchen D. K. (1987) *Dynamic and Attitude Control Characteristics of an International Space Station.* AIAA Dynamics Specialist Conference, AIAA Paper 78-0931-CP.

Weidman D. J., Cirillo W. M., Llewellyn C. P., Kaszubowski M. J., and Kienlen E. M. (1987) *Space Station Accommodations for Lunar Base Elements—A Study.* NASA TM-100501.

Woo H. H., Morgan H. D., and Falangas E. T. (1986) *Momentum Management Concepts for a Space Station.* AIAA Paper 86-2047.

Woodcock G. R. (1986) *Space Stations and Platforms.* Orbit, New York. 232 pp.

DEVELOPING A SAFE ON-ORBIT CRYOGENIC DEPOT

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New U.S. space initiatives will require innovative technology to realize planned programs such as piloted lunar and Mars missions. Key to the optimal execution of such missions are high performance orbit transfer vehicles and propellant storage facilities. Large amounts of liquid hydrogen and oxygen demand a uniquely designed on-orbit cryogenic propellant depot. Because of the inherent dangers in propellant storage and handling, a comprehensive system safety program must be established. This paper shows how the myriad and complex hazards demonstrate the need for an integrated safety effort to be applied from program conception through operational use. Even though the cryogenic depot is still in the conceptual stage, many of the hazards have been identified, including fatigue due to heavy thermal loading from environmental and operating temperature extremes, micrometeoroid and/or depot ancillary equipment impact (this is an important problem due to the large surface area needed to house the large quantities of propellant), docking and maintenance hazards, and hazards associated with extended extravehicular activity. Various safety analysis techniques were presented for each program phase. Specific system safety implementation steps were also listed. Enhanced risk assessment was demonstrated through the incorporation of these methods.

DEVELOPING A SAFE ON-ORBIT CRYOGENIC DEPOT

The National Aeronautics and Space Administration (NASA) currently has new U.S. space initiatives to develop piloted lunar and Mars missions. Central to these programs are orbital transfer vehicles (OTVs) and extensive cryogenic propellant storage facilities operating in Earth's orbit. It is known from *Stubbs et al.* (1987) that large quantities of cryogens, such as liquid hydrogen and oxygen (on the order of 200,000 lb for geosynchronous Earth traffic and 400,000 lb for lunar traffic), require the advancement of on-orbit cryogenic propellant storage technology.

A NASA On-Orbit Cryogenic Depot Technology Task Force is presently studying the various concepts. The definition stage is sufficiently prefatorial that the exact purpose of the depot has yet to be defined. Its primary function is to fuel OTVs. However, secondary functions and modes of completing the primary function have not been determined. Primary functions of the depot will include propellant storage, acquisition, expulsion, conditioning, refill, measurement and control, thermal control, venting, data/communication, inspection and diagnostics, and vehicle proximity operations. All these functions will demand unique systems creation. Technology development requirements for the depot must be identified and solved prior to full-scale deployment.

Several options have been proposed for the propellant depot and maintenance facility. These include (1) a single OTV maintenance facility with refueling capability attached to the space station, (2) a space-station-attached maintenance facility and separate co-orbiting cryogenic propellant depot, and (3) a co-orbiting OTV maintenance and propellant storage platform. The task force is currently strongly pursuing the second option, the attached servicing facility with co-orbiting propellant depot.

Now is an opportune time to seriously develop a safe OTV cryogenic depot. The inherent hazards of the above-mentioned options are considerable. Even though the task force is studying the second option more seriously than the others, a comprehensive system safety effort must be expanded in tandem with technology and trade studies.

Failures of the cryogenic propellant depot would not only affect the facility operators, but may possibly damage the space station. Loss of the depot would severely affect the mission and could cancel the program. If all the fuel were lost at a critical path point, new launch windows (for refueling both the depot and planetary spacecraft) would have to be established. Questions regarding man-tended (or partially man-tended) vs. automated operations must be addressed. The proximity of the depot to the space station is of critical importance. The magnitude of an explosion of 200,000 lb of liquid hydrogen and oxygen could directly affect the space station.

There are numerous depot configuration trades that must be analyzed. One such issue that has been suggested is the utilization of tethers to facilitate and simplify propellant transfers. However, the safety implications are profound; the less automated the system, the greater the human risk. Engineering optimization is fundamental to realizing an efficient and cost-effective system. Another important issue is growth potential, which is key to expanding NASA's dynamic mission capabilities. This well illustrates the need for continual system safety analysis. Any small change in this complex system could negatively affect the system.

Safety trade-offs for efficacious operations will not enhance the overall and continual use of the depot. Because hazards may not be readily apparent, an ongoing safety effort is needed to bring problem areas to light. The results of a serious risk assessment will positively contribute to a viable technology trade-off decision. The purpose of this paper is to show how and where system safety can be applied to develop a safe cryogenic depot.

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SYSTEMS LIFE CYCLE OF THE CRYOGENIC DEPOT

The essential factor in governing a congruous safety effort is to be intimately involved in the entire cryogenic depot program. The way for system safety to become an integral member, from conception through deployment, is to participate as a working member (with equal status and voice) in the following program phases: (1) concept; (2) definition (flight experiment definition and analytical models development); (3) development (pathfinder, technology demonstrator design and testing, prototype hardware design and testing, and final development); (4) production; (5) deployment (space transport system (STS) use and depot amplification) (Fig. 1).

INTEGRATED SYSTEM SAFETY FOR THE CRYOGENIC PROPELLANT DEPOT

Cryogenic Depot Technology Requirements

Various planned programs impel the cryogenic depot development. The significant mission drivers are manned Mars, manned lunar, robotics exploration, and planet Earth (*Ride,* 1987). These drivers have various technology requirements.

Technology requirements are abundant and fall into general categories of fluid storage, supply, handling, and transfer; advanced instrumentation; and materials and structures. A partial listing of technology requirements identified to date includes cryogenic fluid resupply; reusable Earth-to-orbit cryogen transport; long-term orbital cryogen storage; control, instrumentation, and diagnostics; fluid thermodynamic analytical models (chilldown, vapor liquefication, vent characterization, etc.); pressure control techniques for long-term storage; zero-gravity fluid quantity gauging; mass measurement accuracy (expulsion and refill); quick disconnect; fluid leak operations/detection; fluid venting/dumping; thermodynamic vent; refrigeration requirements; fluid motion effects on controls; pretransfer conditioning of receiver vessel (chilldown, ventdown, purge, etc.); nonvented receiver refill; transfer line conditioning; storage loss reduction; and material development (*Stubbs et al.,* 1987).

Many of these technology development requirements are high risk, both in terms of technology payoff and system safety significance. Many must be demonstrated in orbit since the technologies and analytical models cannot be validated in Earth's gravity. Each one of the requirements has a potent system safety implication. Only through a well-defined and integrated system safety effort can the issues be appropriately understood.

Technology Development

To safely develop the appropriate technology, program considerations should be analyzed. NASA has identified three main technology considerations that must be addressed: mission, manufacturing, and performance (*Davis et al.,* 1970). System safety studies should be conducted for all of them. System concerns are discussed below.

Safety Task	Concept	Definition	Development	Production	Deployment
1. Develop system safety plan	Initial	Final	---------------	---------------	---------------
2. Conduct hazard analysis	PHA	PHA/FTA	FMEA/OSHA/FTA	OSHA	Update hazard analyses
3. Define safety design requirements	Initial	Final	Final	---------------	---------------
4. Conduct failure analysis	Historical data review	Historical data review	Test results	Update database	Update database
5. Conduct risk analysis	Preliminary risk assessment	Update risk assessment as new material becomes available	Update risk assessment as new material becomes available	Update risk assessment as new material becomes available	Update risk assessment as new material becomes available
6. Conduct safety test	---------------	Flight definition tests and path finder tests	Pathfinder/technology demonstrator and prototype hardware	Hardware qualification tests	---------------
7. Conduct safety training	---------------	---------------	Support	Monitor	Monitor

Fig. 1. System safety tasks. Safety tests should be both ground- and on-orbit tests. PHA: Preliminary Hazard Analysis; FTA: Fault Tree Analysis; FMEA: Failure Mode and Effect Analysis; OSHA: Operating and Support Hazard Analysis. From *Roland and Moriarty* (1983).

Mission considerations.

Operational pressure: Cryogens may be stored as single phase (supercritical) or two phase (subscritical). It appears that NASA is supporting the subcritical storage system over the supercritical.

Quantity measurement: The accuracy of quantity is paramount for operational use and system diagnostics. Quantity measurement with a subcritical system is much more difficult than with a supercritical system; the liquid-vapor mixture involves a more complex measurement media. NASA is currently developing subcritical cryogen (for anhydrous ammonia) measuring devices for the space station.

Pressure control: A subcritical system may undergo pressure instabilities (including boiloff). If the depot is hard-fixed to the space station, vapor releases may cause small perturbations, affecting space station experiments and possibly polluting the outer skin of the station. Thermal stratification also may affect pressure control.

Manufacturing considerations.

Reproducibility: Manufacturing repeatability and accuracy for system operation is critical for mission success.

Shelf life: The depot has a designated shelf life of 10 to 20 years.

Weight: Launch costs and weight, especially for a 200,000-lb to 400,000-lb fluid, dictate optimal design.

Materials: They must be compatible with the environment (of both deep space and the fluid media itself) and have high strength-to-weight ratios. Some of the major material concerns are fracture toughness, fatigue properties, chemical properties, permeation, creep properties, embrittlement, and joint efficiency.

Envelope constraints: Depot (whether attached, tethered, or completely autonomous to the space station) interfacing mechanisms will influence the physical and structural design parameters.

Performance considerations.

Standby time: The dormant period between use and nonuse is important. Cryogen residue in the line can vaporize and cause a pressure barrier when fuel is extracted in the next run.

Fluid quantity: The quantity cannot be accurately determined until the depot purpose has been more clearly delineated.

Power requirements: Power requirements for pumps, fans, and diagnostics, are contingent on fluid usage requirements.

Environmental conditions: Temperature variations, due to thermal cycling, will significantly affect thermal and thermodynamic design. Micrometeoroid and space debris impact also are important for design.

METHODOLOGY FOR SYSTEM SAFETY

To fully support each phase of the system life cycle, various system safety techniques can be exploited. Most of these methods should be used for every serious technology option. The choice of system safety analysis depends on the program phase and level of developmental detail. Each one of the methods has been successfully proven on numerous NASA, Department of Energy, Department of Defense, and Nuclear Regulatory Commission projects. The most common methods are listed below with a brief description of each safety tool and program phase application. It is not within the scope of this paper to go into in-depth explanations of each analysis technique.

Preliminary Hazard Analysis (PHA)

The PHA is the base-line document for the integrated system safety effort. The word "preliminary" denotes first hazard search of the system. The analysis addresses the major hazards of the system and allows early tracking of problem areas. The initial PHA is not to affect control of the hazard (this will come later in the program life cycle with other techniques), but rather to provide management with knowledge of potential risks for feasibility studies and program definition activities. Tradeoff studies are greatly enhanced with the hazard identification method of the PHA, allowing establishment of design and procedural requirements, to eliminate or control hazardous conditions before the system becomes so advanced that design changes become prohibitive in terms of cost. The PHA is most frequently used during the concept and definition program phases.

Subsystem/System Hazard Analysis (SSHA/SHA)

The SHA (the format and use are exactly the same for the SSHA) is an inductive method of analysis. Interest is focused on system-level design features that may affect overall performance or safety. Special interest is concentrated on interface considerations. System information is then used for the integration of the full system hazard analysis. The analysis is usually conducted during system definition and development phases.

Operating and Support Hazard Analysis (OSHA)

The focus of the OSHA is on system operation. Analysis emphasizes human factors engineering and operating conditions. Areas considered are use of safety guards or devices, special procedures or training, and identification of timing of operations or functions and other ergonomic concerns. The OSHA should be initiated early enough in system development for technical input. However, the technique is very useful in the development phase as an overall safety verification.

Fault Tree Analysis (FTA)

The FTA is a powerful deductive analytical tool. The method employs a Boolean logic model that mimics the relationship between events in a system. The final outcome is called the top event. Even though the method is called fault tree, the top event may be either a desired or undesired outcome. This safety and reliability tool is very useful in the early design phases and in studying operational systems. The output may be of a quantitative or qualitative nature, depending on the input information.

Failure Mode and Effects Analysis (FMEA)

The FMEA is sometimes called a failure mode and effects and criticality analysis (FMECA). Though this is primarily a reliability tool, the analysis does furnish much useful information. The FMEA focuses on single-point and piece-part failures and their propagation effects through the system. The technique tends to concentrate primarily on component failure instead of human error.

Other tools frequently used in system safety are change analysis, human factors analysis, common cause failure analysis, training, audits, and mishap investigations.

Because the PHA, OSHA, and SHA are all very similar, frequently they are grouped into a comprehensive hazard analysis. This

would require that the hazard analysis be updated at program milestones to incorporate operational (and human factors) and system hazards.

Sequence of Hazard Control

It is of importance to remember that risks always exist. The only way to mitigate the hazards is to control them; the use of the sequence of hazard control is the optimal method. The following are important in applying the sequence of hazard control activities: design for acceptable hazard, use of safety devices, use of warning devices, and finally, the use of procedures and training.

Design for an acceptable hazard means to minimize through design methods the hazardous condition. For example, if heat is an added hazard to loss of cryogen control, then one should try to design without the need for external heating. Another example is the use of separate quick disconnects for oxygen and hydrogen cryogens. If incorrect mating is made impossible by separate, incompatible disconnects, then the risk of mixing fluids by incorrect connection is alleviated.

Safety devices are additions to the system to control the hazard. The best example is a pressure relief valve on a cryogen storage vessel. The hazard of tank rupture is always there, but mitigated through the relief system.

Warning devices are used to alert personnel and machinery to impending danger or harm. The purpose of the warning device is to prepare personnel and machinery for an emergency contingency. Gas and leak detection devices are good examples of this.

Procedures and training are the least successful of the sequence elements. Because people are fallible, it is always best to try to control the hazard by hardware design methods. People tend to reach a 50% error rate during highly stressful situations. Even though the operator may be well trained in transferring cryogens to the OTV, the operator is unlikely to perform as well during an emergency situation. The same well-trained operator may also fail during normal operations due to unforeseen stresses such as personal problems, physical distances between operating devices, unusual environmental conditions, tedious tasks, etc.

Risk Assessment Hierarchy

Risk analysis and control are the ultimate goals of system safety. Various techniques, applied during different phases of the life cycle, will achieve that goal. However, to adequately assess and manage the risks, hazard severity and probability of occurrence must be studied. Each of the analyses allows for hazard severity and probability identification. When the hazard is identified, a severity and probability is assigned. This permits one to classify the hazard. A matrix gives an overall risk assessment code. The decision maker now has something tangible to review for trade-off studies or system changes.

It is best to try to be quantitative whenever possible. However, inaccurate or ambiguous numbers can lead to invalid risk assessment. Probability numbers are easily attainable for series-manufactured items or items with a large historical or scientific database. To use quantitative probability analysis for state-of-the-art hardware, in outer-envelope design conditions, is both misleading and dangerous. Therefore, one is forced to assign a qualitative designation for probability of occurrence. An example of a qualitative risk matrix is shown in Table 1.

TABLE 1. Risk assessment code (RAC) matrix.

	Probability Estimate			
Severity Class	A	B	C	D
I	1	1	2	3
II	1	2	3	4
III	2	3	4	5
IV	3	4	5	6

RAC 1:	Considered imminent danger; requires immediate attention and initiation of abatement procedures.
RAC 2:	Considered serious and requires priority attention.
RAC 3-6:	Considered nonserious; however, a priority ranking is established for corrective measures.

Severity Classification

I.	Catastrophic	May cause death or major system destruction.
II.	Critical	May cause severe injury, occupational illness, or major property damage.
III.	Marginal	May cause minor injury, minor occupational illness, or minor property damage.
IV.	Negligible	Probably would not affect personnel safety or health, but is a violation of specific criteria.

Qualitative Probability

A.	Likely to occur immediately.
B.	Probably will occur in time.
C.	May occur in time.
D.	Unlikely to occur.

System Safety Implementation

In order to implement system safety into the program, it is necessary to have a safety representative as a permanent member of the task force. That person must not only have equal status to the other members, but must also be a participant. The representative will be charged with ensuring that all viable safety concerns are addressed and acted upon. An integrated system safety program is only useful if the system safety engineers have adequate power to confirm that safety issues are not only identified and resolved, but, more importantly, that controls are put into place.

A system safety review panel, comprising technical experts, should be established to review trade-off studies and decisions. The knowledge and experience of technical experts at this management level would be utilized to the fullest in order to appropriately review system safety analyses. System reports will be generated by the system safety engineers on each of the various design, analysis, and development teams. The system safety engineering reports emanating from this level must be highly technical and comprehensive. To ensure adequate decision making, the review panel must be equally qualified.

The design, analysis, and development teams will have the most knowledgeable engineers for a particular component, system, or concept. If system safety engineers are not thoroughly integrated on these teams, investigation and research decisions will be made without adequate system safety engineering input.

Because an orbiting cryogenic depot is extremely complex, safety is critical. Unfortunately, pertinent safety information is lacking. The technology is sufficiently new that a system safety database has not been established. A special safety test program and test bed may be required to validate trade studies and create

the database. The primary purpose of this safety test program and test bed would be to investigate safety implications of various technologies in a highly structured and scientific manner. Numerous safety-related scenarios could be investigated before final design acceptance. To provide a cost-effective safety test program, the test bed need not be specially built. Many hazard potentials could be researched in the same test beds as the actual chosen hardware. Only certain destructive tests require remote facility testing (i.e., at NASA White Sands Test Facility). Because of the unique difficulties with zero gravity, some tests will need to be conducted in orbit—not only flight definition tests, but also an orbital subscale test bed (*Schuster et al.,* 1987).

Some Identified Top-Level Hazards of the Cryogenic Depot

Even though the OTV depot is still in the conceptual phase, many generic hazards are readily apparent. A private-sector-company PHA was conducted on various prephase A conceptual alternatives (*Aerospace Corporation,* 1971). Some of those hazards, along with other identified hazards, are fire/explosion, environmental and thermal, mechanical (vibration shock/acoustic), pressure, impact, biological (toxicity), electrical, and human factors (operations), and are discussed below. The generic hazards can be divided into various groups. Please note that this list will expand as the system is more clearly defined.

Fire and explosion are the most serious hazards. Improper mating of oxygen and hydrogen systems, thus allowing the incompatible fluids to mix, can cause an explosion. Another hazard is the rupture, or leakage, of a common bulkhead oxygen and hydrogen system. Because this system is being considered by NASA, a trade study investigating the safety concerns of common bulkhead vs. modular tank design would be interesting. A line or disconnect rupture during transfer operations may release sufficient propellant, causing a fire or explosion. In designing the depot one must assure that ignition is eliminated by preventing pneumatic impact on certain soft goods in oxygen lines.

The most obvious hazards are due to long-term environmental effects. The probable life cycle of the depot will be 10 to 20 years. During the entire life cycle, extreme thermal conditions will affect the depot. This creates heavy thermal loading and fatigue. There are not only thermal cycling problems associated with the cryogen (and its thermal stratification), but also the temperature variances of space. It is obviously not convenient to have the depot receive direct solar radiation. Space vacuum conditions will require careful design. Long-term vacuum, thermal, and radiation degradation of thermal coatings and mechanical components are significant concerns. Material selection and design strategies will be a key factor in confronting this problem. One probable temperature variance concern is the growth and shrinkage of components. Currently there are few data on the effects of large cryogenic storage systems submitted to long-term space environments.

Because of the large surface areas required, coupled with the vacuum environment, the area surrounding the storage facility could be a major heat sink for the cryogen. Adequate insulation around the storage tanks would be needed to prevent heat transfer in either direction. Another possible problem would be how noncondensible gas is purged from the system. The noncondensible gas is a potential hazard for pump cavitation. Studies should also investigate the effects of inadvertent dumping of large quantities of cryogens into Earth's orbit. Not only the combustible hazard, but also the pollution hazard is of concern.

Mechanical-related hazards are a significant category. Vibrations from pumps or other sources, in tandem with normal duty cycles, can cause serious mechanical fatigue problems of components or structures. Another important hazard is the lock-up of the deployment mechanism in either an unlatched, latched, or partially latched position. This immediately compromises the mission capability. Docking integrity will need to be studied thoroughly when a docking mechanism is defined.

Pressure system integrity is critical in the cryogenic system. Normal venting (or burping) would be made difficult if the depot is attached to the space station. Leakage could be catastrophic in this system. A transfer line leakage during active pumping could cause a high overboard dump rate with propellant possibly momentarily residing near the depot, producing a potentially combustible situation. Uncontrolled transfer line boiloff during transfer may cause pressure surges. A loss of pressure can lead to low net pressure suction head to the transfer pumps. Propellant leakage is not only a safety hazard but also a mission hazard. Small leak rates are critical due to the time and expense needed to launch and refuel the depot.

Cryogenic vessel overpressurization is a catastrophic hazard. An overpressurization may be caused by a water-hammer effect during transfer. If a runaway pumping situation exists, then the result could be loss of the pressure vessel. Failures in the propellant quantity measurement device can cause the system to be hardfilled. Ullage problems and thermal stratification may be handled by some type of system rotational or linear acceleration and fluid mixing. These added features introduce potential problems, such as overspeeding of rotational thrusters, thruster gas impingement on the depot, overspeeding of mixing fans, etc.

Approximately 7030 man-made objects are currently in orbit around the Earth. The majority of these tracked objects are from spacecraft breakup or explosion. There was a 10% increase in 1987 (*Johnson Space Center,* 1988). The inherent hazard caused by this situation is the problem of ancillary equipment impact. Depot and OTV dockings are risks themselves. Although NASA has a successful history of low-impact docking, study is needed to map out techniques for docking and rendezvous for this configuration. Gas impingement from the OTV on the depot would be catastrophic. Other impact sources are extravehicular activity (EVA) crewmember, EVA retriever robot, tools, and any other structural device that may be placed in this orbit. Micrometeoroids are another evident hazard. Pressure vessels holding 200,000 lb of oxygen and hydrogen will be very susceptible to this danger. High pressures and large surface areas considerably increase the hazard and risk.

If a cryogenic spill does occur, care must be taken to ensure that crewmembers do not introduce contaminated EVA suits or equipment into the space shuttle, space station, or OTV. Spilling of cryogens on equipment might also damage or affect the reliable operation of that hardware.

Electrical shocks are another hazard group. Improper electrical design or subsystem power surges may create problems. The danger is not only shock to personnel, but also damage or interference to equipment. Arcing at electrical interfaces is a potential hazard to crewmembers, and also may cause a fire or explosion. Arcing sources can originate from a variety of foci: EVA, OTV docking (or docking mechanisms), electric pump motors, and instrumentation and controls. Other radio signals from nearby orbiting spacecraft may affect delicate electronic signals.

Operational hazards (or human factors) add to the list. The use of incorrect procedures and ergonomically poorly designed machinery may cause problems. Emergency evacuation (from the depot area) options must be studied to verify that optimal personnel protection is always maintained.

CONCLUSIONS

The use of system safety techniques, applied in a carefully designed, methodical form, is the most effective avenue to enhanced system risk assessment and control. However, to efficiently engender safety, an integrated system safety approach must be used. The cryogenic propellant depot is at an optimal point in the program for safety to become involved.

The comprehensive system safety approach requires that system safety engineers become intimately involved in the program at all levels. Because the cryogenic depot is now at the prephase A conceptual stage, it is an opportune time to apply an active system safety participation. The system safety engineer must share in the conceptual and definition trade studies. Development of hardware, from pathfinder prototype, and then final flight article, requires safety cooperation.

Various technology requirements have already been identified for the cryogenic depot program. Though the conceptual development has not really begun, system safety can help in reviewing and investigating each technology. The safety effort is then led into the logical progression of technology development participation. Enhanced risk assessment requires that system safety supports three major areas: mission, manufacturing, and performance considerations. Consideration areas must address, and safely control such things as operational pressure, quantity measurements, manufacturing requirements, and environmental conditions.

System safety has well-developed and time-proven technologies that will further good risk assessment. Different analysis methods are applied at all stages of program development. The techniques have been used successfully on programs in the nuclear, chemical, and aerospace industries. Some of the appropriate methods to be used are preliminary hazard analysis, system hazard analysis, operating and support hazard analysis, fault tree analysis, and failure modes and effects analysis.

These safety tools will identify hazards and help categorize them for applying the sequence of hazard control. Optimum hazard control is through design, the least effective control is through procedures and training. The hazard criticality, coupled with probability of occurrence (though both may be qualitative), betters risk assessment and control.

System safety tools are meaningless if system safety is not an integral part of the team. The system safety engineer must be a working member (with equal status and voice) on design, analysis, test, manufacturing, and deployment teams. An independent system safety review panel, comprising technical experts, should ensure the objective and autonomous verification needed before deployment.

Although the depot is in a preconceptual phase, various generic hazards have been identified. The significant hazard categories are fire/explosion, environmental/thermal, mechanical, pressure, impact, biological, electrical, and operational.

The above-mentioned safety techniques, applied at the appropriate program phase, will explicate the hazards and verify the controls, thereby developing a serious and comprehensive risk assessment and control program. The cryogenic depot is replete with inherent hazards. A safe on-orbit cryogenic depot can be designed if an integrated system safety approach is applied.

REFERENCES

Aerospace Corporation (1971) *Orbiting Propellant Depot Safety, Vols. 1-3.* Aerospace Report No. ATR-71 (7223)-3. El Segundo, California.

Davis M., Allgeier R., Rogers T., and Rysavy G. (1970) *The Development of Cryogenic Storage Systems for Space Flight.* NASA Office of Technology Utilization, Washington, DC.

Johnson Space Center (1988) *Johnson Space Center Research and Technology Annual Report 1987.* NASA TM-100463. 106 pp.

Ride S. (1987) *Leadership and America's Future in Space; Report to the Administrator.* NASA Headquarters, Washington, DC. 63 pp.

Roland H. E. and Moriarty B. (1983) *System Safety Engineering and Management.* Wiley, New York. 339 pp.

Schuster J., Brown N., and Hueter U. (1987) *Evaluation of Cryogenic System Test Options for the OTV On-orbit Propellant Depot.* 22nd Thermophysics Conference. AIAA-87-1498. American Institute of Aeronautics and Astronautics, New York.

Stubbs R., Corban R., and Willanghby A. (1987) *On-Orbit Fuels Depot Technology Roadmap.* NASA Lewis Research Center, Cleveland, Ohio.

LUNAR LANDER STAGE REQUIREMENTS BASED ON THE CIVIL NEEDS DATA BASE

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This paper will examine the lunar lander stages that will be necessary for the future exploration and development of the Moon. Lunar lander stage sizing will be discussed based on the projected lunar payloads listed in the Civil Needs Data Base. Factors that will influence the lander stage design will be identified and discussed. Some of these factors will be (1) lunar orbiting and lunar surface lander bases; (2) implications of direct landing trajectories and landing from a parking orbit; (3) implications of landing site and parking orbit; (4) implications of landing site and parking orbit selection; (5) the use of expendable and reusable lander stages; and (6) the descent/ascent trajectories. Data relating the lunar lander stage design requirements to each of the above factors and others will be presented in parametric form. These data will provide useful design data that will be applicable to future mission model modifications and design studies.

As a result of the findings of the National Commission on Space and the Space Leadership Report by Dr. Sally Ride, there is a renewed interest in lunar exploration. Many current lunar study activities indicate the great potential for both scientific and technological benefits from a sustained, sequential lunar exploration progam that would culminate in the utilization of lunar resources. The results of many of the current conceptual engineering studies depend greatly on the specific assumptions made regarding the lunar exploration program. As part of the National Space Transportation and Support Study of 1985, NASA created the Civil Needs Data Base (CNDB). Included in the database is an outline for a sustained lunar exploration program and a detailed listing of payloads that would be transported to the Moon within that program. It was NASA's intention that this database would provide a set of program assumptions that could be used for subsequent engineering studies.

The lunar lander stages that would transport payloads to the surface would be important elements of the lunar program. The design requirements for these landers depend on many factors. It is important that these requirements be determined with the overall lunar program and the entire set of payloads within that program taken into account. The CNDB provides the necessary progam assumptions and payload characteristics to serve as a basis for determining lunar lander stage requirements. There are many requirements that can be determined based on the information in the CNDB. Among these are the requirements imposed by the payloads, flight rate, propulsive requirements, configuration constraints, as well as several other mission factors.

In order to determine requirements for a lunar lander stage, it was necessary to consider two important aspects of the missions that the stages would perform. The first mission aspect that was considered was the overall mission scenario within which the stage would be operated. This aspect of the mission determines the basing options for the stage, overall energy requirements, mission duration, and the type of mission operations associated with the transportation of payloads to the lunar surface. The second mission aspect that was considered was the nature of the lunar payloads themselves. Rather than size the stages for a given maximum payload weight it was necessary to consider the entire range of payload transportation requirements for a sequential build-up of a sustained lunar exploration program. Factors that were considered were the range of payload weights and sizes as well as the distribution of payloads within the year-by-year sequence of the lunar exploration program.

The mission scenario that was chosen in this paper was determined by considering several scenario options. These options consisted of different mission profiles and stage-base locations. There were three mission profiles that were considered, each one using a different transfer trajectory between the Earth and Moon. The first was landing from a direct transfer from low Earth orbit to the lunar surface, the second was landing from a lunar orbit, and the third was landing from the L1 libration point. These mission profiles are shown in Fig. 1 (*Martin Marietta,* 1987, pp. 131, 133, 137). The direct transfer appeared to be the most economical in terms of total propellant requirements and may be operationally less complex; however, the lander would be required to carry more than twice as much propellant than needed to land from a lunar orbit (*Martin Marietta,* 1987, p. 47). The option of landing from the Earth-Moon L1 libration point appears to offer no real advantage as far as propellant requirements for either the lunar transfer stages or the lander itself. It required 63% more propellant to land from the L1 point than from lunar orbit and 14% more propellant to travel from low Earth orbit to the L1 point than to lunar orbit (*Martin Marietta,* 1987, p. 47).

In addition to the obvious trajectory and vehicle performance considerations, the available options for establishing a transportation node or lander stage base within each mission scenario were evaluated. This assumes that the need for such a node will exist and provide operational benefits to the overall lunar transportation system. A transportation node can be envisioned as either the location of some actual platform that would provide services to spacecraft traveling to the node, or as a point in space where two spacecraft meet to perform specific mission operations. The most likely use of a transportation node would be to utilize lunar-produced resources such as liquid oxygen. The node

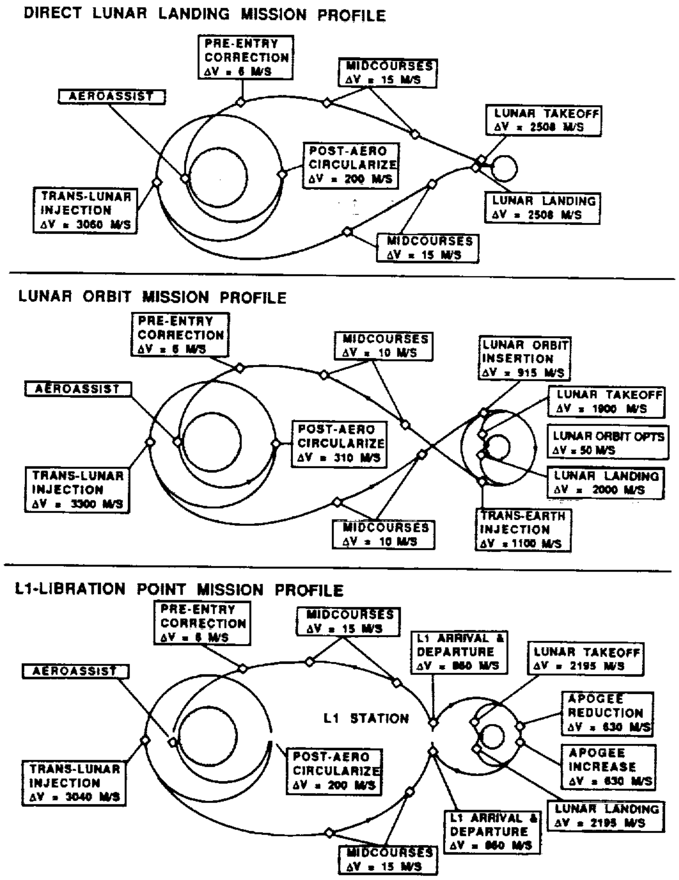

Fig. 1. Lunar mission profile options.

could serve as a base or operating location for reusable lander stages. This would reduce the transportation requirements from Earth.

The direct transfer scenario would allow for transportation nodes in either low Earth orbit or on the lunar surface. The lunar orbit scenario would allow for nodes in low Earth orbit, low lunar orbit or on the lunar surface. The L1 libration point mission scenario would allow for nodes in low Earth orbit, at the libration point or on the lunar surface. Each of these locations has advantages and disadvantages as a transportation node.

If a lunar lander transportation node were located in low Earth orbit, then lunar resources would be conveniently located where the majority of space activity occurs. However, a low Earth orbit transportation node would be a poor location for a lunar communications link or a surface sensing platform due to the large distance from the Moon and the fact that only one side of the Moon would be visible. The greatest disadvantage of basing the lunar landers in low Earth orbit would be the performance penalty of transporting the stages back and forth between the Earth and Moon each mission.

A transportation node at the L1 libration point would eliminate the need to transport landers back and forth from low Earth orbit. Unfortunately, any benefit would be more than offset by the increase in the total energy required to transport payloads or lunar resources between the Earth and Moon within the overall mission scenario. A libration point would provide limited capability as a location for a communications link or lunar surface sensing platform since only one side of the Moon is visible. A libration point node would also have some operational disadvantages. The L1 point is not a stable libration point. The position of a platform at this node would have to be maintained by a propulsion system on the platform. This requirement could be diminished somewhat by placing the stage base in a "halo orbit" around the libration point, but this would add complexities to rendezvous operations with other spacecraft.

A transportation node on the lunar surface would allow reusable landers to be based at the same place lunar resources are produced. This would eliminate the need for an orbiting service station (*Eagle Engineering*, 1984, p. 4). Unfortunately, a strictly surface-based lander would have some operating restrictions since it would be somewhat bound to a fixed location on the lunar surface. If the lander base is located on the lunar surface, propellants brought from Earth would have to be carried all the way to the surface (*Eagle Engineering*, 1984, p. 26). This would reduce the overall payload capacity of the lunar transportation system.

An orbiting lunar station would be the most efficient location for a lander base in terms of payload vs. weight in low Earth orbit but would require propellant transfer in zero-gravity (*Woodcock*, 1985, p. 120). A fuel depot in lunar orbit would eliminate the need to transport propellants for the landers to the surface for storage. There are also some other advantages for a lunar orbit transportation node. One advantage would be better access to the farside of the Moon. Another advantage is that the lander would be less bound to a particular location on the lunar surface. Probably, the biggest advantage of using lunar orbit as a transportation node is that it provides a convenient location for exchanging payloads, crews, and lunar resources between transfer stages and lunar landers. Lunar sensing could be conducted from an orbiting station; in fact the lunar service station could be a derivative of the low Earth orbit space station, with some identical elements in addition to propellant storage capability (*JSC*, 1984, p. D-7).

The mission scenario that was chosen for this paper takes advantage of the benefits of transportation nodes on both the lunar surface and in lunar orbit. Therefore, the lunar orbit mission profile was selected. By using both locations as transportation nodes, there would be more flexibility allowed in the operation of the lunar landers. They could be maintained at a permanent surface base where liquid oxygen is produced, or parked at a lunar orbit service station. If sufficient propellant quantities could be stored in orbit, the landers would be less dependent on the surface base. Even if no lunar orbit service station were available, lunar orbit would be a useful transportation node. Payloads and crews could be efficiently exchanged in lunar orbit. Liquid oxygen from the Moon could be exchanged for liquid hydrogen from Earth. The lander could serve as its own storage facility. Its oxygen tanks would be filled on the surface and its hydrogen tanks would be filled in lunar orbit.

In order to achieve the greatest efficiency possible from the mission scenario, it must be carefully designed to minimize the overall energy requirements without restricting lunar exploration options. A scenario that would achieve these goals has been described by *Woodcock* (1985). The lunar orbit used in this scenario was a 100-km altitude polar orbit, which permits access to any point on the lunar surface since the Moon rotates underneath this orbit once every 27 days. The Earth orbit from

which lunar transfers would originate was assumed to be the space station orbit at approximately 500 km altitude and 28.5° inclination. Due to the precession of the space station orbit about the Earth's polar axis and its orientation with respect to the plane of the Moon's orbit, there is an opportunity for an in-plane transfer to the Moon approximately every 9 days (*Woodcock,* 1985). In order to minimize the energy requirements of the transfer trajectory, it should be designed so that the approach vector of the transfer vehicle when it reaches the Moon is in the plane of the lunar polar orbit. Similarly, the approach vector of the transfer vehicle when it returns to Earth should be in the plane of the space station orbit. A trajectory that would satisfy both these conditions is called a synchronized Earth-Moon round trip (*Woodcock,* 1985). Synchronism is possible when the combined angular displacements, Θ, of the transfer vehicle, Moon, and line of intersection of the transfer orbit plane and space station orbit plane add up to be a complete circle (i.e., $\Theta = N\pi$, $N = 2, 4, 6\ldots$).

A synchronized round-trip trajectory is shown in Fig. 2. The angular displacements of the transfer vehicle, Moon, and line of intersection of the transfer orbit and space station orbit is shown in Fig. 3. The combined angular displacement is shown to be 720° or two complete circles. This particular trajectory requires about 4 days for the transfer between the Earth and Moon and a 15-day stay time in lunar orbit. The entire round trip mission takes approximately 23 days. The ΔV requirements for this trajectory are listed below (from *Woodcock,* 1985, p. 119)

Trans-Lunar Injection	=	3139 M/S
Lunar Orbit Insertion	=	915 M/S
Trans-Earth Injection	=	906 M/S
Earth Orbit Insertion	=	3061 M/S (All Propulsive)
or	=	200 M/S (With Aerobrake)

It is obvious from the preceding discussion that the mission scenario and lander basing strategy would have a large influence on the requirements for a lunar lander stage. In addition to the mission scenario, it was necessary to examine one other important aspect of the mission to determine its influence on lander requirements. This second mission aspect was the nature of the payloads that would be carried on the lander stages. The most important factors in this mission aspect were the payload weights and sizes, delivery sequence, the development of the lunar infrastructure, and the number of lander flights. Most of the assumptions concerning these factors were taken directly from the NASA CNDB.

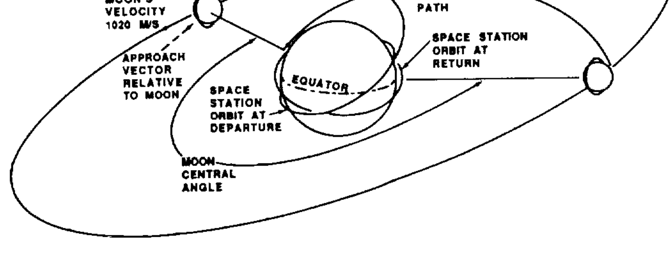

Fig. 2. Synchronized Earth-Moon round trip (*Woodcock,* 1985, p. 116).

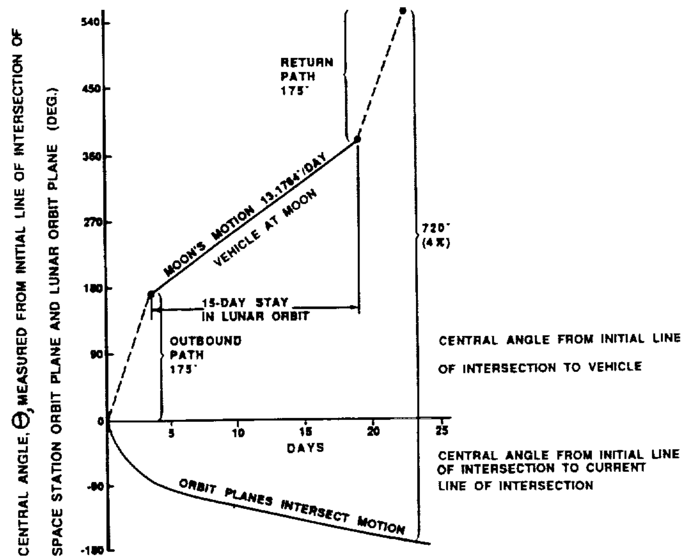

Fig. 3. Angular displacements of the transfer vehicle, Moon, and line of intersection of the transfer orbit plane and space station orbit plane.

The CNDB was developed by NASA in response to the National Security Study Directive of May 1985 (*McCauley,* p. i). Its purpose is to identify technology development necessary to meet U.S. space objectives for the period 1995 to 2010 and to support studies of future space transportation systems. It is a compilation of several databases and other independent inputs including the following sources: Battelle Outside Users Payload Model - 1985; Space Station Mission Data Base; NASA Technology Model; Space Station Transportation Requirements - OSS; Other Advanced/Conceptual Mission Studies; NASA Program Offices; Other Civil Agencies; Dept. of Energy; Dept. of Agriculture; Dept. of Interior, etc.; and National Commission on Space (*McCauley,* 1986, p. 4-2).

The lunar program portion of the CNDB contains a representative listing of all the payloads that would be transported to the Moon during the period 1995 to 2010. The description of each payload includes weight delivered to the lunar surface, payload dimensions, and flight schedule as well as other information. These payloads are defined within the framework of an assumed build-up of a lunar infrastructure that would lead to a continuous human presence on the Moon and utilization of lunar resources especially liquid oxygen. Both the assumed infrastructure and the physical characteristics of the payloads influence the requirements for the lunar landers that would be used.

The CNDB assumes that the lunar infrastructure would be created in three phases (*McCauley,* 1986). The first phase, which would last until 1999, would consist of unmanned robotic exploration of the Moon. Activities during this phase would include searching for frozen water or other raw materials and finding suitable locations for a lunar base. The second phase of lunar exploration would last from the year 2000 through 2004. A temporarily staffed outpost would be established on the Moon. A crew of four would visit the outpost for limited stay times of 14 to 30 days. During this phase, much of a permanent lunar base would be constructed, pilot plants for lunar resource production would be built, and scientific experiments would be carried out.

It was assumed that all landers and ascent vehicles would be expendable since no lunar-produced liquid oxygen would be available. The third phase of lunar exploration would begin in 2005. There would be continuous human presence on the Moon. Crews would be rotated from Earth to maintain a staff of 8 to 12 people at the base (*McCauley,* 1986). It was assumed that liquid oxygen would be produced and that a reusable lunar lander would be used to transport payloads between the surface and lunar orbit. Payloads and crews would be exchanged between transfer vehicles and the lander at a lunar orbit service station. The three-phase infrastructure just described clearly influences whether the lunar lander should be expendable or reusable. To determine additional requirements for the landers, it is necessary to examine the lunar payloads and the delivery sequence of those payloads. The lunar payloads in the CNDB are listed in Table 1. They are grouped by year but there is no particular sequence assumed within each year. Most of the payloads listed in the CNDB are based on the study performed by *Battelle Columbus Division* (1987). They represent a wide range of possible lunar activities from astronomy to life sciences research. Included in the listing of payloads are the lunar base elements, crews, crew logistics, and descent and ascent vehicles. Each payload is given an identification number in the listing. The information shown for each payload consists of (1) weight delivered to the surface, (2) weight returned from the surface, (3) payload dimensions, and (4) number of units delivered during the year. It should be noted that there is more information available on these payloads in the CNDB than listed in Table 1.

One other factor was included in order to determine lander requirements imposed by the payloads, namely, the number of flights (or landings) that would be used to transport the payloads to the lunar surface. The number of flights determines what the payload-carrying requirements would be in terms of both payload weight and payload volume. Obviously, if fewer flights are used,

TABLE 1. Civil Needs Data Base lunar payloads.

PLID	Payload Name	Descent Weight	Ascent Weight	Length	Width	Height	Units
1996							
5024	Lunar Polar Sample Return	8,800	—	22.1	13.0	0.0	1
1997							
5024	Lunar Polar Sample Return	8,800	000	22.1	13.0	0.0	1
5034	Rover (Surf Surv)	2,200	—	14.6	8.0	0.0	1
1998							
5034	Rover (Surf Surv)	2,200	—	14.6	8.0	0.0	1
1999							
5002	Lunar Base Crew Rotation (04/014)	1,800	1,800	3.0	12.0	6.0	1
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	1
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5034	Rover (Surf Surv)	2,200	—	14.6	8.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	2
5052	Lunar Base Crew Logistics (04/014)	300	—	3.0	3.0	3.7	1
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	1
2000							
5002	Lunar Base Crew Rotation (04/014)	1,800	1,800	3.0	12.0	6.0	2
5008	Lunar Based SETI	20,000	—	32.0	14.0	0.0	1
5009	Lunar Far UV Telescope	10,000	—	15.0	6.0	10.0	1
5013	Plant (Power)(Initial)	7,000	—	20.0	15.0	0.0	1
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	2
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5031	Rover (Unpressurized)	4,000	—	17.0	10.0	0.0	1
5032	Soil Mover/Crane/Constr. PH-2	38,500	—	30.0	19.0	0.0	1
5036	Comm Relay (Surf) PH-2	2,500	—	15.9	10.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	4
5052	Lunar Base Crew Logistics (04/014)	300	—	3.0	3.0	3.7	2
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	2

TABLE 1. (continued).

PLID	Payload Name	Descent Weight	Ascent Weight	Length	Width	Height	Units
2001							
5002	Lunar Base Crew Rotation (04/014)	1,800	1,800	3.0	12.0	6.0	3
5013	Plant (Power)(Initial)	7,000	—	20.0	15.0	0.0	2
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	3
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5028	Plant (Liquid Oxygen)(Pilot)	38,500	—	36.0	14.0	0.0	1
5037	Optical Interferometer Telescope	15,000	—	15.0	15.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	6
5052	Lunar Base Crew Logisitics (04/014)	300	—	3.0	3.0	3.7	3
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	3
5082	Module Interface Mode	8,200	—	15.0	20.0	0.0	1
2002							
5011	Habitat Module PH-2	38,500	—	42.6	16.0	0.0	1
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	4
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	6
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	4
5068	Lunar Base Crew Rotation (04/030)	1,800	1,800	3.0	12.0	6.0	4
5071	Mining Equipment (Oxygen)	38,500	—	13.7	13.7	13.7	1
5075	Lunar Base Crew Logistics (04/030)	900	—	5.0	5.0	6.0	4
2003							
5006	Plant (Power)(Advanced)	38,500	—	36.0	14.0	0.0	1
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	4
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5029	Plant (Liquid Oxygen)(Production)	33,333	—	36.0	14.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	7
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	4
5068	Lunar Base Crew Rotation (04/030)	1,800	1,800	3.0	12.0	6.0	4
5073	Geochemical Materials Lab	38,500	—	36.0	14.0	0.0	1
5075	Lunar Base Crew Logistics (04/030)	900	—	5.0	5.0	6.0	4
5082	Module Interface Mode	8,200	—	15.0	20.0	0.0	1
2004							
5010	Lunar Far UV Telescope	2,000	—	8.0	4.0	4.0	1
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	4
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5029	Plant (Liquid Oxygen) (Production)	33,333	—	36.0	14.0	0.0	2
5031	Rover (Unpressurized)	4,000	—	17.0	10.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	6
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	4
5068	Lunar Base Crew Rotation (04/030)	1,800	1,800	3.0	12.0	6.0	4

TABLE 1. (continued).

PLID	Payload Name	Descent Weight	Ascent Weight	Length	Width	Height	Units
2004 continued							
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5075	Lunar Base Crew Logistics (04/030)	900	—	5.0	5.0	6.0	4
2005							
5015	Life Science Research Facility	40,000	—	36.0	14.0	0.0	2
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	4
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5036	Comm Relay (Surf) PH-2	2,500	—	15.9	10.0	0.0	1
5050	Lunar Lander Vehicle (Expendable)	41,660	—	13.0	14.0	0.0	7
5053	Lunar Ascent Vehicle (Expendable)	16,275	—	5.0	14.0	0.0	4
5065	Lunar Base Deep Drilling	4,000	—	10.0	10.0	8.0	1
5067	Lunar Base Crew Rotation (04/180)	1,800	1,800	3.0	12.0	6.0	4
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5076	Lunar Base Crew Logistics (04/180)	6,400	—	6.0	6.0	8.0	4
5079	Life Science Research Facility (Node)	8,200	—	15.0	20.0	0.0	1
5082	Module Interface Node	8,200	—	15.0	20.0	0.0	1
2006							
5012	Habitat Module PH-3	38,500	—	36.0	14.0	0.0	1
5014	Servicing Facility Shop Module	38,500	—	36.0	14.0	0.0	1
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	3
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5062	Low Frequency Radio Array	20,000	—	50.0	20.0	10.0	1
5067	Lunar Base Crew Rotation (04/180)	1,800	1,800	3.0	12.0	6.0	3
5070	Life Science Research Facility	500	100	4.0	4.0	3.0	1
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5076	Lunar Base Crew Logistics (04/180)	6,400	—	6.0	6.0	8.0	3
5080	Lunar Lander Vehicle Logistics (LH_2)	7,000	—	7.0	16.0		6
5054	Lunar Lander (Reusable)	11,500	—	15.0	14.0		1
5019	Personnel Transfer Module (6 Man)	7,200	—	10.0	12.0		1
2007							
5018	Personnel Transfer Module (4 Man)	13,200	13,200	12.0	14.0	0.0	4
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5030	Rover (Pressurized)	38,500	—	36.0	14.0	0.0	1
5035	Comm Relay (Surf) PH-3	2,500	—	15.9	10.0	0.0	1
5061	Low Frequency Radio Array	1,000	—	4.0	4.0	4.0	1
5064	Radio Interferometry	20,000	—	32.0	14.0	0.0	1
5067	Lunar Base Crew Rotation (04/180)	1,800	1,800	3.0	12.0	6.0	4
5070	Life Science Research Facility	500	100	4.0	4.0	3.0	1
5072	Servicing Facility Shop Module	2,000	—	7.0	7.0	6.0	1
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5076	Lunar Base Crew Logistics (04/180)	6,400	—	6.0	6.0	8.0	4
5080	Lunar Lander Vehicle Logistics (LH_2)	7,000	—	7.0	16.0		7

TABLE 1. (continued).

PLID	Payload Name	Descent Weight	Ascent Weight	Length	Width	Height	Units
2008							
5010	Lunar Far UV Telescope	2,000	—	8.0	4.0	4.0	1
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5035	Comm Relay (Surf) PH-3	2,500	—	15.9	10.0	0.0	1
5061	Low Frequency Radio Array	1,000	—	4.0	4.0	4.0	1
5063	Radio Interferometry	1,000	—	4.0	4.0	4.0	1
5066	Lunar Base Crew Rotation (06/180)	2,700	2,700	3.0	18.0	6.0	4
5070	Life Science Research Facility	500	100	4.0	4.0	3.0	1
5072	Servicing Facility Shop Module	2,000	—	7.0	7.0	6.0	1
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5077	Lunar Base Crew Logistics (06/180)	9,600	—	8.0	8.0	10.0	4
5080	Lunar Lander Vehicle Logistics (LH_2)	7,000	—	7.0	16.0		6
2009							
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5033	Soil Mover/Crane/Constr. PH-3	38,500	—	36.0	14.0	0.0	1
5061	Low Frequency Radio Array	1,000	—	4.0	4.0	4.0	1
5063	Radio Interferometry	1,000	—	4.0	4.0	4.0	1
5066	Lunar Base Crew Rotation (06/180)	2,700	2,700	3.0	18.0	6.0	4
5070	Life Science Research Facility	500	100	4.0	4.0	3.0	1
5072	Servicing Facility Shop Module	2,000	—	7.0	7.0	6.0	1
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5077	Lunar Base Crew Logistics (06/180)	9,600	—	8.0	8.0	10.0	4
5080	Lunar Lander Vehicle Logistics (LH_2)	7,000	—	7.0	16.0		7
2010							
5022	Plant (Ceramics)	38,500	—	36.0	14.0	0.0	1
5027	Lunar Science and Field Geology	500	100	15.0	5.0	0.0	1
5061	Low Frequency Radio Array	1,000	—	4.0	4.0	4.0	1
5063	Radio Interferometry	1,000	—	4.0	4.0	4.0	1
5066	Lunar Base Crew Rotation (06/180)	2,700	2,700	3.0	18.0	6.0	4
5070	Life Science Research Facility	500	100	4.0	4.0	3.0	1
5072	Servicing Facility Shop Module	2,000	—	7.0	7.0	6.0	1
5074	Geochemical Materials Lab	500	—	5.0	5.0	2.0	1
5077	Lunar Base Crew Logistics (06/180)	9,600	—	8.0	8.0	10.0	4
5080	Lunar Lander Vehicle Logistics (LH_2)	7,000	—	7.0	16.0		6

more payload would have to be carried each flight. This factor presented several possibilities for transporting the payloads. Three options were chosen for this paper. The first was to use the number of flights outlined in the CNDB. This would serve as a baseline. The second would be to use the minimum number of flights. And finally, the third would have no limit on the number of flights.

The number of flights used in the CNDB is implied by the number of descent and ascent stages (payload I.D.s 5050 and 5053) listed for each year through 2005 and by the number of lunar lander logistics (liquid hydrogen) deliveries (payload I.D. 5080) for each year after that. The CNDB uses the assumption that once lunar-produced liquid oxygen is available (by 2006), a reusable lunar lander would perform all payload deliveries. The number of flights used for the baseline option in this paper uses the same vehicle assumptions as the CNDB except for two modifications. The first is that rather than use the reusable lander for all flights after 2005, an expendable lander would be used for the unmanned payload delivery missions. This would reduce the payload requirement for the reusable lander by 43%. The

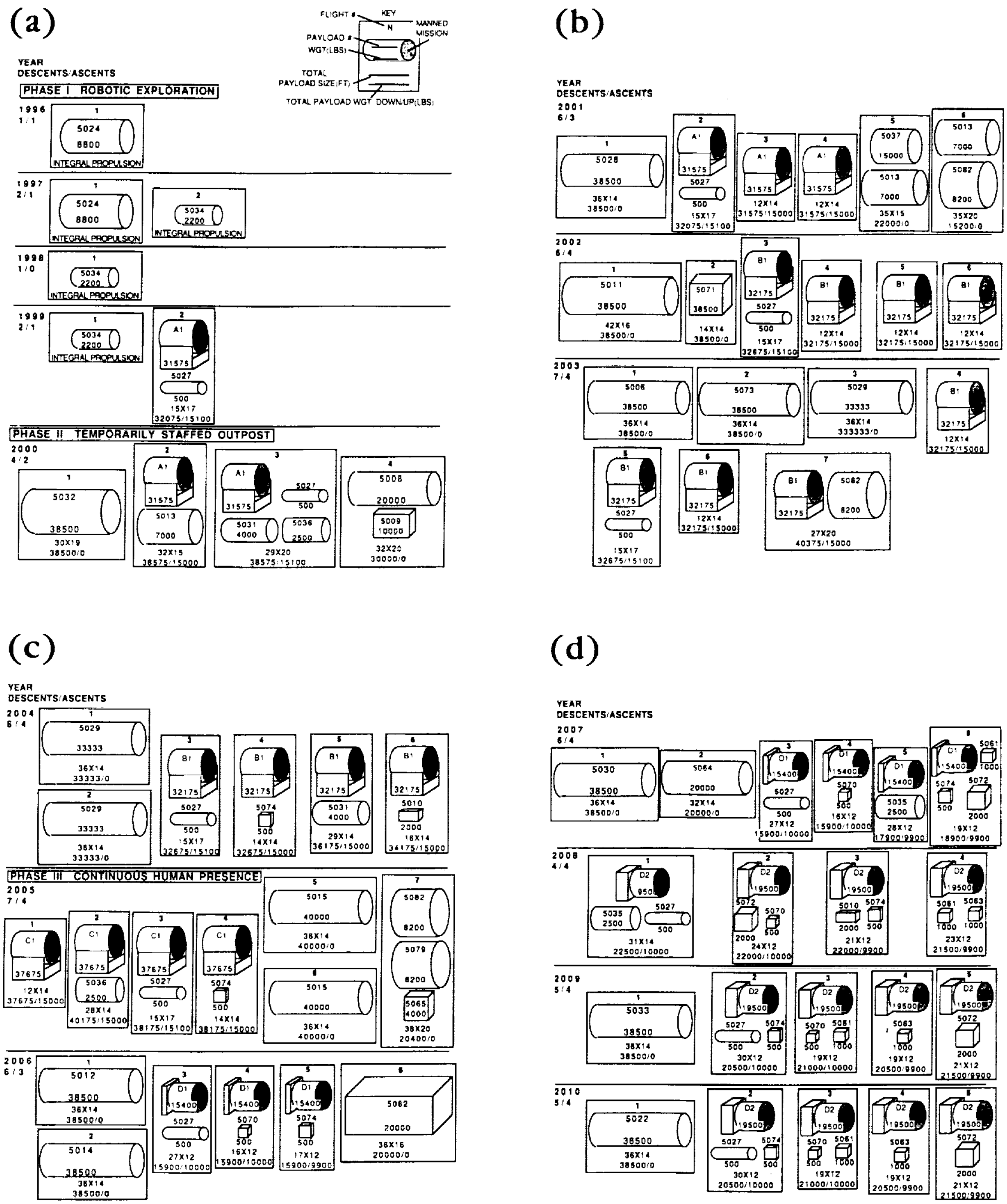

Fig. 4. CNDB lunar payloads: option 1. **(a)** 1996-2000; **(b)** 2001-2003; **(c)** 2004-2006; and **(d)** 2007-2010. Special payload combinations are

PLID	Payloads included	Wt.
A1	5018, 5002, 5052, 5053	31575
B1	5018, 5068, 5075, 5053	32175
C1	5018, 5067, 5076, 5053	37675
D1	5019, 5067, 5076	15400
D2	5019, 5066, 5077	19500

other modification to the CNDB assumptions was to eliminate some of the flights between 2007 and 2010. This would result in a more efficient use of the reusable lander.

The lunar payloads for each year in the CNDB are depicted in Fig. 4. They are drawn to scale, based on the dimensions given in Table 1. Within each year, the payloads are divided into groups representing each flight necessary for option 1. The groups were arranged in such a way as to keep the payload weight and size to a minimum for each flight. The total payload size represents the smallest cargo pad area that would be occupied by the payloads for that particular flight. The data contained in Fig. 4 were used to determine several requirements for the lunar lander stages.

The first requirement for the landers that was found was the maximum payload weight for each of the landers. These requirements can be seen in Fig. 5, which summarizes the payload carrying requirements for option 1. This figure shows that the maximum descent payload weight for the expendable lander is about 41,000 lb and for the reusable lander is 23,000 lb. Figure 4 was also used to determine several other stage requirements. The maximum payload for the expendable ascent vehicle is 15,100 lb and the maximum ascent payload for the reusable lander is 10,000 lb. In addition to payload weight requirements, payload size requirements were found. The maximum payload size for the expendable lander was 42 × 20 ft, for the expendable ascent vehicle it was 12 × 14 ft, and for the reusable lander it was 31 × 14 ft. The final requirements determined from Fig. 4 were the number of trips and the number of vehicles required during the period 1996 to 2010. There are 63 descents and 41 ascents required for option 1. The vehicle requirements for option 1 are 44 expendable landers, 22 expendable ascent vehicles, and at least 1 reusable lander for a total of 67 vehicles.

The second flight option that was considered minimized the number of flights. For this option, it was assumed that after initial robotic exploration, there would be flights to the Moon only when it was necessary to send a crew. For this option, the number of flights corresponds to the number of manned missions listed in the CNDB. All the payloads would be transported on these flights. Also, all flights after 2005 would use a reusable lander as was originally assumed in the CNDB. Figure 6 shows the CNDB lunar payloads grouped by year and flight number for option 2. Since there are fewer flights, more payload must be carried on each flight. The payload carrying requirements for option 2 are summarized in Fig. 7. It can be seen that the maximum descent payload for the expendable lander is approximately 78,000 lb and for the reusable lander is 58,000 lb. Figure 6 was also used to determine other requirements. The maximum ascent payload for the expendable ascent vehicle was found to be 15,000 lb and for the reusable lander, 10,000lb. The payload sizes were found to be 42 × 30 ft for the expendable lander, 12 × 14 ft for the expendable ascent vehicle, and 36 × 28 ft for the reusable lander. There are 41 descents and 41 ascents required for this option. Finally,

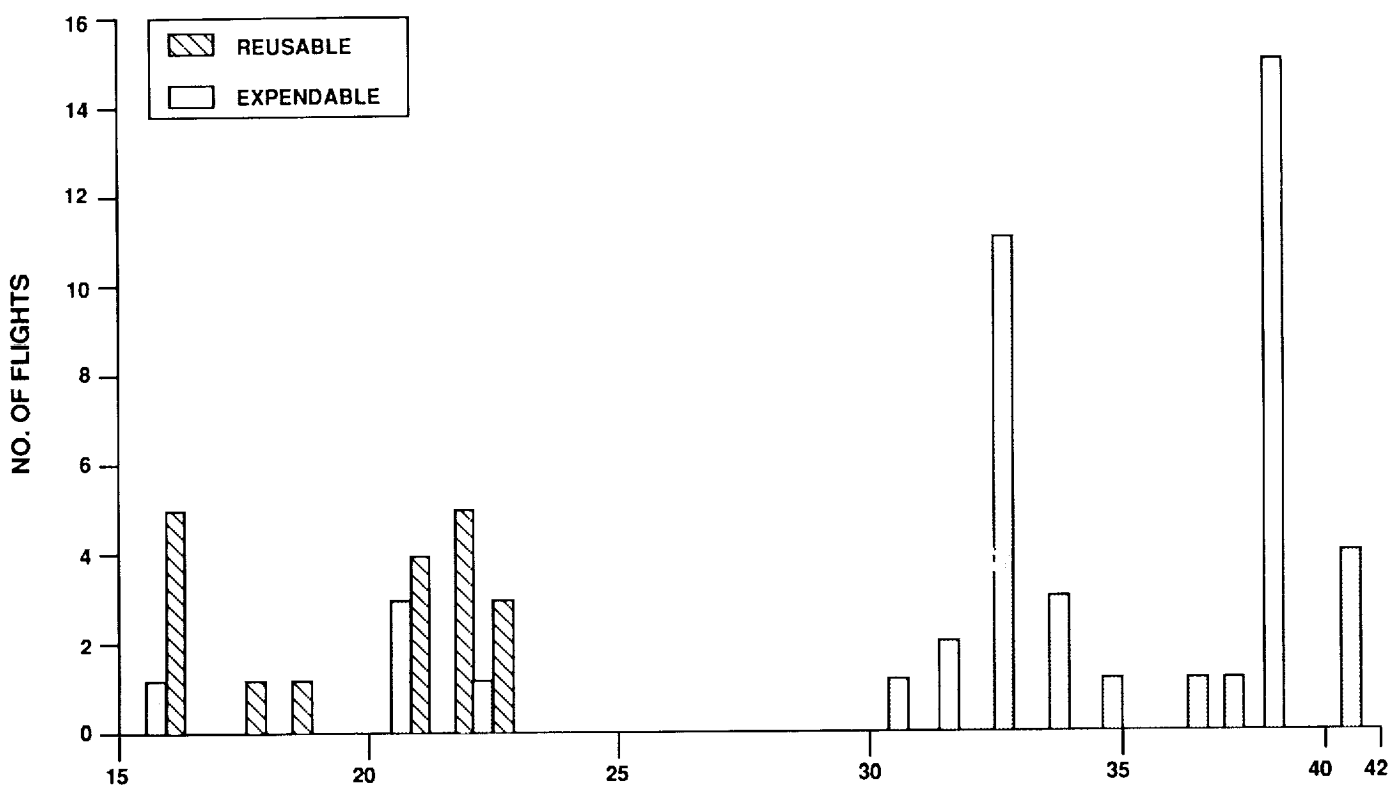

Fig. 5. Number of lunar lander flights required for various payload weights in option 1.

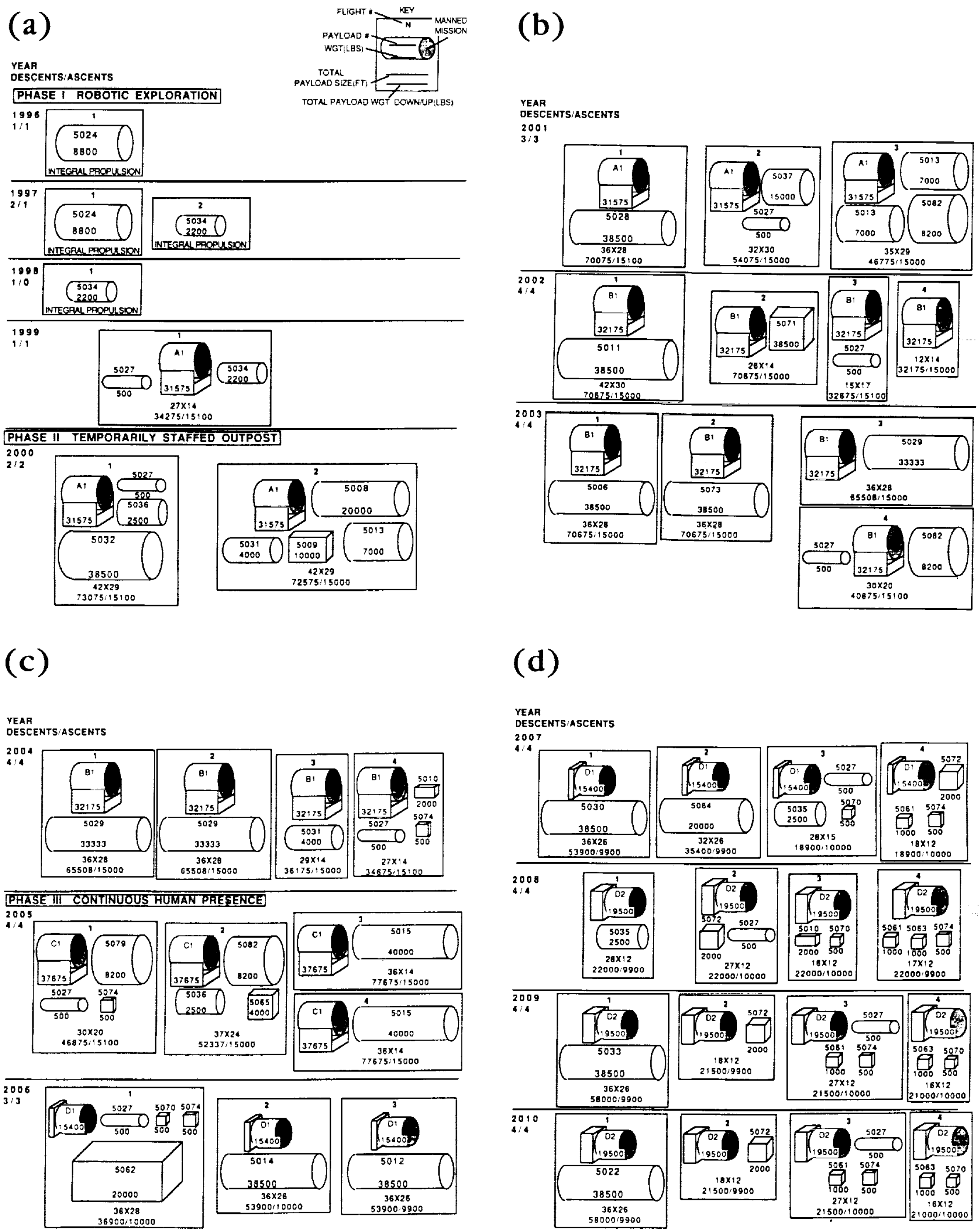

Fig. 6. CNDB lunar payloads: option 2. **(a)** 1996-2000; **(b)** 2001-2003; **(c)** 2004-2006; and **(d)** 2007-2010. Special payload combinations as shown in Fig. 4.

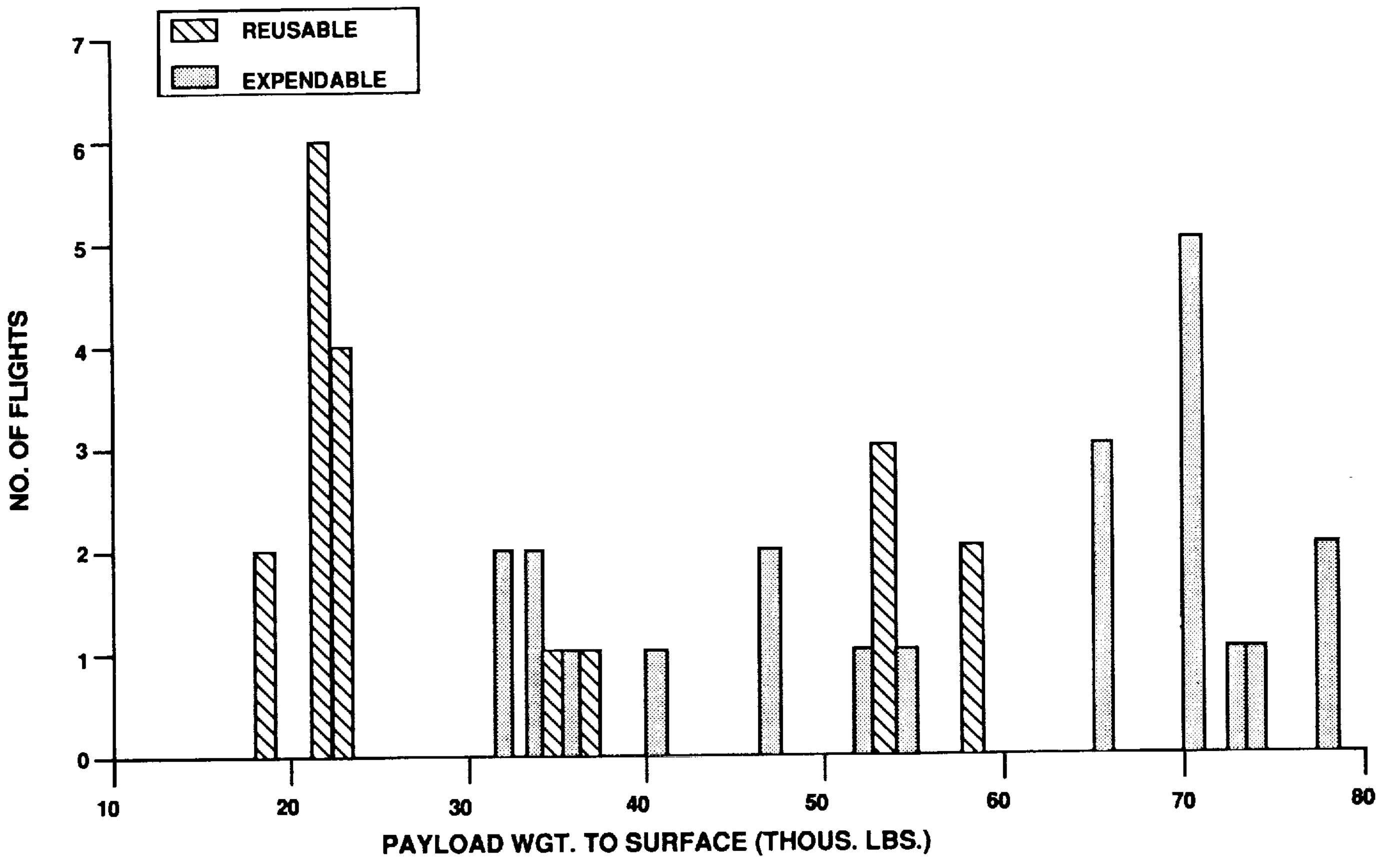

Fig. 7. Number of lunar lander flights required for various payload weights in option 2.

there are 22 expendable landers, 22 expendable ascent vehicles, and at least 1 reusable lander required. The total number of vehicles for option 2 is 45.

The final payload delivery option identified lunar lander requirements if there were no limit on the number of trips to the lunar surface. The key assumption for this option was that every payload weighing more than 500 lb would land on the Moon separately. Payloads weighing up to 500 lb would be transported on manned missions since they usually represent some experiment conducted by the crew. Figure 8 shows the payloads for each flight for each year. It is obvious that using one type of vehicle to transport both 40,000-lb payloads and 1000-lb payloads would not be efficient. Therefore, multiple vehicles were considered, each one designed to carry a specific range of payload weights.

The number of types of vehicles and the payload ranges for each of the stages was found by looking at the overall range of payload carrying requirements shown in Fig. 9. The conclusions that were drawn indicated that two sizes of expendable landers are required and that two sizes of reusable landers are required. A total of 39 expendable landers with a payload capability of 20,000 to 41,000 lb with a payload size of 36 × 19 ft are required. Fifteen expendable landers for payloads under 16,000 lb and 30 × 15 ft in size are required. The first reusable lander would have a descent payload requirement of 20,000 lb and an ascent payload of 10,000 lb. The largest payload size would be 36 × 17 ft. The second reusable lander would have a descent payload of just 3500 lb and no ascent payload requirement. The stage size would be influenced more by the size of the propellant tanks than the payloads. The expendable ascent vehicle requirements were identical to the first two options. For this option, there would be 89 descents to the surface and 57 ascents. A total of at least 78 vehicles would be required.

All the lander requirements determined up to this point were imposed by the payloads. In order to determine other requirements related to the propulsion system or stage weight, it was necessary to develop a conceptual design for the lunar lander stages. First, the propulsion requirements were determined, then the stage weight was evaluated. This information led to the development of a scaling equation for the lander stages.

The descent and ascent trajectories were assumed to be very similar to those used during the Apollo Program (*Alphin et al.* 1968; *Bellcom Inc.*, 1968; *Martin Marietta*, 1987). An initial thrust-to-weight ratio of 0.6 was asumed for both descent and

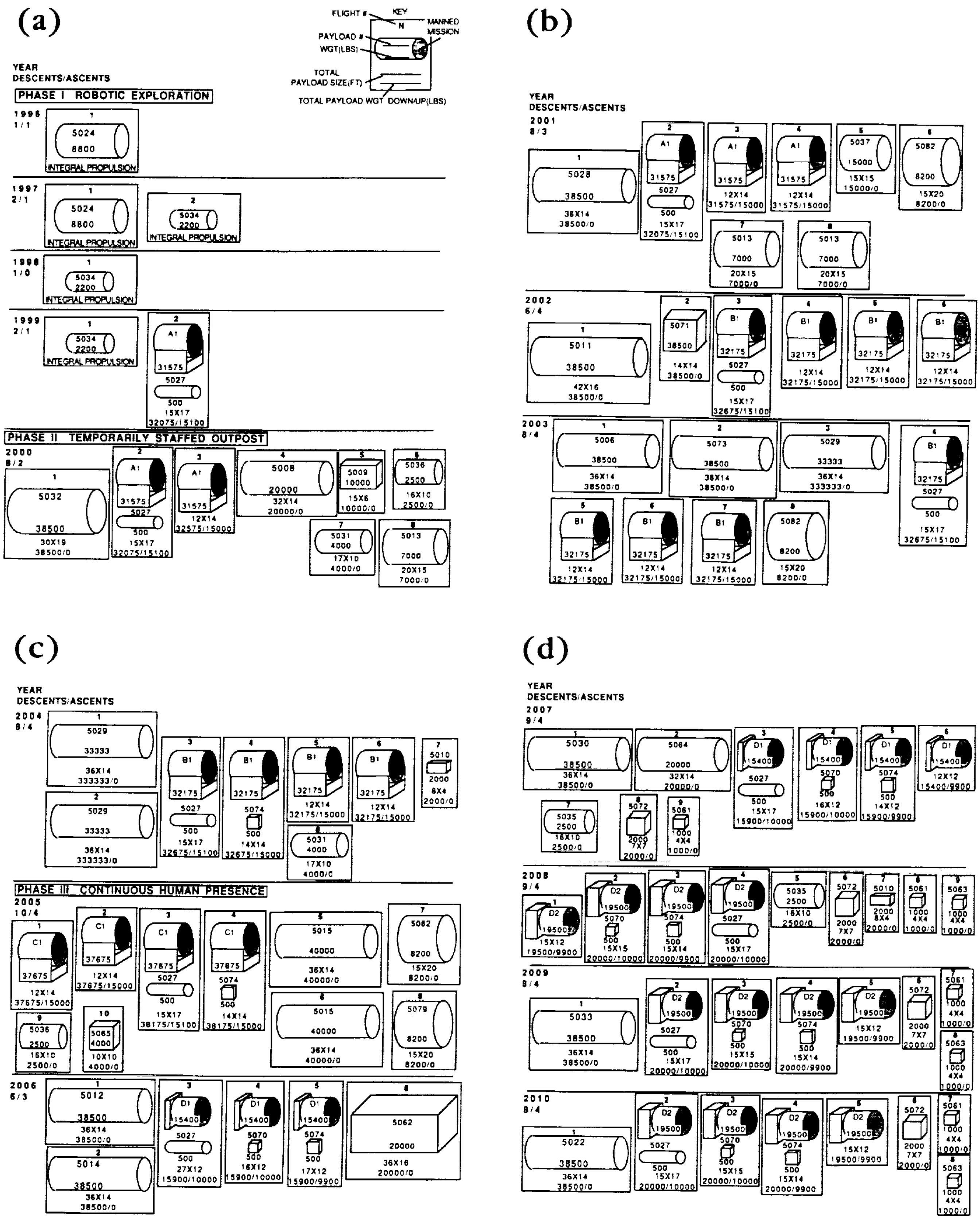

Fig. 8. CNDB lunar payloads: option 3. **(a)** 1996-2000; **(b)** 2001-2003; **(c)** 2004-2006; and **(d)** 2007-2010. Special payload combinations as shown in Fig. 4.

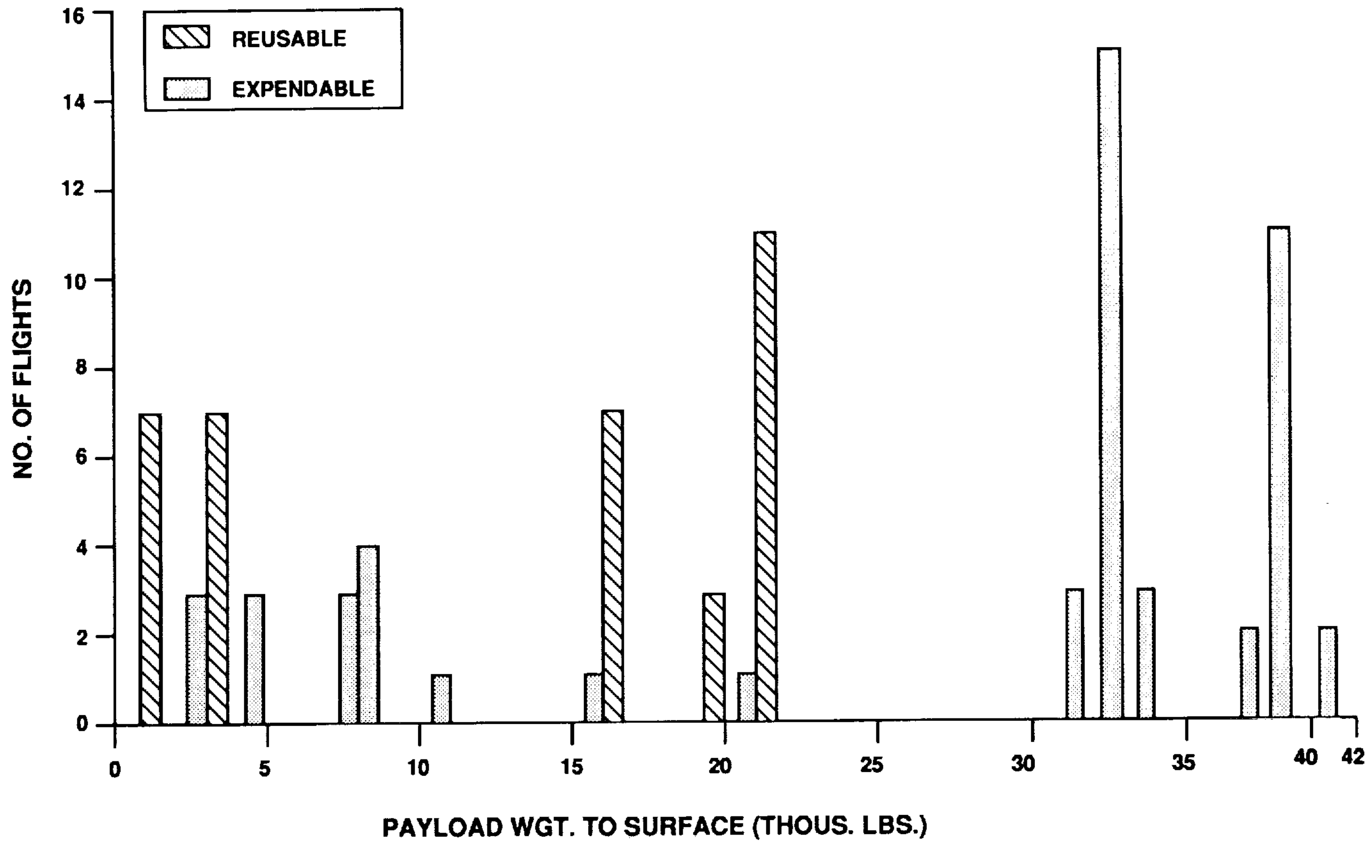

Fig. 9. Number of lunar lander flights required for various payload weights in option 3.

ascent (*Laidlaw,* 1964, pp. 16-17). This determines the maximum thrust required for the lander. In addition to providing this maximum thrust, the lander engines must be throttlable for two reasons. The first is due to the assumed configuration of the stage. In determining the payload sizes for the previous options, it was assumed that the payloads would be arranged on a rectangular platform. Four engines would be placed at the corners of the platform. This configuration would have limited engine-out capability if the remaining three engines and the attitude control system could maintain proper vehicle balance. Since it would not always be possible to balance the payloads around the center of the platform, the engine thrusts would have to be biased to compensate for any center-of-gravity offsets. Figure 10 shows how much the engines would have to be throttled to compensate for center-of-gravity offsets. In addition to this throttling requirement, the stage must have an overall throttle ratio of 21:1 during the descent trajectory (*Martin Marietta,* 1987, p. 49).

The ΔVs chosen for this paper were 2195 M/S for descent (*Martin Marietta,* 1987, p. 133) and 1920 M/S for ascent (*Eagle Engineering,* 1984, p. 24). The initial calculations for stage propellant requirements were based on these ΔVs and the following scaling equation (*Eagle Engineering,* 1984, p. 25)

$$W_I = 5024 + 0.04545\ W_P$$

where W_I = stage inert weight (lb) and W_P = propellant weight (lb). This scaling equation was substituted into the rocket equation to give the following equation for propellant weight as a function of ΔV and payload weight

$$W_P = \frac{(E\text{-}1)\,(5024 + W_{PI})}{0.04545\,(1-E) + 1}$$

where

$$W_{pl} = \text{payload weight (lb)}$$

and

$$E = e^{\left(\frac{\Delta V}{gIsp}\right)}$$

The initial propellant requirements were used to size the propellant tanks for the lander stages.

The propulsion system requirements for each vehicle and option are summarized in Table 2. The propulsion system was based on the use of RL10 engines. The required average thrust per engine to achieve an initial thrust-to-weight ratio of 0.6 is shown in this table. The actual thrust per engine would vary depending on the requirement to compensate for a center-of-gravity offset. The propellant tanks were sized so they would fit within the same area as the payloads. The arrangement of the engines and tank for the landers and ascent vehicles is shown in Figs. 11a,b. The middle portion of the payload platform on expendable landers was left open to allow room for the ascent vehicle as shown in Fig. 11c.

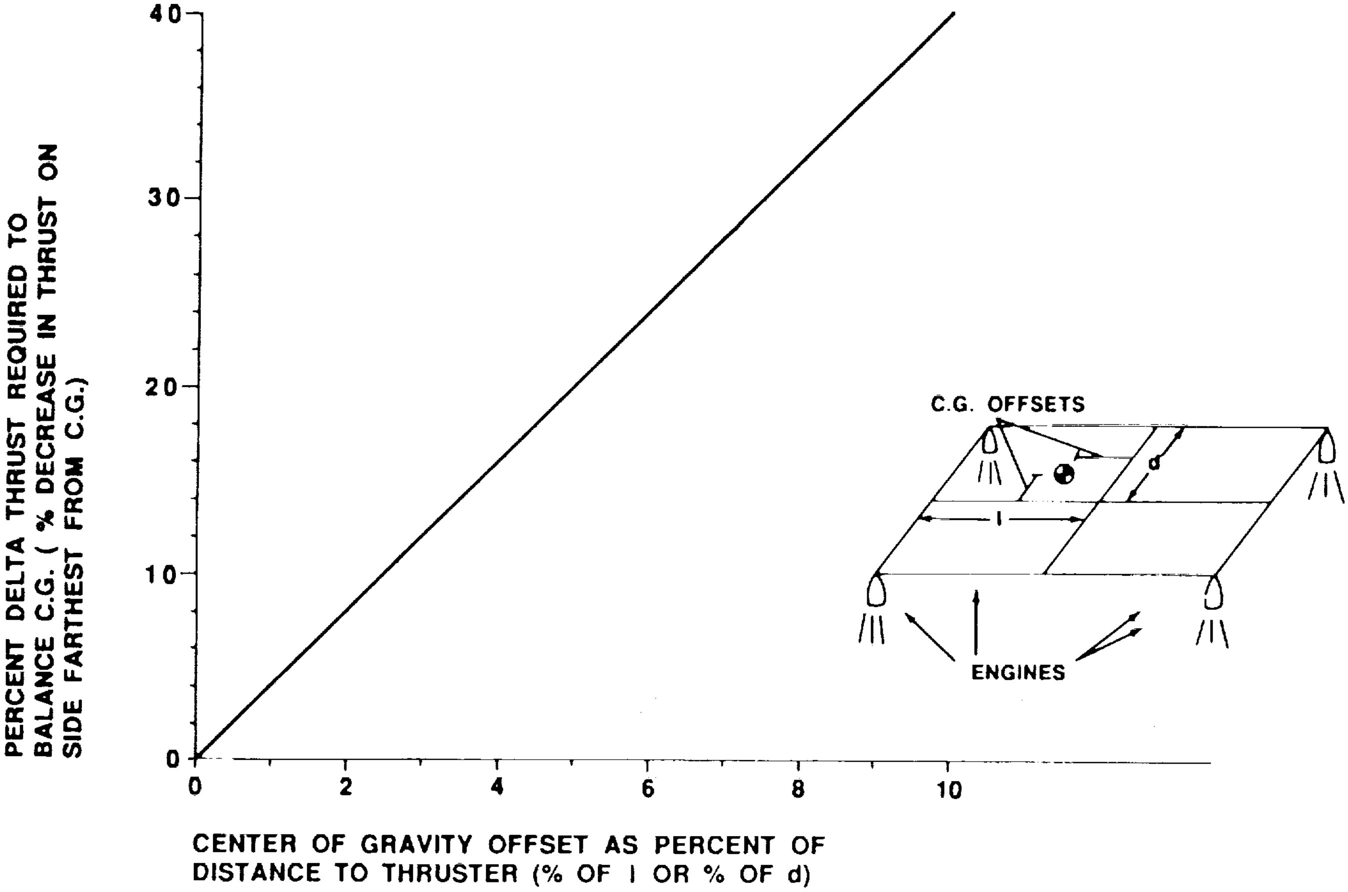

Fig. 10. Throttleability required to balance center-of-gravity offsets.

TABLE 2. Lunar lander vehicle propulsion system summary.

	Option 1, 2, & 3	Option 1		Option 2		Option 3			
	Expendable Ascent Veh.	Expendable Lander	Reusable Lander	Expendable Lander	Reusable Lander	Expendable Lander 1	Expendable Lander 2	Reusable Lander 1	Reusable Lander 2
Engine Type	RL10-IIB	RL10-IIB	RL10-IIB	RL10-IIB	RL10-IIB	RL10-IIB	RL10-III	RL10-IIB	RL10-III
Number of Engines	2	4	4	4	4	4	4	4	2
Thrust per Engine (lb)	9233	12679	10524	23286	21702	12530	5430	10061	4986
LOX Tank Size (ea.)	5.1 × 5.1	7.9 × 6	7.9 × 6.7	22.3 × 4.6	21.8 × 4.7	11.2 × 5	7.9 × 4	7.1 ×	7.1
LH_2 Tank Size (ea.)	9.5 × 6.3	17.8 × 7	13.6 × 8.8	29.8 × 7.2	27.9 × 7.5	19.5 × 6.7	14.9 × 5.1	16.7 × 7.4	10.7 × 5.3
LOX Tank Weight (lb ea.)	79	246	246	596	600	275	116	220	68
LH_2 Tank Weight (lb ea.)	276	800	785	1521	1532	805	357	799	248
Total Tank Weight (lb)	710	2092	2062	4234	4264	2160	946	2038	632
Total Engine Weight (lb)	784	1568	1568	1568	1568	1568	1504	1568	752
Propulsion System Weight (lb)	1494	3660	3630	5803	5832	3728	2450	3606	1384

All stages have two pairs of tanks.

Engines:	RL10-IIB:	RL10-III:
	Isp = 460 (sec)	Isp = 462 (sec)
	Thrust = 15,000 (lb)	Thrust = 7500 (lb)
	Mixture Ratio = 6:1	Mixture Ratio = 6:1
	Weight = 392 (lb)	Weight = 376 (lb)
	Size = 5 × 6 × 6 (ft)	Size = 5 × 5 × 5 (ft)

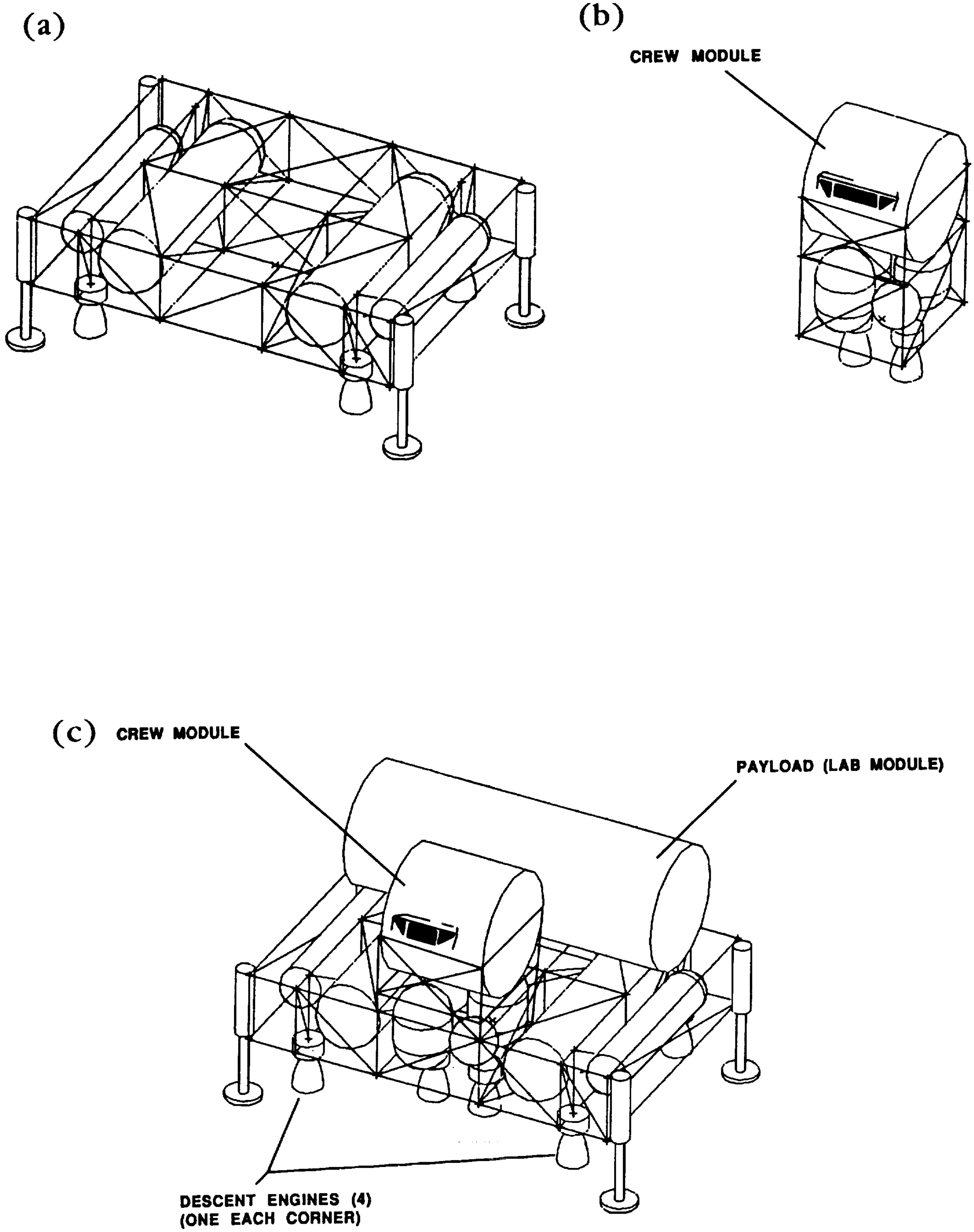

Fig. 11. Lander stage configuration. **(a)** descent stage (expendable or reusable); **(b)** ascent stage (expendable); and **(c)** descent stage with ascent stage plus payload.

The payload size requirement for each vehicle determined the length and width of the payload platform. The diameter of the propellant tanks determined the thickness of the platform. These dimensions for each vehicle and option are listed in Table 3. This table shows the weight characteristics of all the lunar lander stages. The structure weights were calculated using the platform dimensions and weights of the payload, propellant, and propulsion system. An aluminum truss configuration was assumed. The remaining system weights for environmental control, orientation control, avionics, and landing legs were calculated based on previous studies of lunar and martian landers. The final values for propellant weights were calculated using the stage weights listed in Table 3 and the ΔVs selected earlier.

The weight data listed in Table 3 were used to develop a scaling equation for the lunar lander vehicles based on payload weights and platform size. The following equation was derived

$$W_o = 7631 + 1.7972\,W_{pl_{down}} + 1.5682\,W_{pl_{up}} + 3.786\,A$$

(Note: Add 9% for reusable landers) where W_o = vehicle gross weight (lb) (Stage + Propellant + Payload);

$W_{pl_{down}}$ = descent payload (lb);

$W_{pl_{up}}$ = ascent payload (lb);

and A = platform area (ft^2).

This equation is applicable for payload weights over 15,000 lb. Notice that the propellant weight does not appear explicitly in this equation as it usually does in scaling equations. The reason for this is that this scaling equation was derived for the specific descent and ascent trajectories (ΔVs) described earlier. It is not applicable to any other trajectory. The values of vehicle gross weight obtained using this scaling equation are within 2% of the values obtained using the rocket equation if the stage inert weight is known.

The selection of a lunar lander design depends on the criteria used to judge the many design and program options. One criteron that would be used to evaluate these options is vehicle cost. Representative vehicle costs for each lander stage and option are listed in Table 4. These costs include design, development, testing, and engineering cost (DDT&E), production costs, and operations costs. The DDT&E values shown were derived using Apollo lunar lander, space station, and other cost models. The production costs are based on the first unit cost and a 90% learning curve applied to the additional vehicles. The operations costs include propellants, console time (JSC operations), tracking and communication charges, and other operations costs. The total life cycle cost is the sum of the DDT&E, production, and operations costs. These costs represent only the cost of the lander stages and their operation between lunar orbit and the surface. They do not include any cost for transportation of either the stages or the payloads to low Earth orbit or lunar orbit. Therefore, these costs are but a fraction of the overall mission and program costs and represent just one of many criteria that would be considered in evaluating lunar lander design options.

A number of important requirements for the lunar lander stages have been identified for each vehicle and payload delivery option. These requirements indicate how much the design of the lunar landers would be influenced by the mission scenario, payloads, and the type of lunar program within which they would operate. The design concepts used in determining the lander requirements pointed out in this paper are applicable to any other mission or program scenario that may be developed in the future. It is hoped

TABLE 3. Lunar lander vehicle weight summary.

	Option 1, 2, & 3	Option 1		Option 2		Option 3			
	Expendable Ascent Veh.	Expendable Lander	Reusable Lander	Expendable Lander	Reusable Lander	Expendable Lander 1	Expendable Lander 2	Reusable Lander 1	Reusable Lander 2
Platform Size (ft)	12 × 14 × 10	42 × 20 × 7	31 × 14 × 9	42 × 30 × 8	36 × 28 × 8	36 × 19 × 7	30 × 15 × 6	36 × 17 × 8	21 × 11 × 6
Payload Weight (lb)	15,100	40,375	<23,000	77,675	<58,000	41,000	16,000	<21,000	<3,000
			>10,000		>10,000			>10,000	>0
Structure Weight (lb)	2,027	4,530	3,491	6,894	6,014	4,281	2,578	3,612	1,628
LOX Tanks (lb)	158	492	492	1,192	1,200	550	232	440	136
LH_2 Tanks (lb)	552	1,600	1,570	3,042	3,064	1,610	714	1,598	496
Engines (lb)	784	1,568	1,568	1,568	1,568	1,568	1,504	1,568	752
Environmental Control (lb)	137	137	203	137	203	137	137	203	203
Orientation Control (lb)	187	187	265	187	265	187	187	265	265
Avionics (lb)	510	510	754	510	754	510	510	754	754
Landing Legs (lb)	0	1,040	647	1,910	1,495	1,027	445	608	150
15% Contingency (lb)	653	1,510	1,348	2,316	2,184	1,480	946	1,357	658
Total Stage Weight (lb)	5,008	11,574	10,338	17,756	16,747	11,350	7,253	10,405	5,042
Propellant Weight (lb)	10,668	32,560	37,820	59,811	69,931	32,183	13,947	36,667	9,079
Vehicle Gross Weight (lb)	30,776	84,508	70,158	155,242	144,679	83,533	36,200	67,071	16,620
Number of Vehicles	22	44	1	22	1	39	15	1	1
Number of Flights	22	44	19	22	19	39	15	21	14
Descents/Ascents per Option		63/41		41/41		89/57			
Total Number of Vehicles		67		45		78			

TABLE 4. Lunar lander vehicle cost summary (1988 dollars in millions).

	Option 1, 2, & 3	Option 1		Option 2		Option 3			
	Expendable Ascent Veh.	Expendable Lander	Reusabie Lander	Expendable Lander	Reusable Lander	Expendable Lander 1	Expendable Lander 2	Reusable Lander 1	Reusable Lander 2
DDT&E									
Stage	2723	4,191	5886	5,331	7743	4,144	3177	5908	4117
Engine	493	493	740	483	725	493	493	740	725
Total	3216	4,684	6626	5,814	8468	4,637	3670	6648	4842
First Unit Cost									
Stage	190	347	396	486	581	342	236	398	240
Engine	7	13	16	13	8	13	13	16	8
Total	197	360	412	499	589	355	249	414	248
Total Production Cost of Vehicles	3127	10,374	412	7,921	589	9,224	2837	414	248
Total Operations Cost	220	440	190	220	190	390	150	210	140
Total Life Cycle Cost	6563	15,498	7228	13,955	9247	14,251	6657	7272	5230
Total Cost of Option		29,289		29,765		39,973			

Note: These costs do not include transportation to low-Earth orbit or lunar orbit.

that the process of determining lunar lander requirements described in this paper will be useful in the future as program scenarios and payload models change.

REFERENCES

Alphin J. H., Taylor B. G., and Kirkland B. G. (1968) *LM Powered Descent Trajectory for the Apollo Lunar Landing Mission.* NASA-TM-X-69804, NASA Manned Spacecraft Center (Johnson Space Center), Houston. 10 pp.

Battelle Columbus Division (1987) *Advanced Space Transportation System Mission Analysis Final Report* for NASA Johnson Space Center. Contract No. NAS9-17356, Columbus. 278 pp.

Bellcom Inc. (1968) *LM Landing Using Steep Descent Trajectory,* TM-682015-1, NASA Contract No. NASAF-CR-95562, Washington, DC. 10 pp.

Eagle Engineering (1984) *Impact of Lunar and Planetary Missions on the Space Station Final Report.* Report 84-85D for NASA Johnson Space Center, pp. 4, 24-26. Contract No. NAS9-17176.

JSC (Johnson Space Center) (1984) *Lunar Surface Return,* Internal Report 84-00637, p. D7.

Laidlaw W. R. (1964) The characteristics of lunar missions. In *Lunar Missions and Exploration* (C. T. Leondes and R. W. Vance, eds.), pp. 16-17. Wiley, New York.

Martin Marietta (1987) *Orbital Transfer Vehicle Concept Definition and System Analysis Study Contract Extension II Final Review,* MCR-87-2600, for NASA Marshall Space Flight Center, Contract No. NAS8-36108, Denver. 441 pp.

McCauley L. A. (1986) *National Space Transportation and Support Study Civil Needs Data Base Version 1.1 Summary Report,* Vol. I, pp. i, 4-1, 4-2. Washington, D.C.

Woodcock G. R. (1985) Mission and operations modes for lunar basing. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 111-124. Lunar and Planetary Institute, Houston.

LUNAR LANDER CONCEPTUAL DESIGN

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This paper is a first look at the problems of building a lunar lander to support a small lunar surface base. A series of trade studies was performed to define the lander. The initial trades concerned choosing number of stages, payload mass, parking orbit altitude, and propellant type. Other important trades and issues included plane change capability, propellant loading and maintenance location, and reusability considerations. Given a rough baseline, the systems were then reviewed. A conceptual design was then produced. The process was carried through only one iteration. Many more iterations are needed. A transportation system using reusable, aerobraked orbital transfer vehicles (OTVs) is assumed. These OTVs are assumed to be based and maintained at a low Earth orbit (LEO) space station, optimized for transportation functions. Single- and two-stage OTV stacks are considered. The OTVs make the translunar injection (TLI), lunar orbit insertion (LOI), and trans-Earth injection (TEI) burns, as well as midcourse and perigee raise maneuvers.

INTRODUCTION

This paper summarizes work carried out under NASA contract and documented in more detail in the Lunar Lander Conceptual Design (*Eagle Engineering*, 1988). One lander, which can land 25,000 kg, one way, or take a 6000-kg crew capsule up and down is proposed. The initial idea was to build a space-maintainable, single-stage, reusable lander suitable for minimizing the transportation cost to a permanent base, and use it from the first manned mission on. Taking some penalty and perhaps expending expensive vehicles early in the program would avoid building multiple types of landers.

A single-stage lander is feasible from low lunar orbit (LLO) (less than 1000 km). The single-stage lander will be heavier (15-30%) in LLO than a two-stage vehicle. A lander capable of multiple roles, such as landing cargo one way or taking crew modules round-trip, is possible with some penalty (5-10%) over dedicated designs; however, the size of payload delivered to lunar orbit may vary by a factor of 2.

A four-engine design for a multipurpose vehicle, with total thrust in the range of 35-40,000 lbf (12,000 to 13,000 lbf per engine) and a throttling ratio in the 13:1 to 20:1 range is proposed. Initial work indicates a regeneratively cooled, pump-fed engine will be required due to difficulties with regenerative cooling over wide throttling ranges with pressure-fed systems. The engine is the single most important technical development item. Reuse and space maintainability requirements make it near or beyond the current state of the art. Study and simulation work should continue until this engine is defined well enough for long lead development to start.

The lander must be designed from the start for simplicity and ease of maintenance. Design features such as special pressurized volumes will be needed to make the vehicle maintainable in space. Space maintainability and reusability must be made a priority.

Liquid oxygen/liquid hydrogen (LOX/LH_2) propellants show the best performance, but LH_2 may be difficult to store for long periods in the lander on the surface. Earth-storable and space-storable propellants are not ruled out. Liquid hydrogen storage over a 180-day period on the lunar surface at the equator needs study. A point design of a LOX/LH_2 lander needs to be done in order to have a good inert mass data point that shows the performance gain is real.

Initial calculations indicate LLO offers the lowest low-Earth-orbit (LEO) stack mass. Low-altitude lunar orbits are unstable for long periods. The instability limit may set the parking orbit altitude.

Low-Earth-orbit basing for the lander is possible with some penalty in LEO stack mass (10-25%) over a scheme that bases the lander in LLO or expends it. The lander will require a special orbital transfer vehicle (OTV) to aerobrake it into LEO, however. Figure 1 shows a conceptual design of a LOX/LH_2 lander and a large OTV that carries it, single stage, from LEO to LLO and back.

SCALING EQUATIONS

It is difficult to accurately estimate the inert mass of the lander, which is a key issue in several of the trades. An equation was developed to scale the lander so that it matches the Apollo lunar module (LM) at one point, and accounts for different payloads and propellants. The LM provides the best historical data point from which scaling equations can be formulated.

On a lunar lander some systems, such as overall structure, vary with the gross or deorbit mass (M_g). Others, such as tanks, are primarily dependent on propellant mass (M_p). Other systems,

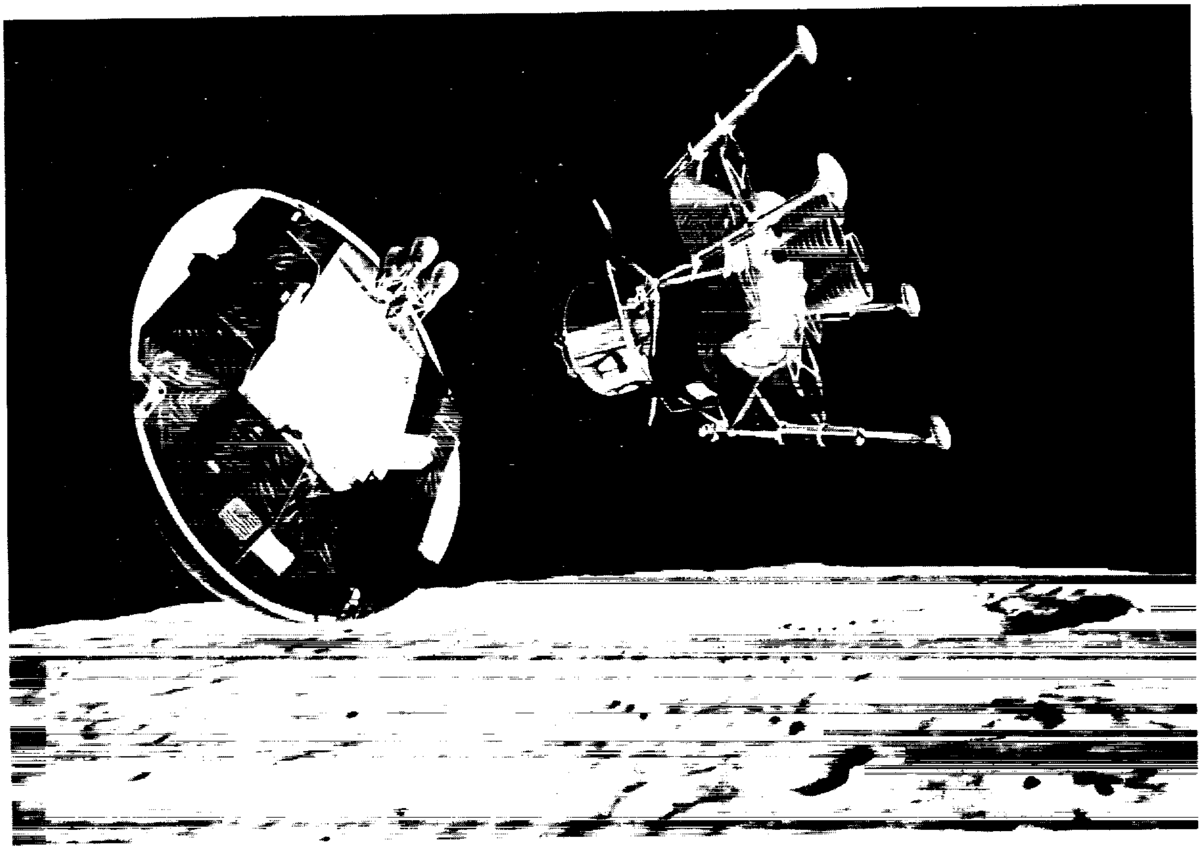

Fig. 1. OTV and lander in lunar orbit.

such as the computers, will change very little or not at all with the lander size. The inert mass (M_i), which is the sum of all of these systems, can therefore be represented using equation (1)

$$M_i = CM_g + BM_p + A \qquad (1)$$

To compare vehicles using cryogenic propellant systems with vehicles using storable propellant systems, the equation needs further modification. Due to the typically high volume associated with cryogenic propellants, it is expected that the tank systems and the thermal protection systems will be larger than for storable propellants of the same mass. Equation (1) does not take such effects into account.

One solution is to make the second term of the equation a function of the propellant bulk density (D_b). The bulk density is the total mass of propellants divided by the total volume of propellant. The tank inert mass is inversely related to the bulk density, therefore the equation should be rewritten as

$$M_i = CM_g + BM_p/D_b + A \text{ (Linear Law)} \qquad (2)$$

M_p/D_b is the total volume of propellant. This equation is a linear scaling function and assumes that those systems that are dependent on the propellant, or bulk density, are scaled linearly with propellant mass or volume.

The coefficients of the linear scaling law in equation (2) are determined by matching the masses calculated from the law with those of the Apollo LM for its various subsystems. The LM ascent stage is taken as a model payload. The coefficients of the scaling equation can be found and equation (2) becomes

$$M_i = 0.0640\ M_g + 0.0506\ (1168/D_b)\ M_p + 390\ \langle kg \rangle \qquad (3)$$

	Propellant lbm/ft^3	Bulk Density kgm/m^3	Mixture Ratio	Isp lbf-sec/lbm
N_2O_4/Aer 50	72.83	1168	1.6:1	300
N_2O_4/MMH	73.17	1170	1.9:1	330
LO_2/LH_2	22.54	361	6:1	450

TWO-STAGE VS. SINGLE-STAGE

The LM true payload was calculated to be 2068 kg. A single-stage vehicle, scaled using the above equation, transporting 2068 kg to and from the lunar surface to a 93-km circular orbit must have a gross mass in orbit, prior to descent, of 21,824 kg. When ascent and descent stages are used, applying the derived scaling equations, and assuming that the descent payload is equal to the ascent gross mass, the total gross mass of the two-stage lander prior to descent from orbit is 18,903 kg. The real LM, which is not an entirely equivalent case, had a mass of 16,285 kg.

As expected, single-stage to and from LLO results in some penalty. This penalty must be weighed against the benefits of single-stage operations, the chief one being easy reusability. Other

benefits include reduced development cost and greater simplicity. Total reusability is not practical without single-stage operation. Once lunar surface oxygen becomes available, the performance losses associated with single-stage operation will go away and single-stage operation will be the preferred mode. Single-stage operation is therefore chosen as the baseline.

SINGLE-STAGE PERFORMANCE PLOTS

Figures 2, 3, and 4 show the lander performance to and from a 93-km orbit using different propellants. The three propellants/mixture ratios/Isps as shown in the above chart are used. The Isps are chosen to be average values for a lunar ascent/descent.

The plots show three cases. In the "Cargo Down" case, the lander does not have propellant to ascend to orbit after delivering its payload. All the propellant capacity is used to deliver a large payload to the surface. The case in which the lander places a payload on the surface and has enough propellant remaining to return its inert mass to orbit is called the "Inert Returned" case. In the "Crew Module Round Trip" case a crew module is taken down to the surface and then back to orbit.

Tables 1 and 2 show performance vs. Isp as well as other variables. The cryogenic vehicle shows better performance, but not as much as expected. The low density of hydrogen drives the propellant mass multiplier up in the scaling equation (3). The equations may be biased against a pump-fed cryogenic system because they are scaled from a pressure-fed storable system.

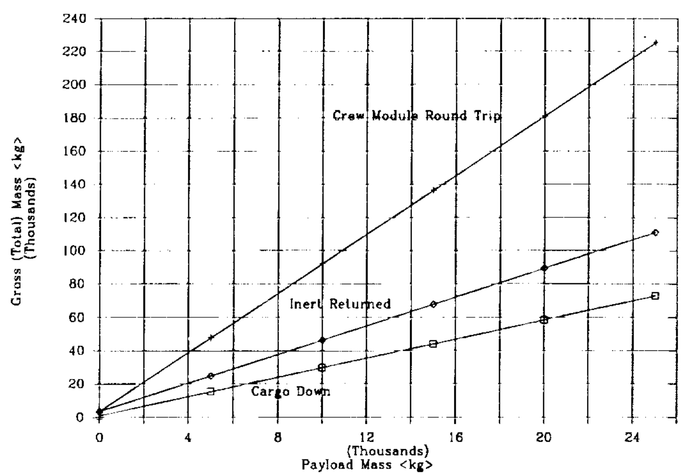

Fig. 2. Single-stage crew/cargo lander. Orbit = 93 km; MR = 1.6 N/A; Isp = 300.

PARKING ORBIT ALTITUDE

Tables 1 and 2 show how lander mass increases steadily as lunar orbital altitude goes up. Table 3 shows how LEO stack mass also goes up with lunar orbit altitude. The LEO stack mass does not rise dramatically until orbits of 1000 km or over are used. From a performance standpoint, the lowest orbits are therefore preferable. Apollo experience has indicated that very low orbits, on the order of 100 km, may be unstable over periods of months. The best altitude will therefore be the lowest altitude that is stable for the period required.

Ascent to a 93-km lunar orbit is assumed to be 1.85 km/sec. Descent from a 93-km lunar orbit is assumed to be 2.10 km/sec. These values were back-calculated from the Apollo 17 weight statement in order to match design theoretical values. They closely match postmission reported Apollo 11 ΔVs of 2.14 and 1.85 km/sec (*Apollo 11 Mission Report,* 1969). Ascent/descent to or from higher lunar orbits assumed a Hohmann transfer.

PLANE CHANGE CAPABILITY

One-time plane changes on the order of 15° in low lunar circular orbit can be built in for modest lander mass increases on the order of 10% for LOX/LH_2 landers. This will also result in a LEO stack mass increase of at least 10%. The plane change ΔV and vehicle mass increase does not vary much with lunar orbit altitudes below 1000 km for a given angle of plane change;

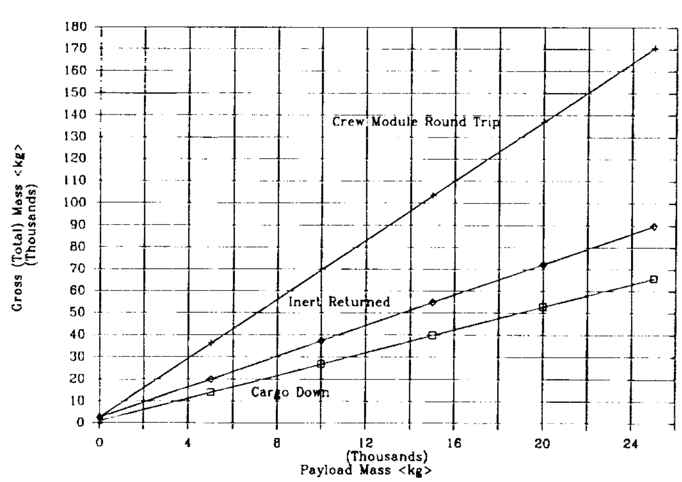

Fig. 3. Single-stage crew/cargo lander. Orbit = 93 km; MR = 1.9 N/M; Isp = 330.

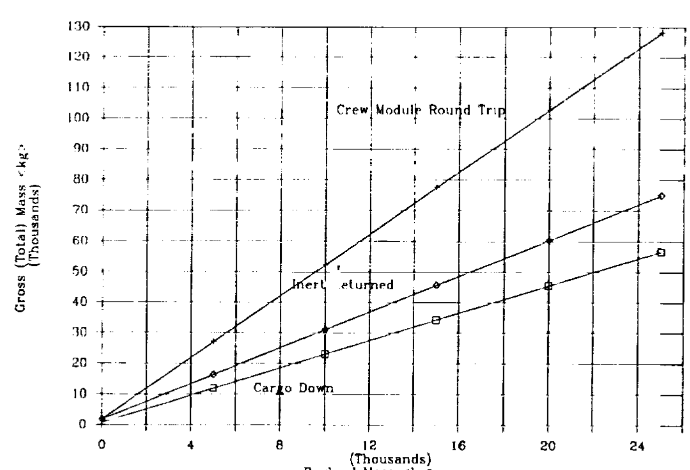

Fig. 4. Single-stage crew/cargo lander. Orbit = 93 km; MR = 6.0 O/H; Isp = 450.

TABLE 1. Lander mass vs. altitude, 6000-kg crew module round trip.

Circ. Orbit Altitude (km)	Isp = 450 sec			Isp = 330 sec		
	Deorbit Mass	Inert Mass	Propellant Mass	Deorbit Mass	Inert Mass	Propellant Mass
93	32	6	20	43	5	32
200	34	6	22	46	5	35
400	37	7	24	50	6	38
1000	46	9	31	66	7	53
L2 (M-LP-E)	166	13	147	344	38	300

TABLE 2. Lander mass vs. altitude, 25,000-kg cargo down case.

Circ. Orbit Altitude (km)	Isp = 450 sec			Isp = 330 sec		
	Deorbit Mass	Inert Mass	Propellant Mass	Deorbit Mass	Inert Mass	Propellant Mass
93	57	8	24	66	6	35
200	58	8	25	68	7	36
400	60	8	27	70	7	38
1000	64	9	30	76	7	44
L2 (M-LP-E)	84	13	46	100	11	64

TABLE 3. LEO stack mass as a function of lunar orbit altitude.

LLO Altitude (km)	Lander Deorbit Mass	LEO Stack Mass			
		One-stage OTV		Two-stage OTV	
		Load lander propellants in			
		LLO	LEO	LLO	LEO
6000-kg crew capsule round trip, LLO-LS-LLO, 450-sec Isp Lander					
93	32	111	136	101	127
200	34	120	142	107	133
400	37	121	150	112	140
1,000	46	142	174	131	165
36,000 (L2)	170	500	535	471	506
25,000-kg cargo one way, 450-sec Isp expended lander					
93	57	190	190	174	174
200	58	192	192	176	176
400	60	195	195	180	180
1,000	64	202	202	187	187
36,000 (L2)	84	268	268	246	246
6000-kg crew capsule round trip, LLO-LS-LLO, 330-sec Isp lander					
93	44	148	169	137	159
200	46	155	172	144	162
400	50	162	184	152	173
1,000	66	205	226	191	214
36,000 (L2)	344	963	1,115	904	1,039
25,000-kg cargo one way, 330-sec Isp expended lander					
93	66	217	217	199	199
200	68	221	221	204	204
400	70	229	229	208	208
1,000	75	238	238	219	219
36,000 (L2)	100	314	314	290	290

All masses are metric tons.
All OTVs are LOX/LH_2, 455-sec Isp.
Space station orbit altitude - 450 km.
Delta Vs as given in Table 4.
All LEO-LLO trajectories are 75-hr transfers.
No plane changes are accounted for.
OTVs are "rubber" and optimized to the given payload.
OTVs assume: 15% of entry mass is aerobrake; 5% of propellant is tankage, etc.; 2.3% of propellant is FPR and unusables.
Other OTV inerts = 2.5 m tons for two-stage, 4.5 m tons, for one-stage.

however, as the orbit altitude increases above 1000 km, plane change ΔV goes down drastically, but the lander mass goes up drastically due to increased ascent and descent ΔV (Table 4).

The ability to change planes widens the launch window the vehicle has to reach high-inclination lunar orbit. For a landing site such as Lacus Verus at 13°S latitude, it might allow a lander to ascend to an OTV or LLO space station in lunar equatorial orbit at any time. This is a highly desired feature. For a high-latitude base and parking orbit, polar for instance, a 15° plane change capability would allow launch on roughly 4.5 days out of 27 days in a lunar month.

TABLE 4. Delta Vs.

Lunar Orbit	TLI	LOI/TEI*	Total
93	3.101	0.846	3.947
200	3.101	0.832	3.933
400	3.102	0.809	3.910
1,000	3.102	0.759	3.861
35,000 (L2, M-LP-E)	3.084	0.863	3.947

* LOI and TEI are assumed to be the same.

PROPELLANT LOADING LOCATIONS

There are several options for lander propellant loading locations. In addition to propellant loading, the lander must be serviced with other consumables, maintained, and periodically tested. Two straightforward options include (1) returning the lander to the space station after each mission to the surface and servicing and loading it with propellants at the space station or (2) loading the lander with propellants in lunar orbit and servicing and maintaining it in lunar orbit.

The concept of maintenance and propellant transfer in space is new. The space station will already have propellant loading, maintenance, and refurbishment facilities for the OTVs. The space station will have the largest stock of spares, most personnel, shortest logistics tail, etc. Maintenance man-hours in space will cost least at the space station. Development cost will be reduced in that facilities required for the OTVs can be designed to service the landers as well.

Bringing the lander back requires a larger stack in LEO. Table 3 illustrates this. Given the OTV transportation system described, bringing the lander back can cost as much as 25% more LEO mass in one mission than loading propellants in lunar orbit. Loading propellants in lunar orbit will also have costs however. The lander will be left in a given orbit that the next mission must fly too. Some performance loss or loss in mission flexibility will be associated with this. If a facility is required in lunar orbit to handle propellant transfer, then the flights needed to place and support this facility represent a performance loss on the system.

It is difficult to integrate the lander with an aerobrake. An OTV specially configured to carry the lander will be required, or the lander will require its own aerobrake and will be an independent vehicle on return to Earth.

If it is practical to design a lander that can be loaded with propellants and other consumables and be maintained and checked out in lunar orbit without a fixed facility (a small lunar orbit space station), then this is a more attractive option. There is debate about the practicality of basing a reusable vehicle at the space station however. The further away from Earth a vehicle is based, the more expensive and difficult maintenance, repair, and testing will become. Other performance losses would be associated with operation from a fixed orbit. These losses will go up as inclination of the lunar orbit goes up. If the base is equatorial, this will not be a problem.

MAIN ENGINES

Table 5 shows various thrusts and throttling ratios estimated to be required in different circumstances. The deorbit cases assume an acceleration of 9 ft/sec^2 or 2.74 m/sec^2 is required at the start of the burn. The ascent case assumes an acceleration of 6 ft/sec^2 or 1.83 m/sec^2 is required. The hover case assumes 40% of the lunar weight is the minimum hover thrust. All these assumptions match Apollo numbers. New trajectories need to be run with these vehicles to see how these numbers can be varied.

The widest range is between deorbiting a 25,000-kg payload from a higher low orbit with a low-performance propellant (43,000 lbf required) and hovering a crew capsule and the vehicle inert mass just before running out of propellant as might occur in an abort to the surface or a normal landing requiring propellant loading on the surface (1760 lbf). The ratio between these two cases is roughly 24:1. The Apollo LM engine was designed with a 10:1 throttling ratio. If the minimum thrust case is taken as a normal landing for an H_2/O_2 lander with a crew capsule (2957 lbf), the throttling ratio becomes 13:1. Table 5 shows a variety of cases and how the throttling ratio might vary.

TABLE 5. Comparison of throttling ratios.

Max. Thrust (lbf) Orbit Alt., Isp, Prop. Situation	Min. Thrust (lbf) Situation	Throttling Ratio
37,00 400 km/450 sec/O_2/H_2 Deorbit with 25,000-kg cargo	1760 40% of hover, near empty with crew capsule only, abort to surface.	21:1
35,665 93 km/450 sec/O_2/H_2 Deorbit with 25,000-kg cargo	1760 40% of hover, near empty with crew capsule only, abort to surface.	20:1
37,000 400 km/450 sec/O_2/H_2 Deorbit with 25,000-kg cargo	2957 93 km, 450-sec Isp 40% of hover before normal landing, 6000-kg capsule	13:1
19,731 93 km/450 sec Deorbit with 6000-kg crew capsule	2957 93 km, 450-sec Isp 40% of hover before normal landing, 6000-kg capsule	7:1
19,731 93 km/450 sec Deorbit with 6000-kg crew capsule	1760 40% of hover, near empty with crew capsule only, abort to the surface	11:1
35,665 93 km/450 sec/O_2/H_2 Deorbit with 25,000-kg cargo	4693 93 km, 450-sec Isp, O_2/H_2 40% of hover, near empty with 25,000-kg cargo	8:1
43,000 400 km/330 sec Deorbit with 25,000-kg cargo	1760 40% of hover, near empty with crew capsule only, abort to the surface	24:1

Reducing the required throttling ratio may have significant advantages. The single, pressure-fed Apollo LM engine was cooled by ablation of the nozzle. A reusable engine must be regeneratively cooled. Pressure-fed regenerative cooling over a wide throttling ratio may not be possible due to the flow changing a great deal. This leads to a pump-fed engine, a much more complicated device, which then leads to two or more engines for redundancy. A single-purpose lander, to land only a crew, might function with a pressure-fed single engine. Table 5 indicates a throttling ratio of 7 or 8 to 1 might be enough if one lander were not required to bring down the 25,000-kg cargo and the crew capsule as well. The table indicates that a dedicated cargo lander and a dedicated crew lander would each require a throttling ratio of 7 or 8 to 1. The crew lander might use one or two engines and the cargo lander four. Other schemes involving shutting off or not using engines are also possible, but result in inert mass penalties. Another option would be to reduce the lander deorbit acceleration. The penalties for doing this should be determined.

On the other hand, pump-fed, cryogenic engines may be able to function well in the 20:1 throttling ratio regime as some individuals have claimed. Less work has been done on storable engines with wide throttling ratios. The pump-fed engine may be required even at low throttling ratios because of cooling problems. The relationship between throttling ratio and engine cooling needs to be determined. In particular, the highest throttling ratio, pressure-fed, regeneratively cooled engine, that will work, must be determined. If it is below 7 or 8, pressure-fed engines can be eliminated as candidates.

Another possibility is a partially ablative engine. The combustion chamber and throat could be regeneratively cooled and the majority of the nozzle could be ablative, designed for easy replacement every few missions, which might allow a pressure-fed system to be used.

The Adaptable Space Propulsion System (ASPS) studies and the OTV studies have narrowed the propellants to N_2O_4/MMH and O_2/H_2, respectively, using pump-fed engine cycles. Some of the technology efforts for the ASPS and OTV engines are underway and more are planned. The lunar lander propulsion system can benefit from this technology to a great extent. However, a propulsion system designed especially for the lunar lander should also be studied and compared to determine the technical penalties of using the ASPS/OTV technology engines vs. the cost and time penalties of developing another engine. Additional technology requirements resulting from the lunar lander studies could be added to the ASPS/OTV engine technology programs. This would decrease cost and development time for the lunar lander engine program.

PROPELLANTS

There are many propellant combinations to consider for the lunar lander study. For initial vehicle sizing the Earth-storable combination N_2O_4/MMH and the cryogenic combination O_2/H_2 are selected (see Table 6). These propellant combinations are being studied for other space propulsion systems and experience has been gained by their use on operational spacecraft and booster vehicles. All the previous tables and figures can be used to compare the performance of these two propellants. In general, the O_2/H_2 lander and LEO stack is 10-30% lighter. The OTVs are all assumed to be O_2/H_2. More study of the inert mass is needed to better qualify this difference, however. A point design of an O_2/H_2 lander is needed to get good inert weights.

TABLE 6. Engine characteristics to be used for initial vehicle sizing.

	O_2/H_2	N_2O_4/MMH
Thrust (lbf)	12,334	
Chamber Pressure (psia)	1,270	
Mixture Ratio (O/F)	6.0	1.9
Max Isp (sec)	460	340
Ave. 14:1 Isp (sec)	450	330
Nozzle Area Ratio	620	
Nozzle Exit Diameter (in)	60	
Engine Length (in)	115	
Weight (lb)	525	

There are other propellant combinations to be investigated such as O_2/C_3H_8 and O_2/C_2H_4, which have higher performance than N_2O_4/MMH; however, the propellant bulk densities are lower. The combinations should be reviewed when the thrust chamber cooling requirements and performance are investigated for high throttling ratios. These propellants could take advantage of surface-produced oxygen at some point in the future without the problems of long-term hydrogen storage.

Pressure-fed propulsion systems with the Earth-storable propellant combination N_2O_4/Aer50 were used for the Apollo spacecraft propulsion systems for simplicity and reliability. The Apollo descent-stage thrust chamber (nonreusable) was ablatively cooled while the lunar lander thrust chamber (reusable) requires regenerative cooling. The estimated throttling for the lunar lander cannot be achieved with a pressure-fed system using a regeneratively cooled chamber and reasonable tank and system weights. Therefore, the lunar lander will be pump-fed unless some innovative method for thrust chamber cooling is discovered that would then allow a pressure-fed vs. pump-fed comparison.

Achieving the required throttling and cooling with an Earth-storable propellant, pump-fed propulsion system will also be difficult and could prove unfeasible. The system would become too complex if two engine designs (different maximum thrust levels) and shutdown of engines became necessary to attain the overall thrust variation.

NUMBER OF ENGINES

The complexity of a pump-fed engine requires at least two engines for a manned space vehicle so that one engine failure will not result in loss of crew. Vehicle control system requirements and effective Isp must be considered in selecting the number of engines, i.e., thrust vector control and loss of Isp due to nonparallel engines if an engine fails.

Four engines have been tentatively selected for the initial study. The engine size is smaller than a two- or three-engine configuration and the throttling ratio is lower. The maximum thrust required for the O_2/H_2 lunar lander configuration is assumed to be 37,000 lb (see Table 5). For manned missions, if one engine fails during lunar descent the mission will be aborted to lunar orbit since redundancy would be lost for lunar launch. Thrust would be adequate with two of the four engines operating, but thrust vector control would be a problem. For unmanned missions, if one engine fails during lunar descent, the mission will be continued to lunar landing since there is no problem with loss of crew, and at some point in the descent insufficient propellant will be available to abort to lunar orbit. With these ground rules, the selected maximum thrust level for each of the four engines is 12,334 lb. This results in a total maximum thrust of 37,000 lb in the event one engine fails during the unmanned lunar descent,

and the lunar lander still has the capability to land, where a normal landing determines minimum thrust on the lunar surface as planned. The throttling ratio required per engine is 13.4:1. An ascent/descent simulation with aborts is needed to refine these numbers.

Another approach to obtain pump-fed engine redundancy is the use of a single thrust chamber with two sets of turbopumps and associated controls. This would result in a single thrust chamber of 37,000-lb thrust with a slight gain in performance (higher area ratio) and a simplification of the thrust vector control. Relying on a single, reusable, regeneratively cooled thrust chamber with the associated deterioration as missions are added would be one reason to reject this approach. An extremely critical inspection of this chamber would be required between missions if this engine system were selected.

The performance figures for N_2O_4/MMH are satisfactory for preliminary vehicle sizing. Further information on engine cooling is required before additional engine characteristics can be determined. The use of a single, 37,000-lb-thrust, pump-fed engine should be investigated since a large engine results in lower thrust chamber cooling requirements. This investigation should include the use of both propellants for thrust chamber cooling, the integration of redundant turbopump operation, and the possible requirement of a variable-area injector as used on the Apollo descent engine to improve performance throughout the throttling range.

The present technology goal for the OTV engine is an operational life of 500 starts/20-hr burn time, and a service-free life to 100 starts/4-hr burn time. Based on the Apollo LM burn times this would allow approximately 58 operational missions and 11 service-free missions. This is a goal. The space shuttle main engine (SSME) requires reservicing every mission and is effectively replaced, on average, every three missions.

REACTION CONTROL SYSTEM (RCS)

The RCS propellants for the O_2/H_2 lunar lander are proposed to be also O_2/H_2 and are loaded into the main propellant tanks. Liquid propellants are extracted from the main tanks, pumped to a higher pressure, gasified by passing through a heat exchanger, and then stored in accumulator tanks as gases to be used in gas/gas RCS thrust chambers. The gas generators to operate the turbopumps use gaseous oxygen/gaseous hydrogen and the exhaust gases are passed through the heat exchanger to gasify the LOX and LH_2 as mentioned previously. Sixteen thrusters are located in four clusters 90° apart, four engines per cluster, to supply the required control and translation thrust. The thrust of each RCS engine is approximately 100 to 150 lb depending upon vehicle requirements. The Isp is 370 sec, steady state.

The RCS propellants for the Earth-storable lunar lander are the same as for the main engine, N_2O_4/MMH with separate RCS propellant storage tanks and pressurization system. The engines are pressure fed and the Isp is about 280 sec, steady state.

Integrating the N_2O_4/MMH main propulsion system and the RCS resulting in smaller RCS tanks and the elimination of the RCS pressurization system is a possibility and warrants investigation.

SUPPORTABILITY

Support of the lander for an extended period of time will require a different approach to all the supportability disciplines than those that have been used for NASA manned spaceflight programs through the space shuttle era. A new approach to reusability, maintenance, and repairability considerations is needed.

Technology available in the early 1990s can, in most cases, produce sufficiently reliable hardware and software to support the lunar lander scenario if proper management emphasis is given to it. The space environment is, in many ways, quite benign and conducive to long life and high reliability.

Past NASA manned space programs, most notably Apollo and space shuttle, have been initiated with the intent of providing in-flight maintenance capability; however, these requirements were either deleted from the program or not pursued with sufficient rigor and dedication to provide meaningful results. It will be necessary for the supportability requirements to be given continuous high priority throughout the life cycle of the lander if it is to achieve the current goals of space basing and long useful life.

If true reusability with acceptable reliability is to be achieved, these considerations must be given high priority from program initiation onward. The current manned spacecraft redundancy requirements will, in general, provide sufficient reliability for the lander. To achieve high reliability it will be desirable to use proven technology in as many of the vehicle systems as possible and still meet the performance requirements. If the lunar lander is adequately maintained and repaired then the reusability goal can be met. The major exception may well occur in the main propulsion system inasmuch as high-performance rocket engines with life expectancies of the order needed to satisfy the lander design requirements are not available.

Designing to achieve efficient space-based maintenance will give rise to new problems and require unique approaches to keep maintenance activity to an acceptable portion of the overall manpower available. Teleoperated robotic technology is one possibility. Another approach, shown in the conceptual design, is a large pressurized volume on the lander that can be docked to the space station and can be designed to hold most equipment requiring maintenance, servicing, or replacement.

DATA MANAGEMENT AND GUIDANCE, NAVIGATION, AND CONTROL (GN&C)

The multipurpose lander must land with cargo unmanned as well as manned. Sophisticated automatic fault detection, identification, and reconfiguration (FDIR) will be required.

The vehicle must be designed from the onset to be entirely self-checking and rely on onboard calibration. Most of the maintainability functions specified for the space station are also applicable to the lunar lander.

In addition, the lunar lander design must be capable of autonomous launch. The Apollo program demonstrated many aspects of the capabilities needed to launch and operate a vehicle without the benefit of a costly launch check-out facility. With the advances in expert system design and the increases in onboard computer power the autonomous checkout goals should be readily achievable but require that these functions are recognized as primary requirements.

The data management system (DMS) is defined as the redundant central processing system, multipurpose displays, data bus network, and general purpose multiplexor-demultiplexors. The software system is also included. Although the DPS processors accomplish the principal function processing, processors are

implemented at the subsystem or black box level to perform data compression, FDIR functions, and other functions amenable to local processing. These local processors would be procured to be card compatible with the main processor. All items required to interface with the standard data bus are procured with a built-in data bus interface.

The DMS processor recommended is a 32-bit machine derived from a commercial chip to capitalize on the advantages of off-the-shelf software, support tools, and the many other advantages that accrue from having a readily available ground version of the onboard machine. For the purpose of this conceptual design a version of the Intel 80386 microprocessor was assumed.

Two multipurpose displays are proposed using flat screen plasma technology. The operations management software supports the monitoring of onboard consumables, system configurations, and failure status, and displays this information for the benefit of space station checkout crews or, when applicable, to the lunar lander crew members. The display system also supports the flight displays for mission phases when manual control is available.

The IMU proposed is a strapped down system based on ring-laser gyro technology. This approach is chosen because of advantages in cost, ruggedness, stability, and ease of integration with optical alignment devices. Projected advances over the next few years also show a clear advantage in weight and power over other types of inertial systems. The ring-laser gyro is readily adaptable to a "Hexad" configuration that provides the maximum redundancy for the least weight and power. The "Hexad" configuration contains a built-in triple redundant inertial sensor assembly (ISA) processor that does the strapdown computations, sensor calibration, redundancy management, checkout, and other local processing assignments. The ISA processor also calculates the vehicle attitude and vehicle body rates required for control system stabilization.

Alignment of the IMU will be required prior to descent and ascent to minimize errors and ΔV expenditure. This is accomplished by an automatic star scanner attached to the case of the IMU to minimize boresight errors.

Guidance functions, control equations, jet select logic, and similar processes are mechanized in the DMS processor. To the maximum extent possible, these and other critical functions will be implemented in read-only memory (ROM) to provide the maximum reliability and lowest power and weight penalties. Commands to the main engines and RCS engines are transmitted via the triple-redundant data bus to the control electronics sections where electrical voting takes place before transmittal of the command to the actual effectors.

Automatic docking of the lunar lander with the OTV is a requirement; however, the OTV is assumed to be equipped with the sensors and intelligence to accomplish this operation, and no provision is made on the lunar lander to duplicate this capability. Wherever the capability resides, it must be developed. The sensors and software to do automatic docking do not exist at this time in the free world.

A variety of systems are possible for updating the onboard inertial system and performing landing navigation. The preferred system is the cruise missile-type terrain-following radar with surface-based transponders. The basic elements of this system will all be part of the landers anyway, and depending on the surface features and the knowledge of their positions, no surface elements at all may be required. A small surface-based radar would be a low-cost addition to the onboard terrain-following system.

The first requirement for terrain-following-type navigation is knowledge of a terrain feature's location to within a certain range of error. If the first landings on the site are manned, they must occur during lighting conditions allowing good visual landing navigation. The first landers can carry a transponder and, if required, place another on the surface at a known location. Subsequent landings will then get positions relative to these transponder(s). Table 7 estimates the mass, power, and volume required for each component.

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS (ECLSS)

Comparison of open and closed systems were made to determine the crossover point where it pays to go from open loop to a partially closed loop. The crossover point is dependent on several factors: mass, volume, energy, and operational considerations. From the mass standpoint, the crossover point was approximately 60 days for the atmosphere revitalization system, and 35 days for the water management system. Neither of these two comparisons took into account the impact on other subsystems such as power and thermal control. With the identified power requirements, these impacts should be added to the ECLSS mass impacts to arrive at a reasonable mass break-even point. As a point of reference, a partially closed loop system is estimated to require on the order of 4 kW of power and have hardware masses of around 3000 kg. Open-loop systems are predicted to require 1 kW of power and have a hardware mass of 1300 kg for 15-day missions. The break-even point will be at an even longer stay time when the additional power system mass required is considered. Three- to 15-day missions are under consideration for the lander. For these reasons, the system design selected was the open-loop configuration (see Table 8).

The choice of power generation method can also bias the choice of ECLSS design selection. If fuel cells are used to generate electricity, then the process byproduct, water, can be used in the open-loop concept.

The atmosphere supply and pressurization system source consists of tanks of gaseous high-pressure nitrogen and oxygen. If fuel cells are used for electrical power, then the system would get oxygen from a common cryogenic supply tank. These sources are fed through regulators to support the cabin, crew suits, airlock, and EMU station. Provisions are available for cabin and airlock depressurization and repressurization. Equalization valves are available at each pressure volume interface. Partial pressure sensors will be connected to the regulators to maintain the proper atmosphere composition mix.

Atmosphere revitalization is supported by LiOH canisters for CO_2 removal. Odors and particulates will be removed by activated charcoal and filters. Cabin fans provide the necessary circulation of the atmosphere through the system and habitable volume. Humidity and temperature control will be handled by heat exchangers and water separators. Thermal control for other equipment in the crew compartment will be handled by cold plates and a water loop connected to the thermal control system. Included in this subsystem will be the fire detection and suppression system.

Shuttle power requirements, itemized by systems that might be comparable to lunar lander systems, were added up. The average power required based on this calculation was 1.81 kW. The shuttle is designed for a nominal crew of 7 with a contingency of 10.

TABLE 7. DMS/GN&C mass and power.

Component	Unit (Vehicle) Weight (kg)	Unit (Vehicle) Power (W)	Unit cubic ft Volume (Config.)	Number/Vehicle
DMS Processor	10 (30)	75 (225)	0.27 (0.81)	3
MDM	7.7 (46.4)	60 (360)	0.25 (1.5)	6
ANK/Display*	8.6 (17.3)	40 (80)	0.35 (0.7)	2
Hexad IMU	16 (16)	75 (75)	0.3 (0.3)	1
Star Track	2 (6.1)	10 (30)	0.1 (0.3)	3
Nav. Sensors				
Landing	13.2 (13.2)	100 (100)	0.4 (0.4)	1
Rendezvous	20.5 (20.5)	200 (200)	0.6 (0.6)	1

* ANK = alpha-numeric keyboard.

Total Weight = 149.3 kg (325.5 lb).
Total Power = 1070 W.
Total Volume = 0.13 cu m (4.61 cu ft).

TABLE 8. Open-loop ECLSS mass required.

No. of Crew	Support Time (days)	Consumables (3 airlock cycl. kg)	Hardware (kg)	Fluids (kg)	Crew Prov. (+crew mass) (kg)	Total (kg)
6	1	72	1264	214	2562	4112
4	3	133	1264	214	1708	3319
6	15	894	1264	214	2562	4934
4	15	612	1264	214	1708	3798

The lander crew module holds four with a contingency of six. The power requirement is assumed to be roughly linear with crew downsized by 4/7, resulting in a requirement for 1.0 kW average power. Increased efficiency in motor design and advanced cooling techniques occurring over the 20-30-year interval between the two vehicles is expected to result in some savings as well.

ELECTRICAL POWER

Two scenarios have been discussed with respect to the crew module. In one scenario the crew only enters the module to descend to the surface and lives in another module in-orbit. In the second scenario, the crew lives in the lander module for the complete trip, estimated to be 15 days minimum. For this reason the lunar lander mission is broken down into two scenarios for the electrical energy storage provisions: (1) Power up in lunar orbit; descent, three days on surface; ascent to lunar orbit — 144 kWhr at 2 kW average. (2) Power up in LEO one day; three days to lunar orbit; one day in lunar orbit; descent, three days on surface; ascent, one day in lunar orbit; three days to LEO; three days in LEO — 720 kWhr at 2 kW average (15 days).

The lander may stay much longer than three days on the surface, but it is assumed that external power will be provided. In either case it is assumed that the power system would be serviced at the space station in LEO.

The 2-kW average power requirement is an estimate based on the Apollo LM (peak power 2.3 kW) and calculations indicating DMS/GN&C and ECLSS will each require about a kilowatt. This may be reduced, but there will be other power requirements. A more conservative estimate might be an average power requirement of 3 kW.

Fuel cells and a number of ambient temperature batteries were compared. The shuttle-derived fuel cell yields the system of lowest weight and greatest flexibility. For large energy (>50 kWhr) requirements the fuel cell becomes the candidate of choice primarily due to the large energy content of the reactants, H_2 and O_2, supplying approximately 2200 Whr/kg (tankage not included). The reactant can be stored as a high-pressure gas, a liquid in dedicated tanks, or the main propellant tanks can be used.

There is no impact from adding the fuel cell reactants to the propellant tanks; 31 kg H_2 adds 26 mm to the diameter of each H_2 tank, an increase of 0.7% for each parameter, and 244 kg O_2 adds 6 mm to the diameter of each O_2 tank, an increase of 0.9% and 0.3% respectively for each parameter. This provides energy storage of 200% of that required for the 15-day mission. Getting the reactants out of the large tanks when only small quantities are left may be a problem, however.

The fuel cell operating temperature range is between 80° and 95°C. It is provided with a fluid loop heat exchanger that is integrated with the ECLSS thermal control loop, just as in the shuttle orbiter. Heat rejection will be approximately 4400 btu/hr at the 2-kW power level.

Fuel cell product water is portable and useful for crew consumption and evaporative cooling. It is produced at the rate of about 3/4 l/hr at the 2-kW power level for a total of 260 kg for the 15-day mission. It is delivered to the fuel cell interface in liquid form for transfer to the ECLSS system. Therefore, storage and plumbing are not included in the power system design. However, for single tank storage, a tank of 0.8 m in diameter is required.

The baseline system used in the weight statements is a dual redundant fuel cell system using dedicated tanks for cryogen storage. Table 9 estimates the total mass of the system that

TABLE 9. Fuel cell options.

H_2/O_2 Fuel Cells (100% redundancy, 15-day mission, 720 kWhr)		
	Energy Density (Whr/kg)	System Weight (kg)
Dedicated Cryo Tanks	391	1842
Integrated with Propellant Tanks*	1051	685

* Added weight of propellant tanks for slight increase in diameter not included. Reactants are included.

Fuel Cell System Analysis (no redundancy)*

Reactants (kg)	Tank Diameter (m)	Tank Weight (kg)	F.C. weight (kg)	System weight Fc,Rx,Tank	Energy Density (Whr/kg)
Gaseous					
720 kWhr (15 days)					
H_2 30.9	1.57	442	68	1000	720
O_2 243.7	1.46	215			
144 kWhr (3 days)					
H_2 6.2	0.92	88	68	254	567
O_2 48.8	0.73	43			
Cryo					
720 kWhr (15 days)					
H_2 30.9	0.94	224	68	921	782
O_2 243.7	0.74	354			
144 kWhr (3 days)					
H_2 6.2	0.55	45	68	239	603
O_2 48.8	0.43	71			

* 1 fuel cell, 1 set of tanks.

Included in weights: 10% fuel cell weight for mounting; 10% tank weight for plumbing/mounting; 5% reactant weight for ullage.

provides 2 kW for 3 days as 478 kg. An equivalent system that uses the main propellant tanks for reactants might weigh 274 kg (dual redundant, not counting tank mass increase).

MULTIPURPOSE LANDER WEIGHT STATEMENTS

Table 10 shows a multipurpose lander weight statement. The cargo landing task results in the largest deorbit mass that scales the structures, engines, RCS dry mass, and landing systems. The round trip with a crew module results in the largest propellant mass that scales the tanks and thermal protection. The electrical power system uses four dedicated tanks for redundant reactant storage. The ΔV includes an additional 0.43 km/sec for a 15° plane change.

The multipurpose lander pays a penalty of 2300 kg (lunar deorbit mass) in the crew module case for being able to do all three tasks, as compared to a lander designed to do only a round trip with a crew module. The scaling equation described previously was used to determine these masses.

The plots shown in Figs. 2, 3, and 4 and tabulated in Tables 1, 2, and 3 are for similar landers, except the 0.43 km/sec ΔV for plane change is not included and no mass for the airlock/tunnel is included. They are therefore smaller landers. Table 11 shows the same lander sized for N_2O_4/MMH propellants.

LH_2/LOX MULTIPURPOSE LANDER CONCEPTUAL DESIGN

Figures 5 and 6 show a conceptual design of an LH_2/LOX multipurpose lander. The tanks are sized to hold roughly 30,000 kg total of propellant. The H_2 tanks are 3.9 m in diameter, and the O_2 tanks are 2.76 m in diameter. The weight statement for this lander is given in Table 10.

Important features of this lander include (1) airlock/servicing tunnel down the center of the lander to allow easy access on the surface, and pressurized volume for LRUs, inside which many engine connections can be made and broken; (2) flyable without the crew module, which is removable; (3) fits in 30′ heavy-lift vehicle shroud with landing gear stowed; (4) electromechanical shock absorbers on landing gear; and (5) emergency ascent with one or two crew possible without crew module (crew would ride in suits in airlock/servicing tunnel). Figure 7 shows this lander being serviced on the lunar surface and illustrates how the airlock/servicing tunnel allows pressurized access to a surface vehicle. An engine is being removed in the figure.

Figure 1 shows this lander in lunar orbit, about to dock with a large (single-stage) OTV. The OTV is designed to return the lander to the space station for servicing. The OTV delivers the lander to LLO, single stage, and waits in orbit for it to return. The OTV tanks are sized to hold 118,000 kg of LOX/LH_2 propellants.

TABLE 10. LO_2/LH_2 multipurpose lander weight statement.

Delta V, Ascent	0	2.28*	2.28*
Payload, Ascent	0	6,000	0, Inert Mass returned to LLO
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	9,823	9,823	9,823
Structure	1,681	1,681	1,681
Engines	822	822	822
RCS Dry	411	411	411
Landing System	784	784	784
Thermal Protection	2,017	2,017	2,017
Tanks	3,025	3,025	3,025
DMS (GN&C)	150	150	150
Electrical Power †	478	478	478
Airlock/Tunnel	455	455	455
Total Propellant Mass	25,251	32,395	30,638
Ascent Propellant	0	11,334	7,240
Descent Propellant	22,597	18,137	20,486
Unusable Propellant (3%)	678	884	832
FPR Propellant (4%)	904	1,179	1,109
Usable RCS	858	689	778
Unusable RCS (5%)	43	34	39
FPR (20%)	172	138	156
Deorbit or Gross Mass (less payload)	35,074	42,218	40,461
Deorbit or Gross	60,074	48,218	54,461

* Delta V = 1.85 + 0.43 km/sec for a 15° plane change in a 93-km circular orbit.
† Electrical power provided for three days only (2 kW). 100% redundant fuel cells have dedicated redundant tankage.

All masses are kg, all ΔVs, km/sec, Isp = 450 (lbf · sec/lbm).

TABLE 11. N_2O_4/MMH multipurpose landers.

Delta V, Ascent	0	2.28*	2.28*
Payload, Ascent	0	6,000	0, Inert mass returned to LLO
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	7,899	7,899	7,899
Structure	1,955	1,955	1,955
Engines	956	956	956
RCS Dry	478	478	478
Landing System	912	912	912
Thermal Protection	1,006	1,006	1,006
Tanks	1,509	1,509	1,509
DMS/GN&C	150	150	150
Electrical Power †	478	478	478
Airlock/Tunnel	455	455	455
Total Propellant Mass	36,398	50,767	45,429
Ascent Propellant	0	15,702	9,406
Descent Propellant	32,861	30,665	31,927
Unusable Propellant	986	1,391	1,240
FPR Propellant (4%)	1,314	1,855	1,653
Usable RCS	990	923	961
Unusable RCS	50	46	48
FPR RCS (20%)	198	185	192
Deorbit or Gross Mass (less payload)	44,297	58,666	53,328
Deorbit or Gross	69,297	64,666	67,328

* Delta V = 1.85 + 0.43 km/sec for a 15° plane change in a 93-km circular orbit.
† Electrical power provided for three days only (2 kW). 100% redundant fuel cells/tank sets.

All masses are kg, all Δ Vs, km/sec, Isp = 330 (lbf · sec/lbm).

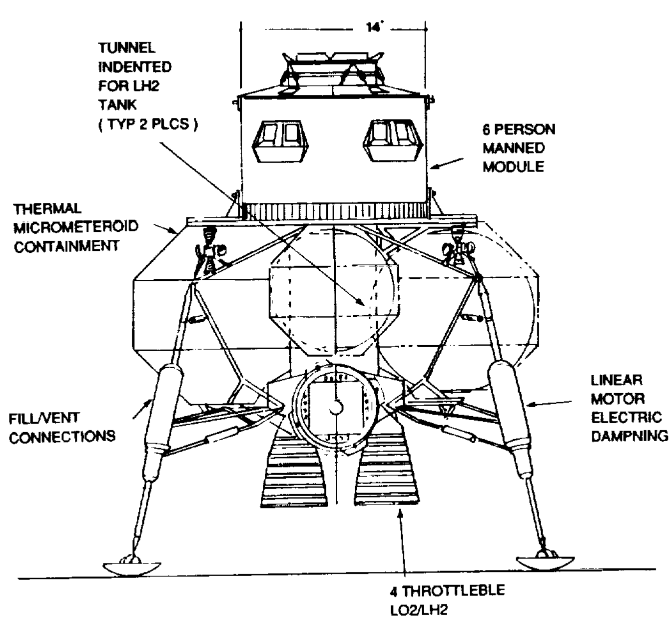

Fig. 5. LOX/LH_2 reusable lunar lander, side view.

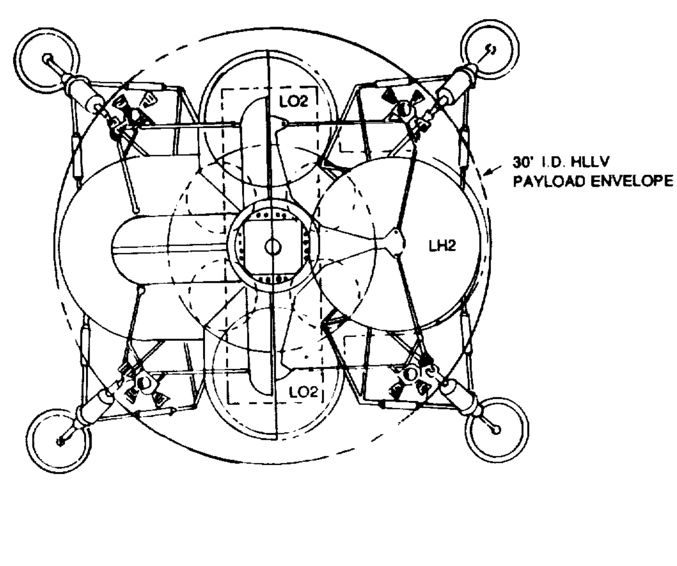

Fig. 6. LOX/LH_2 reusable lunar lander, top view.

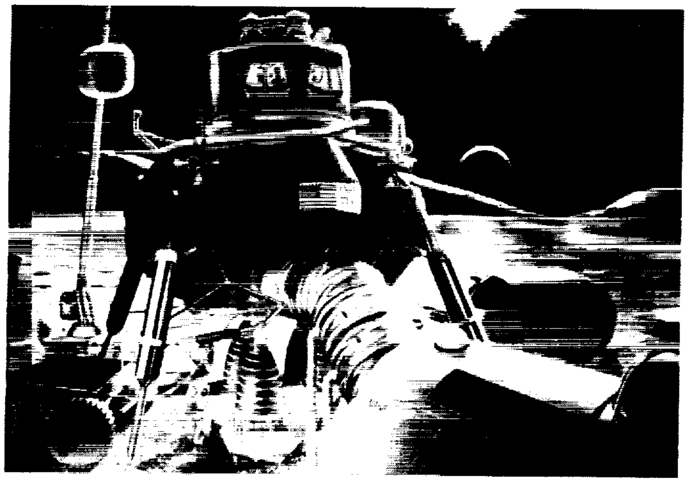

Fig. 7. Lander on surface.

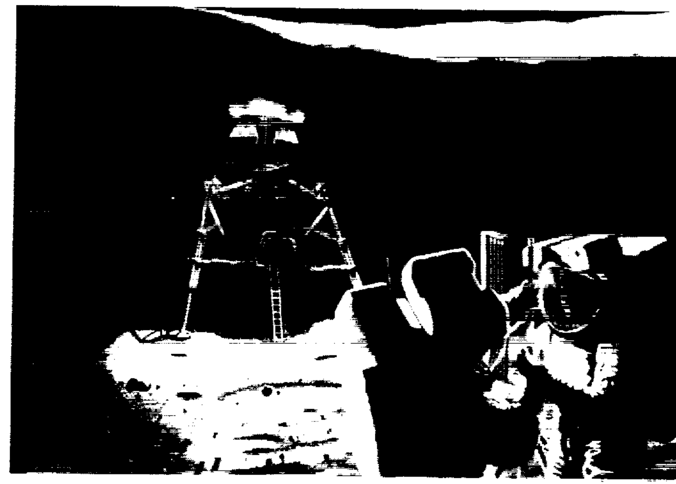

Fig. 8. Lander on surface at pole.

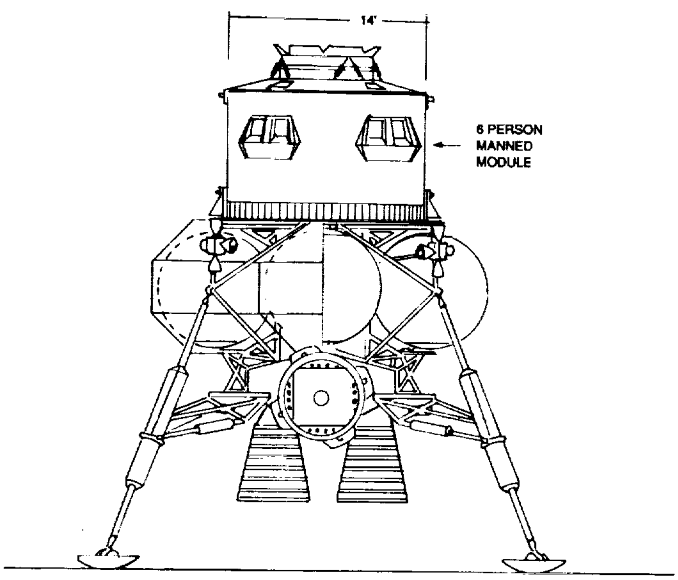

Fig. 9. Advanced storable reusable lunar lander, side view.

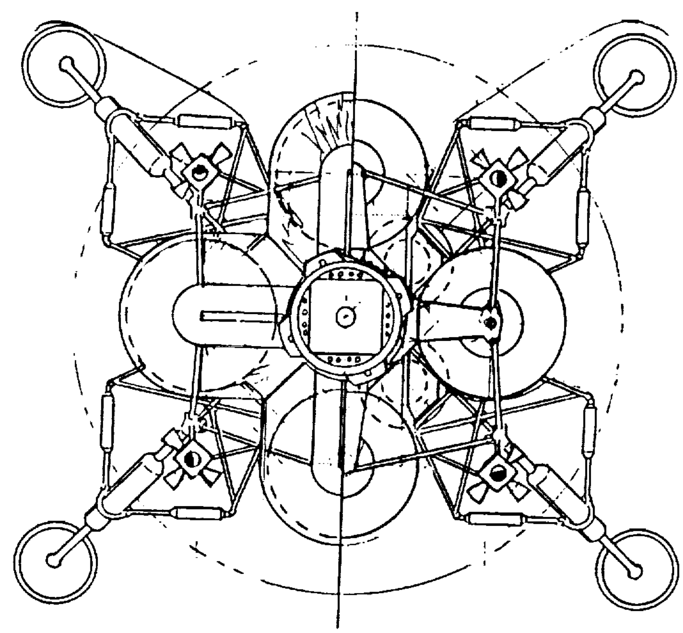

Fig. 10. Advanced storable reusable lunar lander, top view.

Figure 8 shows the lander on the surface at the poles. The lander may also serve as a suborbital "hopper" if propellant loading on the lunar surface is provided. The figure illustrates normal egress, without a pressurized vehicle.

ADVANCED STORABLE MULTIPURPOSE LANDER CONCEPTUAL DESIGN

Figures 9 and 10 show a lander with equivalent capability to the LOX/LH_2 lander, except using N_2O_4/MMH propellants. This lander, though considerably heavier than the LH_2/LOX lander, is much smaller, due to higher propellant density. Its features are essentially the same as the previously described lander.

The propellant capacity of this lander is 35,000 kg divided into four tanks of 16 cu m each. Tank diameter is 2.5 m for all tanks.

COST

Lander production costs were determined using a cost estimating relationship (CER) model. With this method, design and fabrication cost curves are developed for each vehicle component, relating the component's historical costs to its weight. Components from the Gemini, Apollo, Skylab, and shuttle programs were considered when developing the CERs. Where several significantly distinct classes of a given component existed, a separate CER was created for each class. The cost curves generated using this method usually had a correlation coefficient of 0.9 or better. All costs have been adjusted for inflation, and are expressed in 1988 dollars. Program management wrap factors are included in the CERs.

Total design and development cost is estimated to be $1539 million, and total fabrication cost is estimated to be $759 million per vehicle. Total program cost for ten vehicles is $9129 million.

To verify the reasonableness of these estimates, they were compared to actual Apollo LM engineering and fabrication costs. Estimated design and development costs were within 7% of actual LM costs (when adjusted for inflation), and estimated fabrication costs were within 2% of actual LM costs.

Design/Development Costs	
Apollo LM (1967 $M)*	378
Apollo LM (adj. to 1988 $M)	1672
New lunar lander (1988 $M)	1539

Fabrication Costs	
Apollo LM (8 units, 1967 $M)	1354
Apollo LM (1 unit, 1967 $M)	169
Apollo LM (1 unit, adj. to 1988 $M)	745
New lunar lander (1 unit, 1988 $M)	759

*These numbers come from a 1967 document (*Grumman Corp.*, 1967). Other significant development costs were incurred after 1967 that are not shown here.

REFERENCES

Apollo 11 Mission Report, MSC-00171. Manned Spacecraft Center, Houston, pp. 7-11.

Eagle Engineering (1988) *Lunar Lander Conceptual Design,* Report No. 88-181, NASA Contract No. NAS9-17878.

Grumman Corp. (1967) *Lunar Module Program Cost Summary — Total Program.*

PRELIMINARY DEFINITION OF A LUNAR LANDING AND LAUNCH FACILITY (COMPLEX 39L)

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A preliminary definition of a lunar landing and launch facility has been formulated. A permanently manned lunar base and a baseline lunar module are assumed. The major features of the facility are specified and major design areas are described.

INTRODUCTION

We have formulated a preliminary definition of a lunar landing and launch facility (Complex 39L). A Phase III lunar base is considered (*Roberts,* 1987; *Ride,* 1987). Without specifying lunar base scenarios, three traffic levels are envisioned: 6, 12, and 24 landings/launches per year. We have assumed a single multipurpose vehicle for the lunar module, whose characteristics will be described below. The design and specifications of the vehicle and of the lunar base are outside the scope of this study; however, these two items will have an impact upon those items considered within the scope of this study because of the interaction at the boundaries of our system. The scope of this study is graphically illustrated by the systems diagram of Fig. 1. Here, major functions or facilities are represented in a block diagram. The dashed line represents the boundary of Complex 39L. This is a simplified version of this diagram. Other items could be included, e.g., lunar surface transportation and electromagnetic launchers. As previously mentioned, those items either on or outside the dashed lines that will have a significant impact upon the design of those items within the boundary will be discussed. Based upon this diagram, nine major design items are considered: (1) landing/launch site considerations; (2) structure, shelter, safety, and environmental needs; (3) landing/launch guidance, communication, and computing needs; (4) lunar module surface transport system; (5) heavy cargo unloading/loading systems; (6) personnel unloading/loading systems; (7) propellant unloading/loading systems; (8) vehicle storage; and (9) maintenance, repair, test, and check-out requirements.

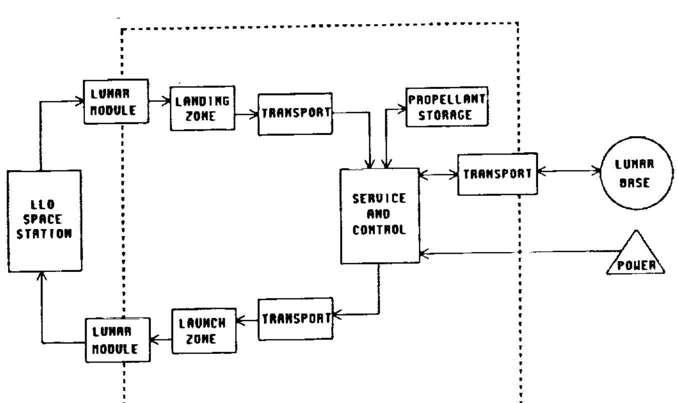

Fig. 1. Lunar landing and launch facility (Complex 39L) systems diagram.

We provide a general, conceptual description of each of these items. We have obtained preliminary sizes, capacities, and/or other relevant design data for some of these items.

DESIGN SCOPE

The Transportation System and Lunar Module

We assume a baseline transportation system (*Astronautics,* 1987). The transportation infrastructure (Fig. 2) consists of a low Earth orbit (LEO) space station, a low lunar orbit (LLO) space station, orbit transfer vehicles (OTVs), lunar modules (landers), and a lunar landing and launch facility (Complex 39L). Both the OTVs and the lunar modules will be reusable with no expendable vehicles considered. For the baseline transportation system all vehicle propulsion systems use hydrogen/oxygen (H/O).

The basing scenario includes the space station in LEO to provide servicing, payload accommodation, and propellant supply. Propellant refers to liquid oxygen as well as liquid hydrogen. A similar basing node located at LLO will be needed as a propellant storage depot, and for the servicing of either OTVs or lunar module systems. The final basing node will be at the lunar surface and will have propellant storage, payload transfer, and lunar module servicing capabilities.

For the flight from LEO to LLO the OTV will carry a manned capsule, payload, and propellant for the lunar module and for its return to LEO. For the flight from LLO to Complex 39L the lunar module will carry a manned capsule, payload, and propellant for its return to LLO. Unmanned OTVs and lunar modules in which the manned capsule is replaced with an increased payload can also be used. For this design we assume a preliminary baseline lunar module design.

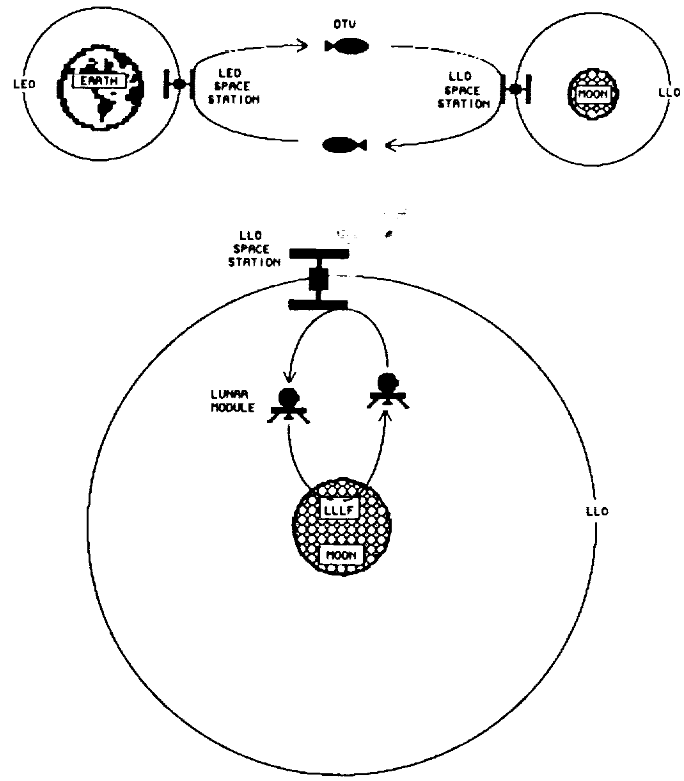

Fig. 2. Earth-Moon transportation infrastructure.

The Lunar Base

The lunar base is assumed to be a permanently occupied facility in the timeframe of 2005-2009 (Phase III). The human population will range from approximately 10 to 30 during this time period. The base will emphasize both scientific research and *in situ* resource utilization. It will be desirable to make use of resources available on the Moon in an attempt to minimize the required Earth launch mass (ELM). For this study we take a conservative stance and assume minimal use of lunar resources.

The lunar base can be broken down into several subfacilities. These will include habitat modules, various lunar production facilities, nuclear power facility, and lunar landing and launch facility. A preliminary plot plan for Complext 39L is given in Fig. 3. We now discuss the interaction between Complex 39L and the other subfacilities, i.e., the boundaries of our systems diagram.

It is assumed that habitat and laboratory modules similar to those used in the LEO space station will be used on the Moon. The modules will be covered with lunar regolith for radiation protection (*Guerra,* 1988). The increase in the number of inhabitants must be accompanied by an increase in the number of habitat modules. The landing, unloading, transportation, and assembly of habitat modules will be an ongoing activity at the lunar base. An increase in the number of inhabitants will also call for an increase in consumables (water, food, oxygen, etc.)

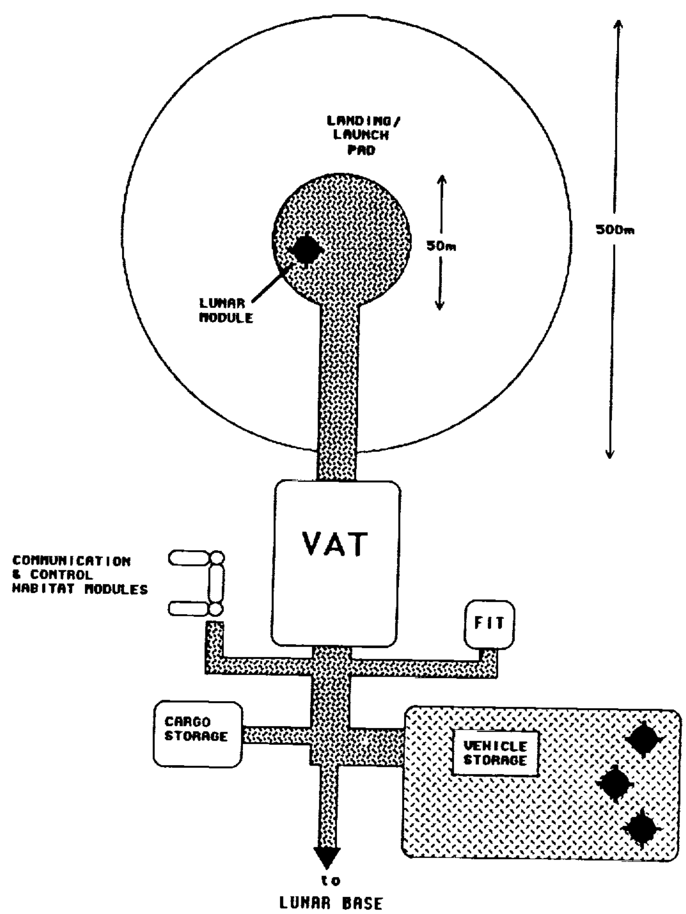

Fig. 3. Complex 39L plot plan.

The baseline lunar module is a reusable, two-engined vehicle capable of delivering 15,900 kg (manned capsule plus payload) to the lunar surface from LLO and returning to LLO with an equivalent payload (*Astronautics,* 1987). Specific engine features are $I_{sp} = 470$ sec, thrust per engine = 33.4 kN, mass per engine = 95 kg, and mass ratio of O_2 to $H_2 = 5.5$. The manned capsule will have the capability of transporting six people (two pilots and four passengers). The lunar module has a propellant capacity of 21,000 and 4000 kg for oxygen and hydrogen respectively. The vehicle lifetime with minimum maintenance is estimated to be 40 flights. Dimensions for the lunar module are estimated from the weight and engine size, and by making comparisons with the lunar excursion module (LEM) used in the Apollo missions. The height (10 m) is the vertical distance from the footpads to the top of the vehicle. The diameter (13 m) is the distance from footpad to diagonal footpad. The lunar module has four footpads. This vehicle is consistent with studies by Johnson Space Center (*Alred,* 1987).

It is desirable that the vehicle have a "modular" design, i.e., main subsystems (propellant tanks, engines, cargo modules, manned capsules, etc.) should be easily removable and replaceable. Due to the expense and hazards of extravehicular activity (EVA) it will be advantageous to make maximum use of robotics to perform the required lunar surface tasks (*Eagle Engineering,* 1987). However, we believe that at this stage in lunar base development many of the tasks will be diverse and complex enough that most repairs will need to be made by personnel wearing spacesuits. This requires special design consideration in an attempt to accommodate the person making the repairs.

required. We assume a baseline crew rotation of six months. With a population of 30 this will require 15 lunar module flights per year.

The construction and operation of a lunar liquid oxygen (LLOX) production facility will occur during Phase III of lunar base development (*Roberts,* 1987; *Ride,* 1987). As the amount of LLOX exported increases, so must the capacity of the LLOX plant. When the capacity must be increased, another LLOX production module must be landed, unloaded, transported, assembled, and put on line. Each LLOX production module is assumed to have a standard production capacity, and the LLOX plant will be made up of these modules operating in parallel.

The power requirements of the lunar base will increase as the production capacity and number of inhabitants increase. The power available is estimated to range from 1 to 10 mW over the given time period (*Roberts,* 1987). To meet the increasing demand, nuclear reactors and supporting elements must be landed, unloaded, transported, and assembled. Other supplemental energy sources such as solar energy may also be used with similar installation requirements.

The lunar base subfacilities will be separated from Complex 39L by a specified distance. This distance will depend on safety considerations and the site of the base. The potential of an explosion, large navigation errors in landing, and rocket engine blast will warrant separating Complex 39L some distance from other facilities. An appropriate distance needs to be determined.

Other Considerations

In this section we discuss a number of items that are considered outside the scope of the design, i.e., outside the dashed boundary of Complex 39L as indicated in Fig. 1. Consideration of items being either inside or outside the scope of design follows good design practice (*Linsley,* 1988).

For purposes of our design, we consider only manned lunar modules. We realize that design requirements for manned as opposed to unmanned vehicles are considerably more stringent and that there will be an effort to use unmanned vehicles to the greatest extent possible. Manned vehicles will also impose more stringent constraints on the design of Complex 39L. The only area where unmanned vehicles will impose more requirements is in the guidance and communications area. These increased requirements can be considered within the margin of error in this preliminary analysis.

It is generally accepted that a high degree of utilization of automation and robotics technology will be used in lunar base activities. While recognizing this, we take many of our design concepts from current technology that has not yet experienced automation or robotics technology advances. Again, this is done partially in the interest of obtaining a conservative design. We also consider that the highly automated and roboticized facilities will be heavily interspersed with rather low-technology devices. Designers of lunar base equipment should look to the seven basic machines of elementary physics for initial design concepts.

A number of advanced concepts have been omitted from this study. We mention two of these. Electromagnetic launchers, which are a popular concept in discussions of lunar base design, are not considered in this Phase III design. Another, less popular, concept that we have considered but omitted is the design of a landing and launch pad from which the recovery of water vapor from the exhaust plume is attempted. We consider these interesting concepts to be beyond Phase III.

MAJOR DESIGN ITEMS

Landing/Launch Site Considerations

The lunar module will touch down vertically on a specified zone (landing/launch pad). For lunar module transportation requirements and dangers from engine blast effects, it is desirable to have a prepared surface. Loose particles on the pad can become dangerous projectiles in the presence of engine blast from the lunar module. With a prepared surface this problem can be greatly diminished. In this study, we assume that the same pad will be used for both landing and launch.

The landing pad will be circular with a diameter of 50 m (approximately four times the diameter of the lunar module). This figure was arrived at by making comparisons with terrestrial vertically landing vehicles. A circular area with a radius of approximately 250 m from the center of the landing pad will be cleared of large rocks and equipment (*Eagle Engineering,* 1988). The landing area will be marked with lights similar to a terrestrial airport. Also, television cameras will be present to aid the controllers in the communication and control facility. This equipment will be within the 250-m circle, and must be designed to handle any engine blast effects that may occur, e.g., replaceable lens covers on camers. The number of pads will depend on the flight schedule and the time required for maintenance. Figure 3 shows one pad, though more may be required.

Shelter, Structure, Safety, and Environmental Needs

It is assumed that the lunar module will spend a significant amount of time on the lunar surface. This could be from two weeks to two months. It will be desirable to control the temperature of the vehicle by removing it from direct sunlight. This will decrease the boil-off of cryogens and also provide a more constant thermal environment.

We propose the use of a quonset hut tent-like structure (Fig. 4). This structure will be referred to as the vehicle assembly tent (VAT). The facility will be large enough to contain four lunar modules. The dimensions are 50 m long, 36 m wide, and 18 m high at the center line. Entrances, 15 m high and 16 m wide, will be located at each end of the structure. A framework will be

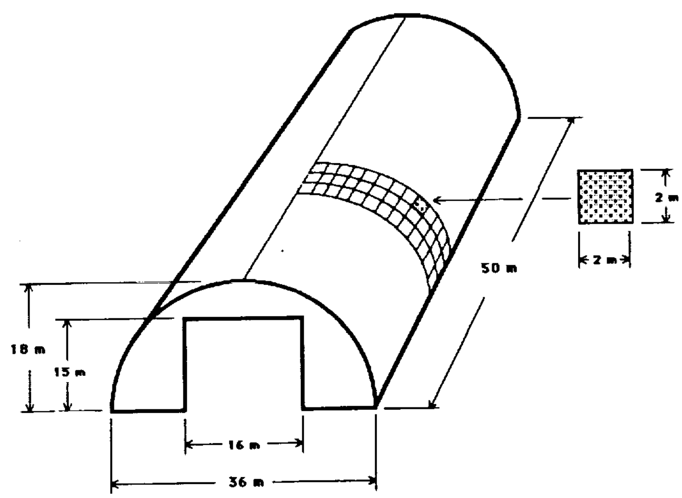

Fig. 4. Complex 39L vehicle assembly tent (VAT).

constructed of a material such as 2014-T6 aluminum (*Aluminum Association,* 1975). Highly reflective panels made of a mylar/evaporated aluminum laminate will shield equipment inside the VAT from incoming thermal radiation. These panels are expected to reflect approximately 90% of the thermal solar spectrum (*Incropera and DeWitt,* 1985). Other panel materials and laminates are being investigated. Initial calculations, with one layer of panels, give a surface-level temperature inside the VAT of approximately 0°C during the lunar day. It was found that using two layers of panels separated by 0.1 m gave a decrease in surface temperature of only 8°C. Movable flaps will be used to cover the entrances at each end of the structure. They will serve to block glare and possible particles from engine blast. These will be made of the same material as the panels. They will cover an area of 240 sq m at each end. A total mass of approximately 10,000 kg has been derived for the proposed structure.

Most servicing and unloading/loading operations will be performed in the VAT. Artificial lighting must be provided where men or video cameras are working. This is not a pressurized facility and personnel must wear spacesuits. This facility will not block radiation that is potentially dangerous to humans. The amount of time humans can work in this environment will be limited (*Adams and Shapiro,* 1985). It may be determined that all surface operations will be best performed during the 14-day lunar night. Electronic devices will also be affected by high doses of radiation in the form of both hardware and software upsets, and should be specially designed for lunar application.

Landing/Launch Guidance, Communications, and Computing Needs

The lunar module will be manually controlled by two pilots. Assistance will be provided by a surface communications and control facility similar to that of a terrestrial airport. It has been determined that currently available terrestrial navigation systems can be applied to achieve high degrees of landing and positioning accuracies, and these systems will be fully operational by Phase III of the lunar base development (*Eagle Engineering,* 1988). A communications and control facility will be located at Complex 39L. Habitat modules will be used to house the operators and equipment. Approximately two people will be required to operate the facility.

Lunar Module Surface Transport System

The lunar module will be transported from the landing/launch pad to the VAT. We envision the use of self-powered dollies. A dolly will be placed under each footpad and the lunar module will be guided to the VAT. It is assumed that the positioning of dollies and guidance of the lunar module will be performed by a person in a spacesuit, though this is an area with potential robotic application that should be investigated. This same system will be used for transporting lunar modules to vehicle storage.

Heavy Cargo Unloading/Loading Systems

Heavy cargo items such as habitat modules, construction equipment, nuclear reactors, and LLOX production modules will be landed at the lunar base on a regular basis. These items will be attached to the lunar module and may or may not be stored in containers. The lunar module will be transported into the VAT fully loaded with payload. Once in the facility the module will be unloaded. We envision the use of a bridge crane. The crane will encompass an area of 15 m by 30 m at a height of 15 m. For a preliminary design we assume a maximum load of 45,000 kg. We design the center beam to have a deflection less than 0.05 m with the maximum load applied at the center point. It was found that a standard 24×62 wide flange beam constructed of 4340 low-carbon steel will meet these design requirements. This is a baseline design and other construction materials are being investigated. We assume that the entire structure will be constructed of the same members. This gives a total crane mass of approximately 20,000 kg.

Operations will begin by detaching (unstrapping) a payload from the lunar module. The crane will then be positioned and attached to the payload. The payload will be lifted, transported away from the lunar module, and lowered onto one or more dollies. If the cargo is not to be delivered directly to its final destination, it will be transported to cargo storage to await further processing. Complex 39L cargo storage will be a separate tent structure similar to the VAT and located nearby (Fig. 3). The lunar module will be loaded in a reverse manner.

Personnel Unloading/Loading Systems

We envision three modes of personnel unloading/loading. The first requires EVA and is similar to that used in the Apollo missions. The personnel will don spacesuits and exit the lunar module by climbing down a ladder that is attached to one of the module legs. This can be done either on the landing/launch pad or in the VAT. If the personnel exit or enter the vehicle on the pad they must either walk or be transported to or from the pad.

The second mode of unloading/loading is for the personnel to remain in the module until it has been transported into the VAT. They will then disembark into the pressurized compartment of a lunar surface transport vehicle. This is a "shirt-sleeve" transport operation where spacesuits are not required. This mode will require a pressurized transport vehicle and an airlock mechanism to connect the two vehicles.

The third mode of unloading/loading is again for the personnel to remain in the lunar module until it has been transported into the VAT. Here, the manned capsule of the lunar module will be detached, lifted by the bridge crane, and placed on one or more dollies. The entire manned capsule will then be transported to the habitat modules where the personnel can disembark through an airlock. This is also a "shirt-sleeve" operation where EVA is not required. A separate pressurized transport will not be required as in the second mode. This example illustrates the integration (modularity) that we believe is necessary for a successful lunar base.

Propellant Unloading/Loading Systems

The lunar module will land at Complex 39L with some propellant remaining in its fuel tanks. Assuming no LLOX is available this will be all the hydrogen and oxygen required for the return flight to LLO. The propellant can either be left in the fuel tanks or transferred into propellant storage tanks. If boil-off from the fuel tanks is large, then it would be preferable to store the cryogens in larger tanks with active cooling systems. Hydgrogen and oxygen storage tanks will be located at Complex 39L. We assume that active cooling systems will be used. The cooling systems will be designed to achieve a specified maximum boil-off.

As a design criterion we require that enough propellant be stored to evacuate the entire lunar base population. For a

population of 30 this would require storage of approximately 150,000 kg of oxygen and 30,000 kg of hydrogen. If one spherical tank is used to store each cryogen, this would require tank diameters of roughly 6 m and 9 m for oxygen and hydrogen respectively. Multiple tanks of differing geometries may be used.

The storage tanks and pumps will be located in a separate tent near the VAT. This tent is referred to as the fuel inventory tent (FIT) (Fig. 3). One method for defueling/fueling the lunar module is to remove the propellant tanks from the module with the bridge crane, place them on dollies, and transport them to the FIT in the same manner that cargo is transported.

Vehicle Storage

A long-term vehicle storage area will be provided at Complex 39L (Fig. 3). This will be an area near the VAT that has been cleared of large objects. At this stage in the lunar base development, we envision an area large enough to contain six lunar modules (approximately 1000 m^2). With an increase in the lunar module fleet and landing/launch rate, this area will need to be enlarged.

The lunar module will be transported to vehicle storage if it has been damaged beyond repair, exceeded its operational life, or will not be used for a long period of time. The lunar modules will have been defueled prior to storage. The module will be transported to a storage location as previously discussed. A dome tent will then be pitched over it. This will be a tent made of the same material used in the VAT attached to a support frame. Lunar modules in vehicle storage will be used for cannibalizations. We assume that lunar module components will have varying operational lives. Some components will still be operational when the vehicle as a whole is not. Working components from vehicles in storage will be used to repair operations vehicles in the VAT.

Maintenance, Repair, Test, and Check-Out Requirements

The lunar module is a reusable vehicle and will require regular maintenance with each flight. Unlike presently operated reusable terrestrial space vehicles, the lunar module should have minimal maintenance requirements. For our highest frequency flight schedule (24 flights/year), the lunar module turnaround time will be two weeks (14 days). For a baseline case we assume that routine maintenance will be performed by two personnel. However, more manpower will be required if a significant problem develops or if major systems alterations are required. We identify four main procedures: initial safing, postflight servicing, lunar module modification, and preflight servicing. The following is a preliminary description of some of the operations that will be performed during each of our identified main procedures.

Initial safing will include transportation of the lunar module to the VAT, defueling propellant tanks, attachment of ground power and purge lines to the lunar module, purging main engines and fuel lines to remove possible moisture resulting from hydrogen/oxygen combustion, and unloading payloads. Also, the lunar module crew will disembark sometime during the initial safing procedure. This is a preliminary list of required operations that can easily be expanded upon.

After initial safing is complete, postflight troubleshooting begins to determine anomalies that may have occurred during launch, spaceflight, or landing. An umbilical cord containing electrical, communication, instrumentation, and control lines is connected to the vehicle. Visual and electronic inspections are performed on the lunar module. Along with postflight inspection, routine servicing will include lubrication, recharging of environmental systems, recharging/regenerating batteries and fuel cells, and others.

Lunar module modifications will then be made if necessary. Modifications will include replacing damaged components, adding or removing equipment to meet future mission requirements, and replacement of outdated hardware/software. Lander modification, if extensive, can be performed over a long period of time while the craft is in vehicle storage. However, many modifications will be performed in parallel with routine servicing.

The lunar module will finally be prepared for launch. Preflight servicing will include installation of flight supplies and payload, fueling of propellant tanks, loading of personnel, final visual and electronic check-out, and transportation from the VAT to the pad.

CLOSURE

We have presented short descriptions or specifications of our nine designated design items. The next stage in our design process is to determine preliminary estimates for the major resource requirements of our system. We identify three major resources to be mass, power, and manpower. The cost of the lunar base will be directly related to the resource requirements. While mass and power requirements can generally be determined by standard engineering methods, assessments of manpower requirements can be difficult. One assumption that is made is that all operations will be undertaken by a minimum of two personnel. This is a safety consideration that mimics the "buddy system" that is used in SCUBA diving.

In addition to our three major resource requirements, we recognize three main resource requirement areas: construction, operation, and maintenance. The construction area represents the resources that will be required during the construction phase of Complex 39L. This will include, but is not limited to, clearing a site, landing/launch pad preparation, pitching of various tents, and assembly of cranes and other structures. The operation area represents the resource requirements for the "steady-state" operation of the facility. The maintenance area represents the facility maintenance requirements, e.g., refurbishment of landing/launch pads, tent structures, and other hardware. More resources and resource areas can be incorporated into this design methodology as the need for greater and greater design detail is required.

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REFERENCES

Adams J. H. and Shapiro M. M. (1985) Irradiation of the Moon by galactic cosmic rays and other particles. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 315-327. Lunar and Planetary Institute, Houston.

Alred J. (1987) *Lunar Transportation Study.* Lunar Transportation Team/ED23, Advanced Programs Office, NASA Johnson Space Center, Houston.

The Aluminum Association (1975) *Engineering Data for Aluminum Structures.* 750 Third Avenue, New York, New York.

Astronautics Corporation of America (1987) *Lunar Surface Base Propulsion System Study.* NASA Contract NAS9-17468.

Eagle Engineering Inc. (1987) *Lunar Surface Operations Study*. NASA Contract NAS9-17878, EEI Report 87-172.

Eagle Engineering Inc. (1988) *Lunar Base Launch and Landing Facility Conceptual Design*. NASA Contract NAS9-17878.

Guerra L. (1988) A commonality assessment of lunar surface habitation. Paper No. 85d, AIChE Spring National Meeting, New Orleans.

Incropera F. P. and DeWitt D. P. (1985) *Fundamentals of Heat and Mass Transfer*. Wiley, New York.

Linsley J. (1988) Analogies between the design and operations of batch chemical plants and launch and landing facilities. Paper 85h, AIChE Spring National Meeting, New Orleans.

Ride S. K. (1987) *Leadership and America's Future in Space; Report to the Administrator*. U. S. Govt. Printing Offie, Washington, DC. 63 pp.

Roberts B. B. (1987) *Lunar Bases in the 21st Century: The First Step Towards Human Presence Beyond Earth*. NASA Johnson Space Center, Houston.

LUNAR BASE LAUNCH AND LANDING FACILITIES CONCEPTUAL DESIGN

N93-17430

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The purpose of this study was to perform a first look at the requirements for launch and landing facilities for early lunar bases and to prepare conceptual designs for some of these facilities. The emphasis of the study is on the facilities needed from the first manned landing until permanent occupancy, the Phase II lunar base. Factors including surface characteristics, navigation system, engine blast effects, and expected surface operations are used to develop landing pad designs and definitions of various other elements of the launch and landing facilities. Finally, the dependence of the use of these elements and the evolution of the facilities are established.

INTRODUCTION

The likelihood of the establishment of a permanent lunar base has become sufficiently real that serious efforts are underway to mold plans and scenarios for its development. Issues surrounding the facilities needed to support safe and consistent landings must now be addressed to ensure they do not represent primary drivers of the early lunar base. This study was performed to examine the requirements for launch and landing operations and to prepare design definitions for the elements of these facilities. The focus of the study is on the lunar base, beginning at the first manned landing until permanent occupancy. This period of base development has generally been called Phase II, since it is the second in a three-stage process. This paper documents a study of launch and landing facilities done as a part of the Lunar Base Systems Study being performed by the Johnson Space Center Advanced Programs Office.

Requirements and design considerations must be defined generally before concepts for facilities can be developed. The surface characteristics of the Moon will cover site preparation issues, some landing capability requirements, and the degree of autonomy the vehicle must possess. The navigation systems on the flight vehicle will dictate what sort of navigation support must be provided by lunar base facilities. Another type of interaction with the flight vehicle, the effects of blast from the rocket engine, defines requirements for many aspects of facilities designs. Finally, the expected general operations of the base and its landing facilities must be described to provide a framework for selection of what elements must be designed.

Once the elements of the launch and landing facilities have been defined, they can be fitted into more specific plans for the lunar base. The growth and evolution of launch and landing facilities will naturally be coupled with the growth and evolution of the lunar base itself. To complete the conceptual design, the dependencies between these base and launch and landing facilities must be defined. These dependencies can be used in the future in planning the lunar base.

SURFACE CHARACTERISTICS

The first task in the definition of landing facilities is the characterization of possible base locations. These site characteristics have general effects on the design requirements and setup operations of landing facilities. The characteristics of interest are surface roughness, soil mechanics data, lighting, and Earth visibility. Given its age, the lunar surface is fairly homogeneous in many respects. Landing pads can be designed without regard to base site.

Roughness

In general, landing sites with relatively low slopes of 4° to 6° for 25-m ranges can be found over the entire lunar surface. Some locations, such as the sides of large craters and mountainsides, may have unacceptable slope characteristics. Mountainside slopes of around 30° are not uncommon. Data on the roughness of the surface comes from several different sources:

1. Photogeologic terrain assessment is the first and most straightforward. This simply involves assuring that candidate landing sites do not lie on the sides of mountains.

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2. Photogeologic measurements of slopes based on high-resolution vertical photography taken from lunar orbit provide surface slope distributions. Published data is available for all the candidate Apollo landing sites, as well as other areas of the Moon. Figure 1 shows some of these data.

3. Counts of the number of impact craters in a series of size classes based on high-resolution vertical photograpy taken from lunar orbit provide general roughness data. Figure 2 presents a summary of crater counting data before the Apollo 17 mission (*Minutes of Apollo Site Selection Board*, 1972).

Soil Mechanics

Bearing strength, slip resistance, and grain size are important characteristics when landing surfaces are considered with respect to landers. Strong variations are generally not found over the lunar surface, indicating that landing pad preparation and lander foot pads and legs may be designed without regard to specific base sites. Considering Apollo experience, landers can be designed for an unfinished surface.

The lunar surface consists of a fine-grained soil with over half the material finer than 0.075 mm (*Mitchell et al.*, 1973). Table 1 summarizes other soil physical properties for the Apollo 14 through 17 landing sites. For reference, the Apollo lunar module placed a stress on the surface of about 0.69 N/cm^2. Such stresses resulted in penetrations of the lunar surface of less than a centimeter in firm soil to a few centimeters in soft soil. The angle of internal friction of lunar soil is equivalent to the angle of repose for loose soil such as on the side of a mountain. The tangent of the angle is equal to the coefficient of internal friction, 0.73 to 0.90.

Earth Visibility

The visibility of Earth from the selected base site will affect the degree of autonomy of the lander and its interaction with the landing site. The ability of vehicles to receive Earth-generated navigation updates will influence the need for lunar-based navigation systems. Continuous, real-time communication with Earth is highly desired. Earth support of most operations will be required to make the best use of crew time on the lunar surface. The effects must be described for each specific landing site.

Sites on the limb of the farside will not present good opportunities for updates without prior placement of either surface or space-based relays. The western limb does allow considerable Earth tracking of landers in the initial parts of the descent, but final descent will generally be invisible to Earth systems.

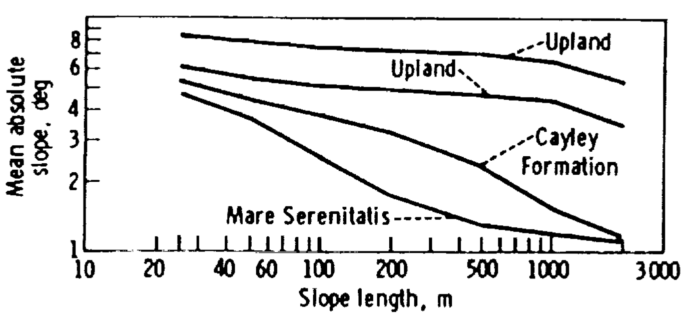

Fig. 1. Lunar slope-frequency distribution (*Moore and Tyler*, 1973).

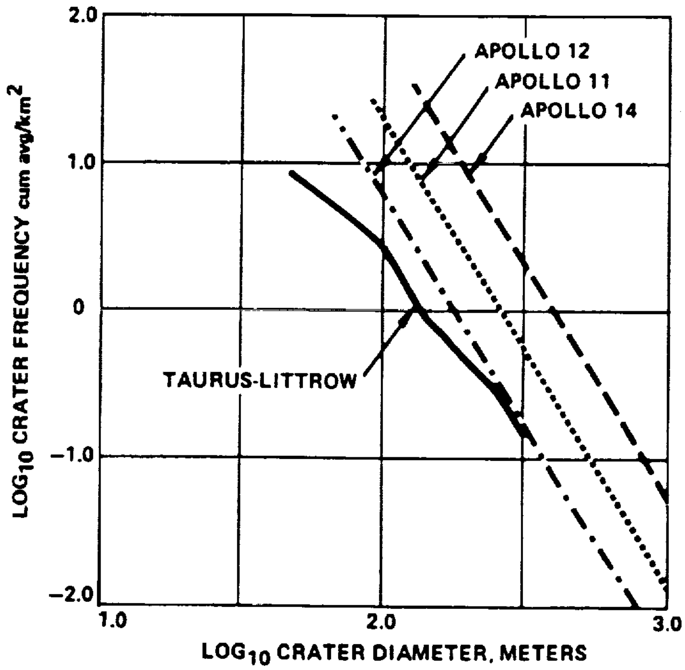

Fig. 2. Size distribution of lunar craters.

Lighting

Lighting mainly affects the time crew-controlled landings may occur for most sites. Polar sites, however, have continuously low solar angles and landing systems, especially during early missions, and must be able to handle hidden features and long shadows. Again, these effects must be analyzed with respect to each particular site.

NAVIGATION SYSTEMS

Flight operations are intended to result in landings with meter accuracy. One of the primary purposes of the landing facilities and the equipment they encompass is to ease flight vehicle operations from orbit-to-surface and surface-to-orbit descent and ascent.

During descent, flight vehicle navigation and guidance systems must be provided position updates, and during final stages of landings must be able to find relative positions and velocities to within accuracies of meters. In particular, unpiloted cargo landers will require this level of accuracy to land on a specific site. The vehicle inertial platforms should be updated on the orbit before descent and then continuously from the time of descent to landing.

The navigation systems provided as part of the lunar base landing facility may be relatively simple systems of radar transponders with known locations. Onboard systems will use terrain- and feature-matching systems, similar to those used by current cruise missiles, during periods when the base is out of view. In short, the navigation systems can use currently available terrestrial systems applied to the lunar surface to achieve high degrees of landing and positioning accuracies.

TABLE 1. Soil properties.

	Mechanical Data					
Soil Consistency	G N/cm³	Porosity	Void ratio, e	D_r	Φ_{TR}	Φ_{PL}
Soft	0.15	47%	0.89	30%	38°	36°
Firm	0.76 to 1.35	39% to 43%	0.64 to 0.75	48% to 63%	39.5° to 42°	37° to 38.5°

G = penetration resistance gradient; D_r = relative density = $(e_{max}-e)/e_{max}-e_{min})$, based on standard ASTM methods; Φ_{TR} = angle of internal friction, based on triaxial compression tests; and Φ_{PL} = angle of internal friction, based on in-place plate shear tests. From *Mitchell et al.* (1973).

TABLE 2. Navigation system advantages and disadvantages.

System	Advantages	Disadvantages
Lunar Orbit Global Postitioning Satellite (GPS) type system.	Terminal, perhaps landing accuracy navigation over entire surface.	Many satellites required. Expensive to place. Accuracy limited. Not adequate for touchdown navigation.
Earth-orbit GPS system or Earth-based radar.	Nothing to place or power on lunar surface. Good for orbit determination on the nearside.	GPS accuracy unknown. May require large antenna. Earth side only.
Long and Medium Range Lunar Surface Transmitters: TACAN, LORAN, low frequency.	Several low-frequency transmitters may provide low-accuracy global coverage. Can be placed and powered at base for local navigation and orbit updates. Terminal accuracy.	Heavy ground stations. Large antennae. Accurate over a limited range only. Low frequency does not provide high accuracy for any location. Low-frequency global coverage requires several transmitters at different places.
Instrument Landing System or Microwave Landing System at base.	Can be placed and powered at base. Landing accuracy.	Terminal and landing navigation only for area close to transmitter.
Lunar Surface-Based Radar (located at base).	Enables range safety thrust termination. Can provide updates to vehicles in orbit. Low mass system.	Local area navigation only.
Cruise missile type onboard terrain matching radar on lander with transponders on surface.	Transponders only on surface in landing area. Very low mass. Landing accuracy navigation probable over entire surface.	Landing accuracy depends on accuracy of surface feature maps.

The Apollo landers used a combination of Earth-based radar, crew recognition of local features, space sextant work, and inertial navigation to achieve an impressive accuracy. In addition, the vehicles had radar altimeters, and radars measured relative velocity. The radar altimeter was used to determine certain checkpoints later in the program. The crew always did the landing navigation visually.

Table 2 shows a variety of possible systems for updating the onboard inertial system and accomplishing landing navigation, including the terrain matching and transponder system. The advantages and disadvantages of each are discussed. All these systems can be related to similar Earth-based systems.

ENGINE BLAST

The effects of engine exhaust blasting the lunar soil are far reaching. Blast from the lander engine will affect virtually every aspect of lunar base design. While the effects will not present insurmountable problems, serious consideration must be paid to them in the design of nearby facilities. The distance between the landing pads and surface facilities and equipment, especially the base itself, will depend on how far away blast damage can occur. The design and protection of equipment that must remain in the vicinity of the landing pad will be governed by how serious the damage from blast will be. When permanent reusable landing pads are needed, the stabilization of those pads will depend on the expected impingement of engine blast.

In addition to being far reaching, blast effects are probably the single most complex to analyze of any affecting pad designs. The analysis prepared for this study was a rough order of magnitude calculation. Many assumptions and simplifications were made. Where needed, they were made as conservatively as possible. Comparison to known data and effects were made where information is available. The nature of the rocket plume was quantified using data provided by Alred (J. W. Alred, personal communication, 1982). These data characterize the exhaust plume of a small engine that is scaled up to an engine the size of the 50,000-N lunar module (LM) engine. The effects of

backpressure were not included. Calculations are broken into four sections: (1) lofted particle sizes; (2) lofted particle trajectories; (3) particle flux at a distance; and (4) particle damage.

Lofted Particle Sizes

Lofting of surface particles is assumed to occur by stagnation of plume flow directly under the particle. The vertically upward force resulting from this pressure is balanced against the vertically downward gravity force and the angled drag force caused by direct impingement of the plume. Maximum particle size for the landed configuration is 5 mm. Particles in the 75 μm or less category, which make up 50% of the soil, can be lofted from a lander altitude of 15 m to 20 m. This is generally consistent with Apollo data, which show that dust usually first appeared at 15 m. Variation of the maximum sizes with respect to thrust variations is nearly linear. A fivefold increase in thrust to 250,000 N shows that rocks of up to 25 mm may be lofted, although they do not go far.

Lofted Particle Trajectories

Particle trajectories were found by assuming that ejection of particles occurs by direct drag acceleration of particles in the plume. The ejecta trajectory calculations from the baseline engine show the maximum distances and velocities shown in Table 3.

Figure 3 shows graphically the ballistic trajectories of the particles after they leave the plume. The trajectory data are generally consistent with the findings of *Cour-Palais et al.* (1972), which, based on Apollo 12 and Surveyor interaction, indicate that particles with velocities in the neighborhood of 100 m/sec were ejected from the engine blast. Increases in thrust result in roughly linear increases in distance and velocity increases that are proportional to the square root of thrust increase.

TABLE 3. Landing blast ejecta.

Particle Diameter (mm)	Impact Distance (m)	Impact Velocity (m/sec)
4.0	20	10
2.0	40	15
1.5	50	20
1.0	75	25
0.5	150	35
0.25	325	50
0.075	1200	100
0.050	2000	125

Particle Flux

Particle flux will obviously vary with the inverse square of the distance from the lander. The original flux was calculated assuming a percent surface obscuration due to particles and converting this to a number of some sized particles. The calculations were made using 50-μm particles and 50% obscuration. This provides conservative estimates of the number of surface impacts due to ejecta flux. In general, at 50 m over 30,000 particles per cm^2/sec can be expected. If larger particles are included, fewer impacts can be expected. At 200 m the flux drops to around 2000; at 2 km the flux is below 50. The flux will vary with the square root of power increase, so a fivefold increase in power will only roughly double the flux at a fixed distance.

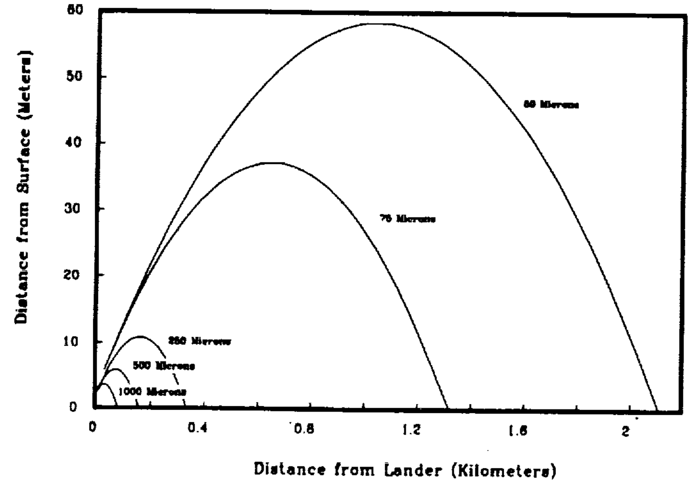

Fig. 3. Lofted particle trajectories.

Particle Damage

Finally, particle damage to surface facilities and equipment can be assessed using the calculated flux, velocity, and size data. Cratering by the low velocity impacts can be studied with known relationships such as those presented in *Wilbeck et al.* (1985). For the purposes of this study, cratering by ejecta on aluminum and glass surfaces was considered. To evaluate net effects of impacts on surfaces, the flux of 50 μm particles calculated above was used. A typical final 10-m descent should last approximately 5 sec. Combining this with flux data, the number of impacts per landing can be found. From crater diameters, surface areas of each crater may be established; thus, the percent of the surface area pitted by craters for each landing can be established. Figure 4 presents the effects for both surfaces with respect to distance from the landing event.

At 50 m an aluminum surface can be expected to have about 5% of its area covered by pits after one landing. This generally will not affect surface properties unless high reflectivity is needed. Glass at this same distance can be expected to have all of its surface pitted. Generally speaking, this will ruin optical-quality glass surfaces. Some pits resulting from bigger ejecta could achieve depths as high as 0.1 mm, easily visible to the naked eye.

At 200 m, about 0.5% of an aluminum surface will be pitted. This is only minor damage. If degradation of the surface radiative properties is not at issue, aluminum surfaces should not present problems even after numerous landings. Glass, however, can have as much as 10% of its surface pitted after a single landing event. For optical instruments, this will be unacceptable. Pit depths of 0.03 μm are possible. This would not ruin vision glazing until several landing events had taken place.

At 2 km, the aluminum surface will sustain virtually unnoticeable damage. Reflective surfaces will degrade after numerous landings and should be protected. Glass surfaces will sustain about 0.1% surface pitting. This will be unnoticeable in vision glasses after a single event but may show up as haze after several landings. Optical-quality glasses should certainly be protected.

SURFACE OPERATIONS

During early operations, landing facility activity will be coupled closely with overall base operations. Lunar surface operations will use the lander/ascent vehicle as a hub, and crews will live in the

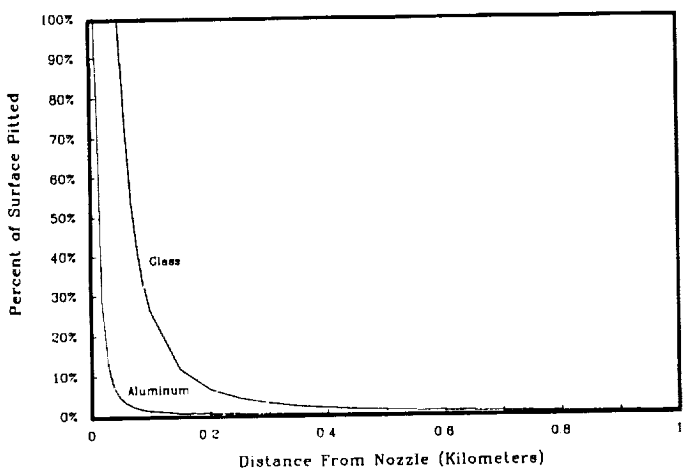

Fig. 4. Single-event surface damage.

vehicle. Because of this, the landing pads will tend to be as close to the base as possible. The first crews will arrive on the lunar surface and select or verify a base site with an area suitable for landings nearby. They will place remote navigational aids and lay out the additional pads needed. The number of pads will actually depend on the scenario, but it should be sufficient to handle all landings up to and including the next piloted mission. Subsequent crews will do the same, except they will not need to place the remote navigation aids. Vehicle off-loading will take place when appropriate according to mission plans. When each crew leaves, sensitive surfaces and equipment installed at the base will be protected from the blast of the next landings as appropriate for each case.

The general area of the temporary pads will be selected by crews near the end of the early landing site development stage. They will move the remote markers at the same time to accommodate the new landing location. Each crew will lay out at least a sufficient number of landing pads to accommodate all the missions up to and including the next piloted mission. To minimize the effects of blast and to eliminate danger to base facilities from landing errors, the pads will be located away from the base. Mission planning may indicate that all temporary pads may need to be marked during one mission. Crews will move to the base after arrival on the surface. Until pressurized transfer from vehicle to base is available, EVA will be needed to get crews into the base. This will necessitate careful mission planning to ensure that every EVA hour is used appropriately. Since the stay times for a temporarily occupied lunar base may be significant, the vehicles must be provided with survival support including power to operate systems, supplemental cooling to accommodate the extra loads from the lunar surface, and meteoroid protection. Crews will unload cargo vehicles as indicated by the mission plans. When each crew leaves, they will protect the equipment left behind near the pads, such as surface transportation vehicles, from the blast of the next landers. In addition, some of the equipment, instruments, and facilities left at the base may need protection.

At the end of the temporary stage, the best site will be picked by the crews, and the pads will be leveled and stabilized. These pads will be marked using the standard markers. Since the temporary and permanent pads will be close together, the remote markers may be left where they are. Depending on the availability of pressurized transfer, the crew may or may not need EVA to get into the base itself. In addition to offloading vehicles, the reusable pads will need to be cleared of empty cargo vehicles and expendable lander platforms. Piloted vehicles will be provided with survival support for the long stay on the surface. Some vehicles may require loading and servicing. The activity of the crew as they leave the lunar surface will depend on whether the base is permanently occupied or not. Temporary occupation will indicate the same preparation as needed for the temporary stage. Permanent occupation indicates the same sort of preparation but may also require suspension of some ongoing activity such as EVA operations.

FACILITY ELEMENTS

From examination of the surface operations, the elements needed for the launch and landing facilities may be ascertained. Many of the elements used as part of the launch and landing facilities will be used to support other lunar base operations. In general, these relate to transportation of crew and cargo and to construction-related activity such as surface grading and equipment handling. Some elements are truly unique to the launch and landing facilities. The elements of the launch and landing facilities described in the following section are generally unique to the facilities.

Landing Pad

The most obvious and indeed most important of the site facilities is the landing pad itself. Two basic types of pads must be designed: permanent reusable pads for later base development stages and nonreusable, unprepared pads for early use. Several issues combine to define the degree of surface preparation and refurbishment needed, the size and configuration of the landing pad, and the distances at which other base elements must be kept.

The stage of lunar base development affects two aspects of landing pad design: pad preparation and pad location. Unprepared nonreusable pads are appropriate during early stages of base development when surface crew time is at a premium. The maximum distance between the base and landing pad is 250 m to 400 m before base habitation is possible, since crews must be able to easily walk between the vehicle and base site. After base habitation and until highly reliable surface transportation is available, the base and landing area must be within maximum crew walking range, so 3 km to 5 km is the maximum separation distance.

Surface slope and obstacle characteristics affect the degree of landing pad preparation required. Landing area selection efforts, the degree of pad preparation, and lander capabilities can all be traded against each other. As a first-order discussion of these trades, the Apollo lander capabilities will be assumed. Lunar base site selection must be done for an area at least large enough to handle all planned landings as well as gross navigation errors. This area may be as large as an ellipse 14 km by 6 km or greater, typical of Apollo missions. Unprepared landing pads can be located within this area with only modest amounts of *in situ* inspection by crewmembers. Adequate sites were found by Apollo astronauts within several minutes from some 10 km away while the LM was in flight. When precise alignment of surface systems with vehicle systems is required, level landing pads are needed. For example, placing a large cargo in a set of trunnion attachments will require significant alignments. If the series of fittings is not near

horizontal, proper attachments to all fittings at one time will be difficult, time consuming, and dangerous due to the possibility of cargos coming loose. Significantly off-horizontal landing configurations may present unacceptable requirements for cargo loading and vehicle servicing equipment.

Landing errors affect the size of the landing pad and the distance between the pads and base. Landing pad size should be about 100 m across. The Apollo nominal 3σ landing areas were about 2000 m across, assuming good navigation system updates from landmark recognition and Earth-based tracking. The additional aid of site-originated, precise, and continuous navigation system updates will be available for lunar base landings. This precise position data coupled with the maneuvering capability experienced in Apollo 17 should easily allow the 3σ landing area to be reduced by an order of magnitude to 100 m. There is a risk of the vehicle landing in an area 100 m from the target landing spot. Consequently, equipment and facilities located within 150 m to 200 m of the target are at risk of the same damage they would experience if they were located on the pad itself. Because of this risk, the base and related equipment should be at least 200 m to 250 m from the landing pad. During later stages the landing pads should be at least 3 km away from the base to remain outside the landing ellipses.

Lander and pilot visual and radar resolution will mainly affect the distribution of pad markings. Markings may be placed at the apexes of a triangle inscribed within a circle 100 m in diameter. The placement of three pad markings on this circle in a triangular pattern will result in separations of about 90 m. This presents a 1° separation at 5 km and should provide adequate resolution for final approach and landing sequences. Apollo landing operations only allowed direct line-of-site viewing at 8 km. This should be sufficient for piloted landings and present little or no problems for radar guidance, assuming transponders are provided.

Blast effects will dictate the distance between the landing pads and surface facilities and equipment, especially the base. The interaction of the blast with the lunar soil was described previously. From 0 m to 50 m, metal objects will experience significant surface damage, and glass surfaces will experience severe damage. From 50 m through 200 m to 400 m, metal objects will experience only minor pitting after one landing, while glass surfaces will experience significant damage. From 400 m past 2 km, metal objects will sustain only very minor and probably unnoticeable pitting damage after numerous landings. Glass surfaces will sustain minor damage after numerous landings. The damage will eventually be unacceptable for optical-quality glasses. Optical instruments should face away from landings.

The conceptual designs of the landing pads resulting from accommodation of these issues are shown in Figs. 5 and 6. A reusable pad will have a flat, leveled and stabilized surface inside a 100-m diameter. Surface stabilization techniques will be described later in this report. The pad will be marked by three markers on the circle. The slopes within this area should be as close to 0° as is practical and certainly not over 1°. These slopes will allow easy alignment between surface and flight vehicle systems so complex surface support activities can take place. An area 200 m in diameter should have slopes not greater than 4° so that small dispersions can be accommodated with little off-nominal surface support efforts. Usable items should be outside a 250-m radius to prevent damage from stray ejecta that may break away from the pad. The pad should be located 3 km from the base to accommodate 3σ landing dispersions determined for gross navigation update failures, for crew safety during permanently occupied operations, and to minimize blast effects on the base.

An unprepared pad will be of the same dimensions and markings as the reusable pad. Slopes of 6° over 20-m distances, and 1-m humps and depressions are acceptable. Boulders over

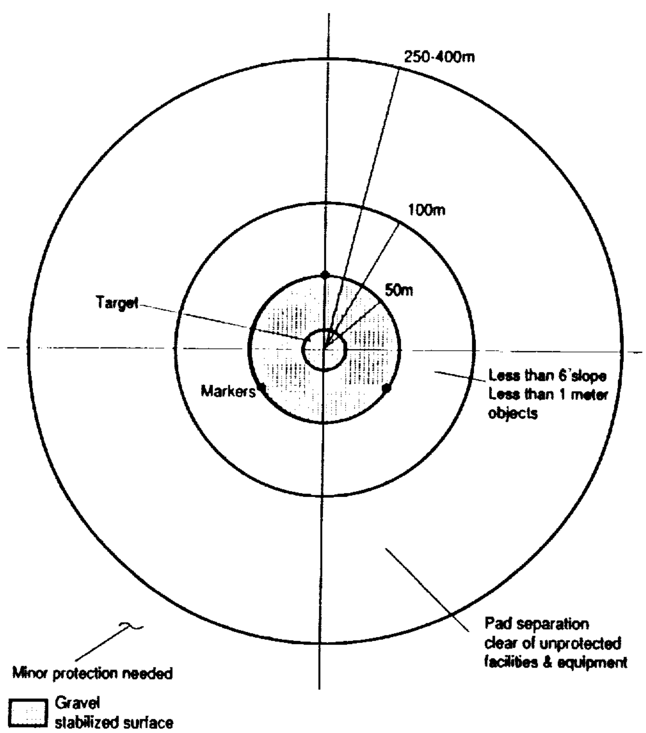

Fig. 5. Permanent landing pad.

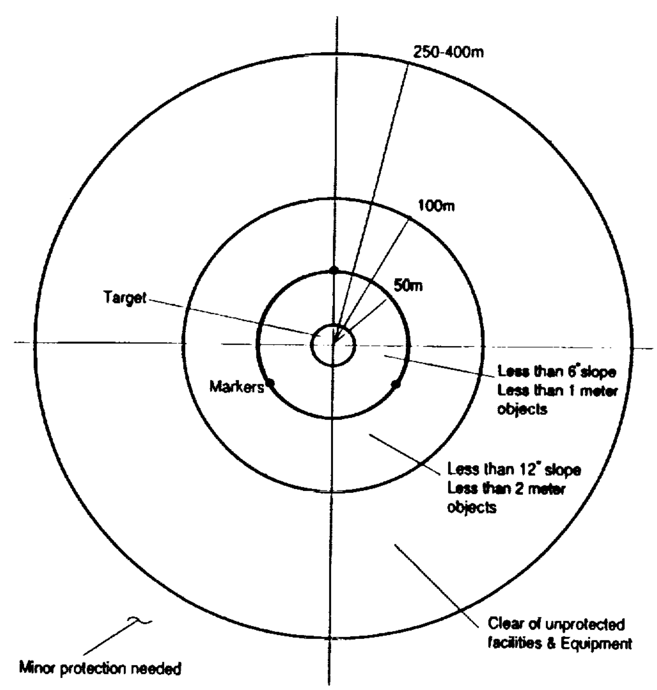

Fig. 6. Unprepared landing pad.

about 0.5 m should be eliminated or avoided to provide footpad stability and clearance for descent engines. The 200-m area should have no slopes over 12° and no humps or depressions over 2 m in relief. Slope restrictions are based primarily on landing stability limits in this case instead of surface support interface requirements. Pads may be located as close as 250 m to 400 m from the base and each other. However, at these distances precautions must be taken to protect reflective and optical surfaces on base equipment. When the base can support habitation, the pads should be located 3 km from the base. In addition to accommodating safety and navigation errors, this distance relieves some of the facility and equipment surface protection precautions.

Surface Stabilization

Surface stablization will be required once the conditions for the establishment of reusable landing pads occur and area grading has been accomplished. This stabilization reduces the amount of pad refurbishment required between landings, reduces or eliminates ejecta, and provides easier surface transportation and more consistent roadway surfaces. There are several methods for stabilizing the lunar surface. Paving tiles, gravel, and simple compaction represent three methods of various degrees of complexity of the setup equipment and operations, and the extent of maintenance operations. The results of these trade-offs seem to indicate that deposition of either natural gravel or man-made gravel is the best surface stabilization method.

Paving tiles, depending on the tile design, offer the best overall surface. Maintenance of the surface is virtually nonexistent, but paving tiles are very difficult to set up. Simple compaction, at the other end of the spectrum, offers the lowest quality surface. Even though soil cohesion is high, fine particles are still exposed to lander blast and to wheeled and foot traffic. This will eventually result in blast ejecta and dust problems.

In addition, maintenance of surfaces will be the same as initial setup operations, since the surface will require releveling and recompaction after exposure to traffic. Finally, gravel provides a good, although not superior, surface. The surface is not as stable or easy to travel on as paving tiles, but fine soil particles are not exposed to lander blast or surface traffic. Proper selection of gravel sizes should provide roads and pads that are well within acceptable specifications. Gravel is readily available from natural screening or as a by-product of the resource utilization processes, which will just precede the need for stabilized surfaces. Leveling and spreading of gravel surfaces can be accomplished easily by the same operations used for leveling the surfaces below them. Maintenance may involve periodic leveling of gravel surfaces, but these operations should be minimal if gravel sizes are selected appropriately. In short, gravel deposition surface stabilization provides adequate surface characteristics without the need of any signficant unique equipment and without the need for exotic operational activity. Figure 7 shows the three types of stabilization.

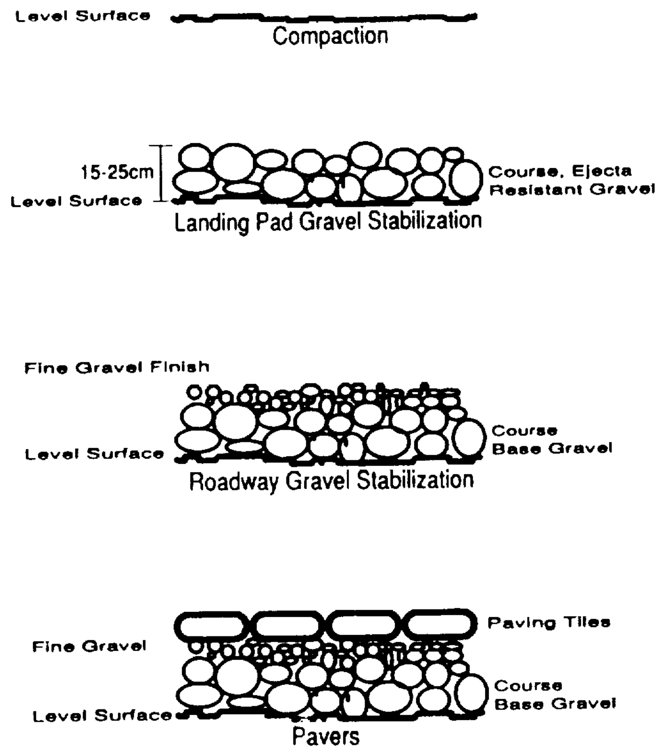

Fig. 7. Surface stabilization.

Blast Barriers

Blast barriers are used to protect equipment from the effects of the ejecta from the landing events. There are two primary philosophies for the design of these barriers. First, blast barriers can be erected as permanent structures close to the landing pads. Second, smaller temporary or permanent structures can be erected at individual equipment locations to shield small areas from the effects of the ejecta. Examination of the nature of the blast and the effects of small off-nominal landing conditions indicate that the second philosophy of protecting equipment and facilities is the most desirable. Figure 8 shows some of the methods of local blast protection.

Close barriers must be tall enough to block the bulk of the particles and yet must be far enough away so as not to represent hazards for off-nominal landings. Blast calculations above indicate that at 50 m, maximum particle altitude is 7 m, and at 100 m particle altitude is 12 m. Barriers 7 m to 12 m high are major items. With these heights, it is safe to assume that the barriers must be made from local resources such as piles of soil or gravel. A soil barrier 12 m high, beginning at 50 m and peaking at 100 m, will have a slope of 13° (only marginally acceptable) and will be a considerable construction project.

Local equipment and facility barriers appear to provide easy forms of surface protection for modest efforts and minimal weight penalties. Several methods are available depending on the particular application. First and most simple is careful orientation of equipment so that sensitive surfaces face away from the landing. If this proves unfeasible because equipment cannot be moved, installation of a barrier will be needed.

For glass surfaces, two methods may be considered. If the surface must be used to view the landing event, double glazing should be used such that the outer layer is easily replaced once surface erosion has progressed too much. If the viewing is not needed during the event, a movable opaque shield can be installed. This could consist of thin plastic or aluminum sheets.

For equipment with complicated geometries and extensive sensitive surfaces, covering by a blanket or erection of a vertical barrier may be used. Blankets of mylar or lightweight fabrics provide the simplest method of protecting sensitive equipment that is not used without crewmembers. A shield such as metal

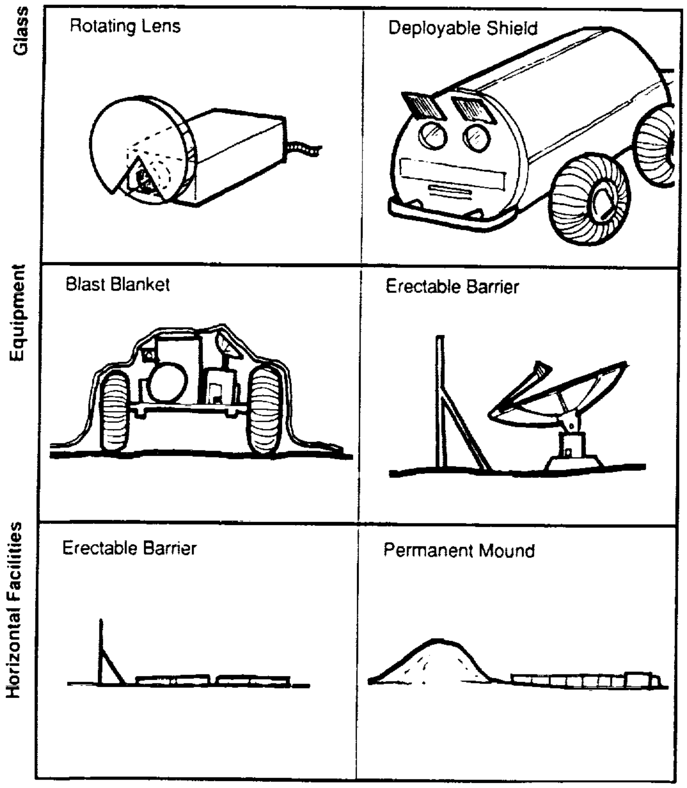

Fig. 8. Blast barriers.

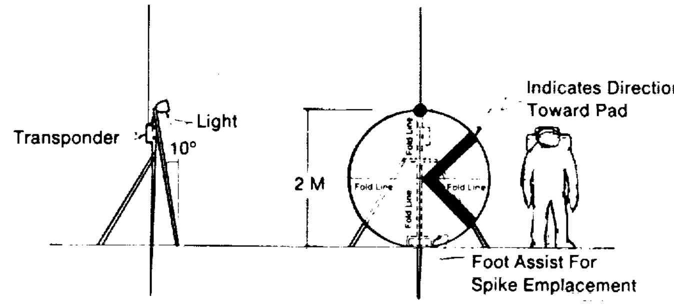

Fig. 9. Landing pad marker.

plate or fabric stretched on a frame of suitable size can be easily leaned against or propped next to equipment that must remain active during the absence of crew.

Pad Markers and Navigation Aids

Pad markings and navigation aids are present to assist flight crews and automated landers in locating the landing pads and in adjusting trajectories to ensure precision landings. Visual marking is intended to provide identification of the pad to the crew for piloted missions. Navigation aids are intended to provide visibility to automated guidance systems.

Figure 9 shows one possible device to serve as a pad marker with a radar transponder. The marker should have stowed dimensions of 50 cm × 50 cm × 10 cm and a mass no greater than 10 kg. The device contains a transponder, a visual marker, and a laser range finder. These markers are placed at three positions on the 100-m diameter of the landing pad as discussed for the landing pad design. In addition, two of these markers are placed at about 1.5 km downrange and 1.5 km crossrange from the landing site. The two will be visible above the horizon, both from each other and from the landing pad. These long-range transponders provide detailed navigation data to the lander guidance system. They will show 1° separation at 90 km at which point the base will be visible to the lander and the lander will have plenty of time to make needed course corrections. Three markers are needed for each pad along with the two downrange and crossrange. Each crew will generally set at least two pads for a subsequent cargo and piloted landing. As a result, the first crew will need eight markers. The two long-range transponders will be set once and left in place. The three pad markers will be set each time a pad is selected, whether for unprepared pads or reusable pads.

Crew Access

Methods for transferring crewmembers to and from vehicles can be extremely simple. The initial method will be via extravehicular mobility units (EMUs) already carried by crews for other purposes. This method will be considered the trivial case, since only a ladder will be needed. Of primary interest is transfer between two pressurized spaces. The employment of IVA transfers will relieve operational issues such as mission planning for EVA on the first and last days of the surface stay.

Several concepts are available for accomplishing pressurized transfers including rigid and flexible tunnels, systems fixed to either the lander or the pressurized surface vehicle, and independent systems. One concept involves a dedicated ramp vehicle similar to the mobile stairways used for airline passengers. The difference is that the lunar version would be pressurized. After the landing, the ramp approaches and connects to the landed vehicle. Soon afterwards the rover vehicle attaches to the other end of the ramp. Crewmembers then pass from the lander to the rover, reseal the ramp, and depart for the base.

Figure 10 shows one concept for a ramp-type transfer tunnel. The tunnel ramp is basically a trailer with a special pressurized tunnel and universal docking adapters/hatches at both ends of the tunnel. The tunnel ramp is estimated to have a mass of about 3 t. The wheels will be powered so that the ramp may be operated independently. It can either be controlled by connection to the pressurized rover itself, or it may be teleoperated. The ends of the tunnel are flexible so that it can mate with the unlevel docking adapters of the lander and rover. It is anticipated that the height difference between the rover and the lander hatches will be approximately 2 m from center to center.

Cryogenic Transfer

Cryogenic storage equipment is needed for resource utilization activities in which liquid oxygen or hydrogen is produced in quantity on the lunar surface and is used in off-surface operations. Options for transfer could involve either permanently installed lines from storage equipment to pad locations or transfer vehicles with tankage for transfer. Since the vehicle needed for transfer can also be used for filling the storage facilities from plant

supplies, vehicles can easily be designed to have the same connections. Installation of permanent lines to each pad will be major operations and beyond a Phase II lunar base.

Figure 11 illustrates a propellant refill vehicle (PRV) that represents one concept for performing fuel transfer. The PRV consists of a storage tank for either liquid hydrogen or liquid oxygen, the necessary support equipment to transfer the fluid to a flight vehicle, and the required hardware to run the vehicle. The PRV is used for filling and draining dedicated tanker vehicles with fixed tanks, filling propellant tanks of a reusable vehicle, and scavenging unspent fuel from landers.

The propellant tank for the PRV is 3 m in diameter and has a 3-m long cylinder with spherical ends. This allows it to carry 35 cu m of propellants, which is equivalent to about 2500 kg of liquid hydrogen and about 40,000 kg of liquid oxygen. A boom with flexible propellant lines is included with the PRV to

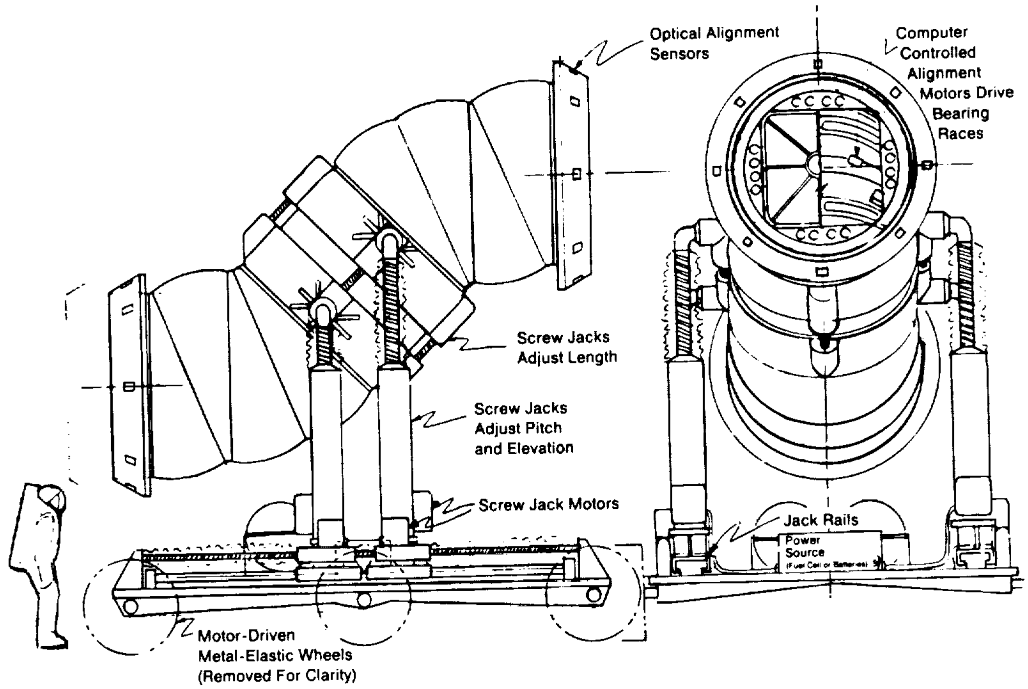

Fig. 10. Crew transfer tunnel.

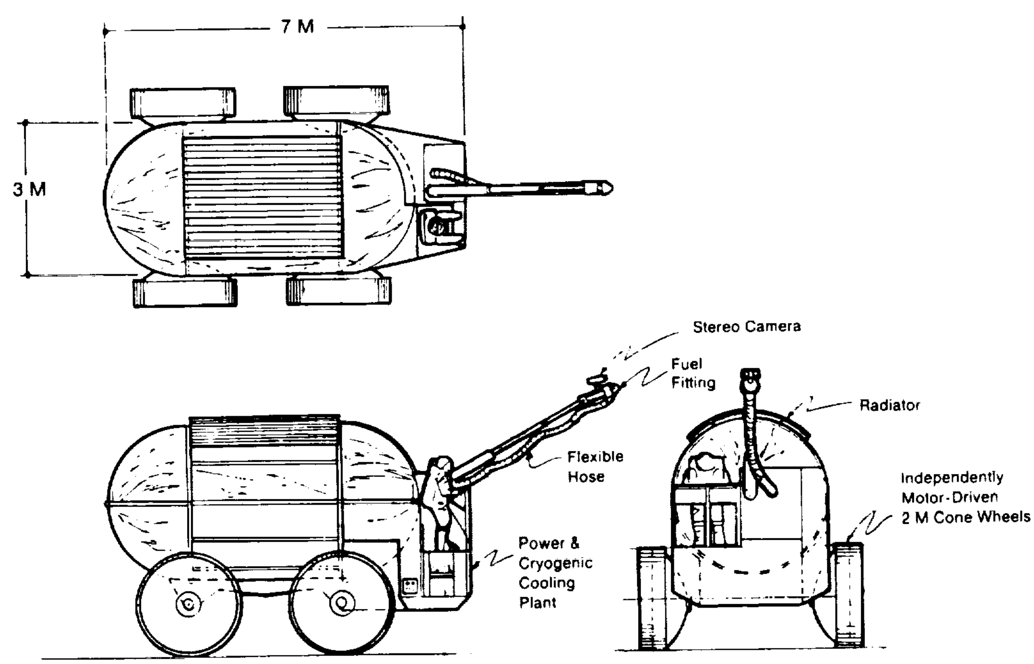

Fig. 11. Propellant refill vehicle.

accomplish fluid connections. The base of the fluid transfer boom is anchored to the front deck of the vehicle. The crewmember is situated at the base of the boom from where he controls boom positioning during propellant transfer manuevers or controls the vehicle while traversing to the landing site. The fluid transfer nozzle is positioned by rotating the boom base and extending the telescoping boom elements. For accurate positioning, fine adjustments are made at flexible joints near the nozzle before mating to the lander. While the PRV is in motion, the boom is stored in the collapsed position.

No serious attempt has been made to find the mass of the PRV, but estimates are that it will mass 14,000 kg empty. This includes an estimated 10,000 kg for the tank, 2000 kg for the structure, power, locomotion, and other subsystems, and about 2000 kg for the refrigeration and radiator system.

Power Supply

Electrical power is a vital utility for piloted vehicles on the landing pads. The vehicle systems must be kept in working order, and appropriate overall vehicle thermal conditions must be maintained. Although these vehicles will have their own onboard power systems, the lunar environment is significantly different from that of space, and mass considerations may limit electrical energy storage capabilities. Without performing detailed study, it is evident that some sort of supplemental power supply for long surface stay times will be justified.

There are two basic ways to provide the needed supplemental power to the landing pad. One involves the use of an electric cord extended from a central base power system and the other a self-contained portable power supply. Some baseline requirements must be established to allow comparison of these two types of systems. To that end, it is assumed that the lander will require 2 kW of power for a period of 28 days. For the application described, the possibility of an inaccurate landing some kilometers away from the planned site, along with other versatility needs, will weigh heavily toward the self-contained power supply. If vehicle surface stay times increase, the balance may be shifted towards the cord system. This will occur for alternate ascent stage concepts in which the crew leaves the Moon in the vehicle used by the last crew providing complete ascent stage redundancy. Figures 12 and 13 show drawings of both type of systems.

The cord system consists of a 1-km long cord on a spool that is mounted on a four-wheeled cart. A power conditioning system consisting mainly of a transformer and rectifier is available on the cart to provide a variety of voltages including the standard 28V DC spacecraft electrical power. The overall mass of the system is estimated at 910 kg. Table 4 provides a mass breakdown and dimensional data. When needed, the cord is plugged into the base power system and unreeled to the site needed. Another cord can be connected between the vehicle and the power supply, and the lander will have the needed power. If additional distance is needed, another extension cord can be connected to the first, bypassing the transformer system.

There are several options available for the portable self-contained system. Among them, fuel cells and nuclear isotope generators appear to provide the best possibilities. Batteries will not be examined for this system, since the storage requirement of nearly 1500 kWhr will result in a massive system. Masses as low as the 5 kg per kWhr of zinc-silver batteries would result in a 7.5-t system. In addition, solar cells will not be considered as a primary power supply. Since the system must be operated during

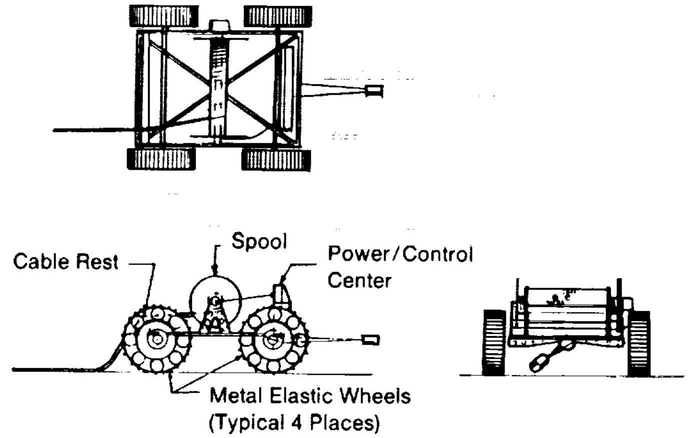

Fig. 12. Electric cord system.

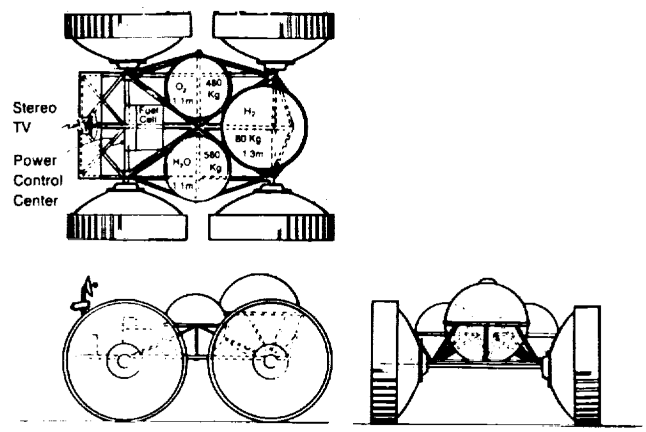

Fig. 13. Fuel cell power cart.

the lunar night, solar cells cannot be used for continuous power. These solar cells can be used as a source supplemental to the primary power generation system. Nuclear systems use technology that is not well known, and they involve some difficult political and safety issues. As a result they will not be considered here. Fuel cell technology is well developed, and application to the space shuttle and previous programs has proven it to be an operational technology. As a result, a fuel cell system is proposed for the self-contained power supply or "power cart."

The power cart consists primarily of cryogenic hydrogen and oxygen tanks, liquid water tanks, and a fuel cell system mounted on a four-wheeled cart. A solar cell can be mounted on the cart to provide extra power during sunlight periods. The estimated mass of the fuel cell power cart is 1290 kg. Table 4 provides a mass breakdown and dimensional data for this system. When a lander needs power, the cart is taken to the landing pad. The power cart is connected to the vehicle in the same way as the electric cord system. The fuel cell is then activated, and the

TABLE 4. Vehicle power supplies.

Electric Cord System (1 km)	
Conductor	490 kg
Insulation	250 kg
Power Conditioner	20 kg
Cart	90 kg
Total	820 kg
Dimensions: 2.0 m long; 1.4 m wide; 1.1 m high.	
Fuel Cell Power Cart (2 kW, 28 days)	
Tanks	
Hydrogen	190 kg
Oxygen	130 kg
Water	130 kg
Fuel Cell	90 kg
Solar Panel (1 kW)	40 kg
Cart	150 kg
Dry Mass	730 kg
Reactants	560 kg
Total	1290 kg
Dimensions: 4.3 m long; 1.3 m wide; 1.3 m high.	
Tanks	
Hydrogen	1.3 m diameter
Oxygen	1.1 m diameter
Water	1.1 m diameter

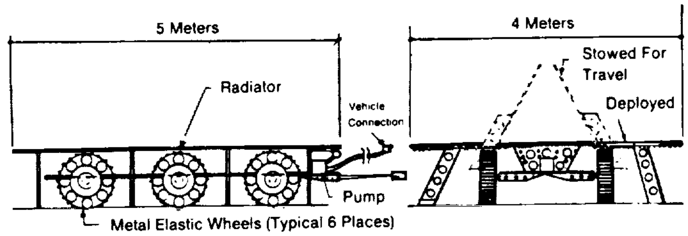

Fig. 14. Supplemental cooling cart.

TABLE 5. Supplemental cooling system.

Radiator	340 kg
Pipes	390 kg
Pump	20 kg
Cart	190 kg
Water Working Fluid	230 kg
TOTAL	1170 kg

vehicle has the appropriate power. After use, the cart can be taken back to a central regeneration station where it is charged for its next use.

Both the cord and cart systems have compelling and complementary advantages. Both systems can be used for many tasks other than simply supplying power to a lander. There will be need for power away from the base for a variety of transportation, construction, and other tasks, as well as for vehicle maintenance. Because of these needs, both types of systems are recommended. In fact, more than one of each may be required depending on how many simultaneous tasks are undertaken.

Supplemental Cooling Cart

For reasons analogous to the need for electrical power, a supplemental cooling system will be needed for piloted vehicles on the landing pads. The vehicles and their systems must be kept cool during the lunar day when reflection and reradiation from the lunar surface will add to the direct sunlight experienced in space. These vehicles will have their own onboard cooling power systems sized only for direct solar heat loads. A supplemental cooling system (SCS) will add radiator surface for the lander cooling system to allow it to handle the additional cooling loads of the lunar day.

The SCS will consist primarily of a radiator sized at a minimum to reject the added cooling load from the lunar surface and at a maximum to reject the entire vehicle cooling load. Since these loads are unavailable at this time, a load of 2 kW is assumed. The radiator can be sized at 2 kW for an average radiator surface operating temperature of 15°C. At this temperature, estimates of heat rejection are about 100 W per sq m for simple radiators (*Lunar Bases Synthesis Study,* 1971). At this rate, the radiator will be 20 sq m or about 4 m x 5 m.

The SCS shown in Fig. 14 has a deployable radiator system in three sections. The system mass is about 1170 kg. Table 5 provides a mass breakdown and some dimensional data. The system is mounted on a cart similar to the one used for the fuel cell power cart described above. This is a simplistic radiator system. Other more sophisticated radiator designs have been proposed for applications such as this. The design presented here is intended to provide a conservative, rough order of magnitude size and weight. Further detailed design must be performed once better data on the expected heat load are known. Coolant choice must also be considered to ensure proper operation over the entire range of surface conditions.

Micrometeor Protection

It is probable that some vehicles that will remain on the surface for long periods will need to be protected from exposure to micrometeors. One concept for providing this protection is the use of a vehicle cover or blanket that can be draped over the entire vehicle or over selected systems sensitive to the expected micrometeor bombardment. These blankets would serve as bumpers supplemental to those already provided on the vehicle itself. Blankets such as this will be needed for blast protection. The same sort of material can be used. Multilayer mylar sheets or kevlar fabrics may provide appropriate protection.

SITE DEVELOPMENT

The evolution of lunar base landing facilities can be summarized in what will be known as a Site Development Plan. This plan must be meshed with other plans for lunar base development to ensure that appropriate facilities and equipment are available when they are needed. The Site Development Plan will indicate how and when the facility elements defined above will be used at the launch and landing facilities. The needs and evolution are translated into particular schemes. Generally known as "scenarios," these objectives, goals, and schemes are dynamic. Many scenarios for lunar development have been proposed and continue to be proposed. Scenario development and evaluation is a current and continuing process; thus, it is obvious that no one Site Development Plan may be proposed with hopes of it being valid for long. Each lunar base scenario must have its own Site Development Plan.

The primary interest in this planning is to affect the evolution of the lunar base in only modest ways if at all possible. This approach allows delivery schedules and crew activities to relate to the objectives of the base itself and not to a sideline effort such as development of landing facilities. The development of lunar landing facilities for a Phase II lunar base follows one general path. There are three stages along this path: early landing facilities, temporary landing facilities, and permanent landing facilities. Depending on the nature of the individual scenario, the length of any of these stages may vary. However, the activities within each stage are the same no matter which scenario is chosen. Figure 15 illustrates layouts of landing pads with respect to the base for the three development stages.

Base Objective Dependence

The objective of the base will affect primarily the transition from temporary to permanent facilities, although the early stage can be affected indirectly. The main dependence is derived from cargo operations and the need for cargo loading and alignment operations. Support for a scientific base can generally be characterized by the need for instrument and construction equipment, logistic resupply deliveries, and sample returns. All other things aside, if the sample return requirements are low, permanent, reusable landing pads may never be needed. A resource-oriented base will have an obvious export. While the specific operation is not of importance here, when the export

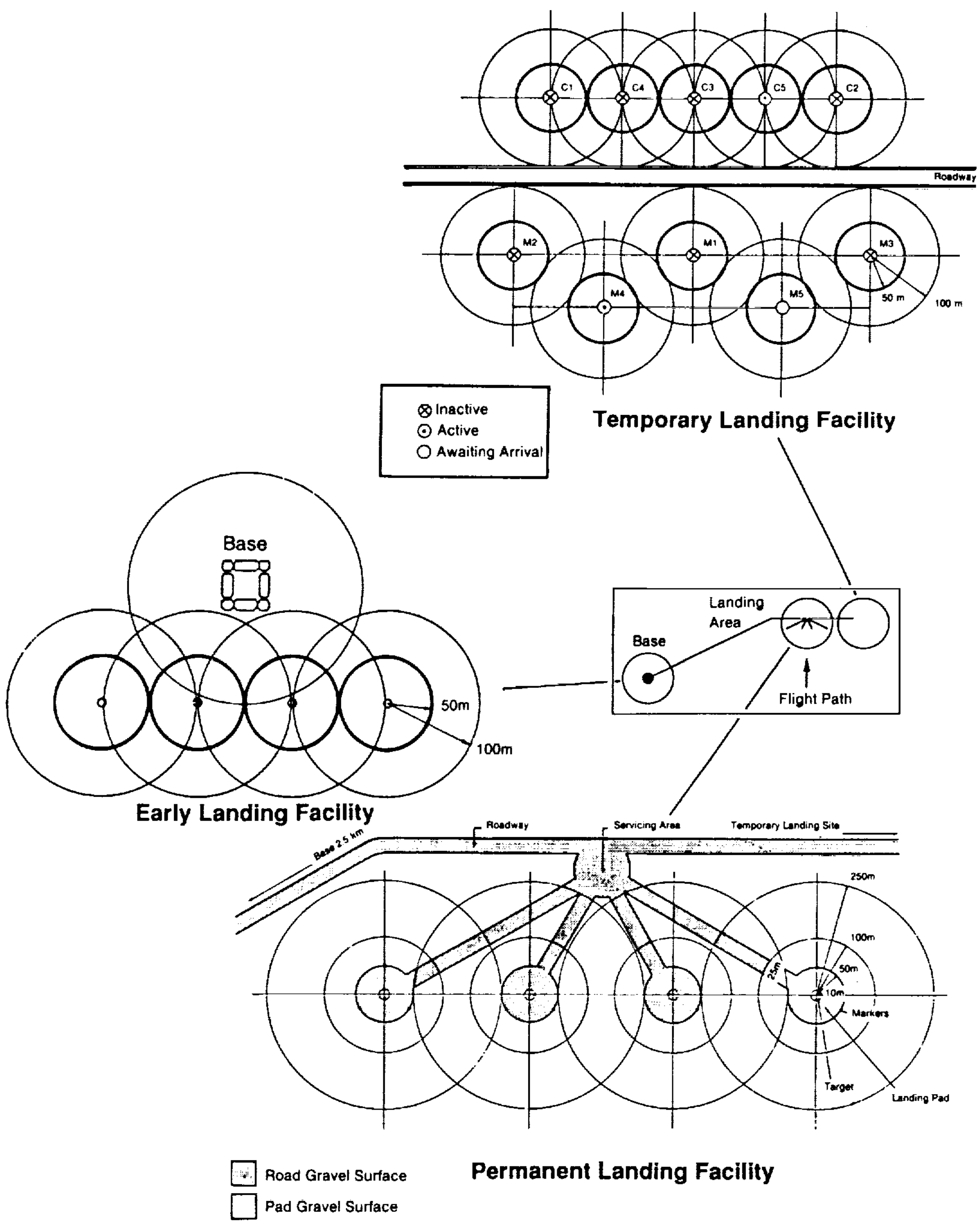

Fig. 15. Landing pad layouts.

activity begins in earnest, permanent pads will be needed and the transition from temporary to permanent stages will occur. A habitation base alone will, in general, not require a permanent landing facility. Since no product is shipped from the surface, no major cargo loading takes place.

Base Growth Dependence

The rate of growth of habitation facilities and the growth of surface stay times affect landing facilities in different ways and at different periods. Habitation growth relates directly to the early temporary stage transition. If habitation is important, the base will be rapidly developed to allow dwelling in the base. At this point, as long as some sort of vehicular surface transportation is available, the pads may be moved away to the remote sites and the temporary stage can begin. If the base is developed slowly, the early stage will be protracted, and the vicinity of the base may actually become littered with spent stages and used landing pads.

Stay-time growth will affect the transition from temporary or early stages to permanent stages. When surface stay times increase to the extent that reusable pad setup and maintenance becomes a fairly small percentage of available time, the permanent stage can be justified. Although not necessarily, this is usually associated with permanently occupied operations and would be very near the end of Phase II operations.

Flight Vehicle Dependence

The specific design of the lunar lander will affect pad location, equipment protection, and servicing requirements. These effects are related to the size and thus the thrust levels and the expendable vs. reusable nature of the vehicle.

If growth of flight vehicles increases or decreases the size of the engines, some change to site development may be indicated. Generally, ejecta from larger engines will be larger and travel farther and faster than for smaller engines, and facilities will need to be spread out.

The use of a reusable lander will affect the transition to permanent stages and the nature of facilities located at the pad. When a reusable vehicle begins to need servicing on the lunar surface, facilities for this servicing will be required. If the nature of the servicing is such that simple EVA is unacceptable, whether because of crew time or servicing complexity, the transition to the permanent stage must be made. This will occur regardless of the current stage. If the early stage is the current one, the temporary stage may be skipped altogether. If the facilities needed to handle the permanent operations are not available, they must be provided.

CONCLUSIONS

Launch and landing facilities and their growth rate depend on the base development scenario. The major emphasis of the base, the rate of emplacement of facilities, and the design of the flight vehicle will all play major roles in the requirements for facilities. Resource utilization bases will require more and different landing facilities than will science or habitation bases. The more rapidly some base capabilities are achieved, the more rapidly landing facility capabilities are required. Vehicles that require extensive surface-based servicing will require leveled permanent landing areas. These permanent reusable landing pads are not needed or desired before major resource export or vehicle servicing activities take place. For some lunar base scenarios, permanent landing pads may never be needed.

With few exceptions, lunar landing facilities and equipment are present on the lunar surface for other reasons before they are needed for landing operations. Landing equipment and facilities will probably not be major drivers of delivery schedules and mission plans.

Based on the calculations done during this study, the effects of engine blast are significant. While they are not critical or life threatening, they must be considered. Equipment within 50 m of a landing may experience severe damage due to the impact of fairly large grains of lunar soil. Equipment over 400 m away will require only minimal protection. At 1 km to 2 km blast effects are very small.

Landing pads can be designed without general regard to the specific landing site because overall surface conditions are fairly uniform across the entire lunar surface. Landing pads, whether prepared or not, should be about 100 m across. The area just outside this circle to 200 m across should not include any major obstructions such as boulders or expended landers. Lunar-derived gravel may be used to stabilize prepared landing pads.

RECOMMENDATIONS

More work is needed concerning blast effects, vehicle servicing on the surface, site planning and development, and safety and rescue operations. More design definition is needed for surface stabilization methods, cryogen storage and transfer facilities, servicing and maintenance equipment, and other items.

The launch and landing facilities of a permanently occupied base need to be defined. This study was limited to the initial lunar base, and the facilities needed for extensive permanently occupied or Phase III bases have only been reviewed in a cursory fashion.

REFERENCES

Cour-Palais B. G., Flaherty R. E., High R. W., Desler D. J., McKay D. S., and Zook H. A. (1971) Results of the Surveyor III sample impact examination conducted at the Manned Spacecraft Center. *Proc. Lunar Sci. Conf. 2nd*, pp. 2767-2780.

Minutes of Apollo Site Selection Board, February 1972. On file at Lunar and Planetary Institute Library, Houston.

Mitchell J. K., Carrier W. D., Costes N. C., Houston W. N., Scott R. F., and Hovland H. J. (1973) Soil mechanics. In *NASA SP-330*, pp. 8-1 to 8-22.

Moore H. J. and Tyler G. L. (1973) Comparison between photogrammetric and bistatic-radar-slope-frequency distributions. In *NASA SP-330*, pp. 33-17 to 33-26.

North American Rockwell, Space Division (1971) *Lunar Base Synthesis Study: Shelter Design*. Final Report for NASA Contract NAS8-26145, pp. 1-4-4 and 3-1-8.

Wilbeck J. S., Anderson C. E., Wenzel A. B., Estine P. S., and Lindholm U. S. (1985) *A Short Course on Penetration Mechanics*, pp. 15-27. Southwest Research Institute, San Antonio.

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THE CHOICE OF THE LOCATION OF THE LUNAR BASE

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The development of modern methods of remote sensing of the lunar surface and data from lunar studies by space vehicles make it possible to assess scientifically the expediency of the location of the lunar base in a definite region on the Moon. The preliminary choice of the site is important for tackling a range of problems associated with ensuring the activity of a manned lunar base and with fulfilling the research program. Based on astronomical data, we suggest the Moon's western hemisphere, specifically the western part of Oceanus Procellarum, where natural, scientifically interesting objects have been identified, as have surface rocks with enhanced contents of ilmenite, a possible source of oxygen. A comprehensive evaluation of the region shows that, as far as natural features are concerned, it is a key one for solving the main problems of the Moon's origin and evolution.

INTRODUCTION

The main criteria for choosing a site for the first section of a manned lunar base are (1) the most favorable conditions for transport operations, (2) the presence of natural objects of different types in a relatively limited region (the study of these objects may provide answers to the principal problems of the Moon's origin and evolution), and (3) the presence of natural resources, primarily oxygen-bearing minerals in surface rocks, needed for ensuring the base's functioning even at the first stage.

No doubt the final selection of the base's site, as well as the final decision concerning its establishment, will call for new space flights for collecting appropriate information. However, the current level of remote sensing of the Moon and already available data of direct studies by space technology enable one to make preliminary estimates and forecasts of the most reasonable location of a manned lunar base in keeping with the above criteria.

When the lunar base project was first discussed over 20 years ago, the Moon's mare regions were considered the most suitable location for a lunar base (*Shevchenko*, 1968). At present we possess data that make it possible to substantiate this viewpoint in more detail and to determine which of the Moon's mare formations best meets the requirements.

THE MOON'S HEMISPHERIC ASYMMETRY

Upon completing the global survey of the lunar surface the asymmetry of the lunar sphere became obvious from the different structure of the near- and farside of the Moon. This asymmetry is determined by the location of lunar maria concentrated mainly within the limits of the nearside.

Analysis of the structural features and distribution of the surface rocks of the eastern and western hemispheres also showed a sharply pronounced western-eastern asymmetry. The asymmetry of the near- and farside in the complete form appeared about 3000 m.y. ago, when basaltic lavas of most of the modern maria erupted to the surface from the Moon's interior. This structure of the Moon's surface can now be explained qualitatively. In the final period of shaping the lunar crust's upper horizons (this period coincided with the final equalization of the Moon's periods of orbital revolution and axial rotation), the impact of terrestrial gravitation on the internal structure of the Moon increased. Between 4 and 3 b.y. ago the thickness of the solid crust within the nearside became 1.5-2 times lower than the thickness of the crust in the farside. This led to outcropping of lunar rocks predominantly within the limits of the nearside. As mentioned above, the distribution of maria constitutes the directly observed indication of hemispheric asymmetry of the near- and farsides of the Moon.

Asymmetry of the Western and Eastern Hemispheres

Over the past few years the work on the morphological catalog of lunar craters has come to an end. The catalog includes data on all craters more than 10 km in diameter (*Rodionova et al.*, 1985). This information characterizes the Moon's large-scale cratering, which relates mainly to the premare period of the Moon's history. Statistical analysis shows the differences in the density distribution of the craters in the highlands of the western and eastern hemispheres. Figure 1 is a generalizing map of the density distribution of about 15,000 lunar craters more than 10 km across (*Rodionova et al.*, 1988) and shows that the highest density distribution (120-150 craters per 10^5 km^2) occurs in the region in the northeastern part of the Moon's farside. The meridian with longitude 180° is central on the map. The western (during observations from the Earth) hemisphere takes up the map's right-hand side of the map. If one excludes the regions of maria and young ring structures similar to Mare Orientale, it turns out that the density distribution of craters is systematically higher in the western hemisphere. Hence, in the premare epoch the asymmetry of the western and eastern hemispheres with respect to the density of large craters was the decisive external sign of the structure of the Moon's surface. Most of the multiring basins and the most ancient of them—Oceanus Procellarum (*Whitaker*, 1981)—are concentrated in the western (during observations from the Earth) hemisphere.

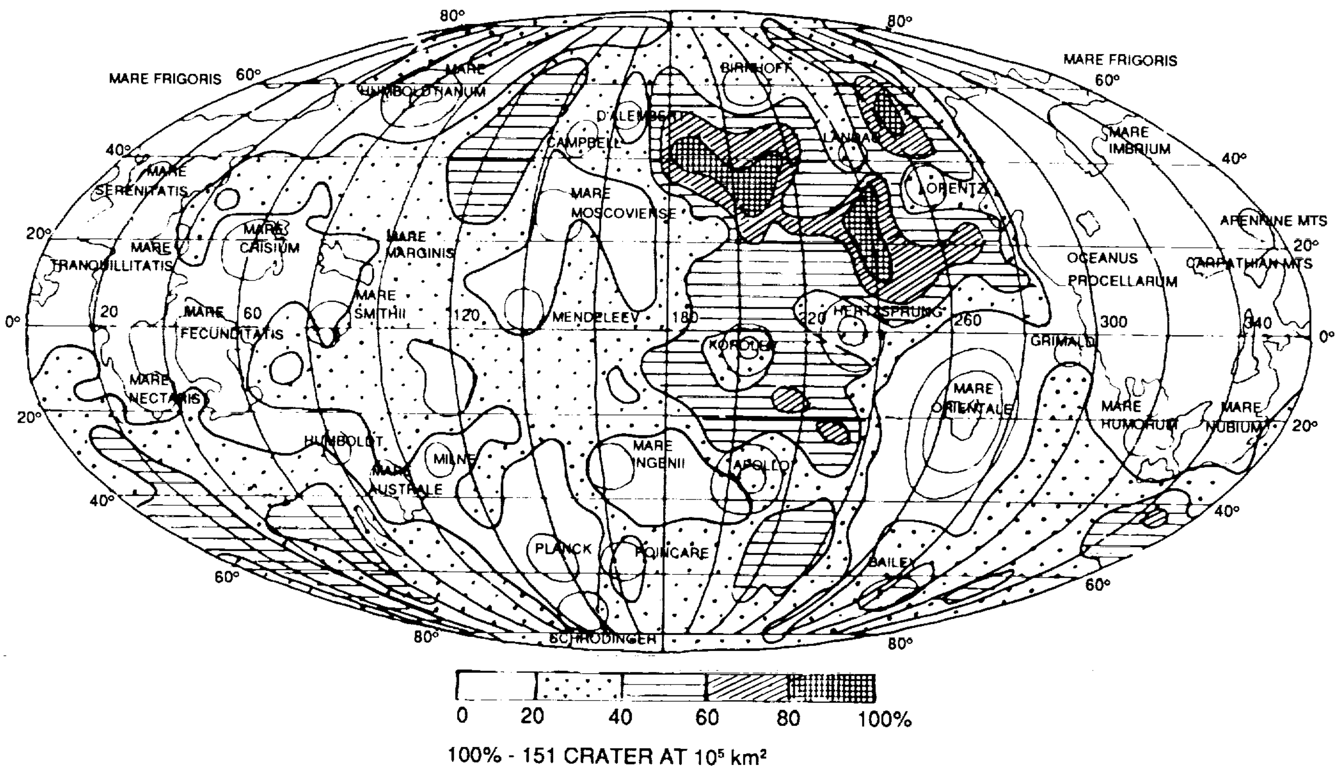

Fig. 1. The density distribution of lunar craters on the entire surface of the Moon.

On the basis of data on the Moon's reflectivity obtained by space vehicles and ground-based telescope, the albedo of 80% of the lunar surface, including the farside, was measured. When the character of the albedo distribution and the data from the orbital X-ray photography from Apollo 15 and 16 were compared, it was established that the distributions of the albedo correspond to the different chemical composition of the basic types of lunar rocks. The distribution of lunar rocks, based on available data, is given in Fig. 2 (*Schevchenko,* 1980). Excluding the relatively small regions of mare basalts, we again detect the asymmetry of the western (left) and eastern (right) hemispehres. It is worth noting that maria are within the regions of prolific lunar norites, i.e., the supposed product of premare volcanism. Hence, in the premare period the asymmetry of the western and eastern hemispheres manifested itself in the abundance of brighter (anorthositic) and darker (noritic) rocks on the surface of the Moon's single continental shield.

Gamma-ray spectrometry from lunar orbit by Apollo 15 and 16, revealed iron concentrations in the surface layer along mission routes (*Metzger et al.,* 1974). Analysis of these data also leads to the conclusion on the existence of western-eastern asymmetry. Highland rocks of the eastern hemisphere predominantly contain 6.5 to 9.5 wt% of iron, while the surface layer of the highlands of the western hemisphere contains less than 6.5% of iron. Since the rocks of these regions formed more than 4 b.y. ago this asymmetry characterizes the structure of the Moon in the premare period of lunar history.

Thus, the western-eastern asymmetry of the Moon relates to the earliest and the least understood period of the Moon's history and the evolution of terrestrial-type solar system bodies. The study of this period is of prime importance for understanding the origin and evolution of the solar system in particular and for lunar sciences in general. That is why the choice of the site for the lunar base requires that the western-eastern asymmetry of the hemispheres be taken into account. The enhanced concentration (density) of craters and multiring basins in the western hemisphere presupposes a vast and sample-rich field of ejecta from different depths. The task of selecting such samples in their natural bedding cannot be solved by automatic devices due to its complexity; a specialist's direct participation is needed to conduct such studies. Hence, taking into consideration the features mentioned above, it can be concluded that on a global scale the western hemisphere should be preferred for implementing scientific programs at the first stage of the functioning of a lunar base.

INDIVIDUAL OBJECTIVES OF THE LONG-TERM STUDY

Although the Moon's farside is very interesting, it seems that at the first stage of establishing a lunar base we should limit ourselves to the nearside due to simpler conditions of transport

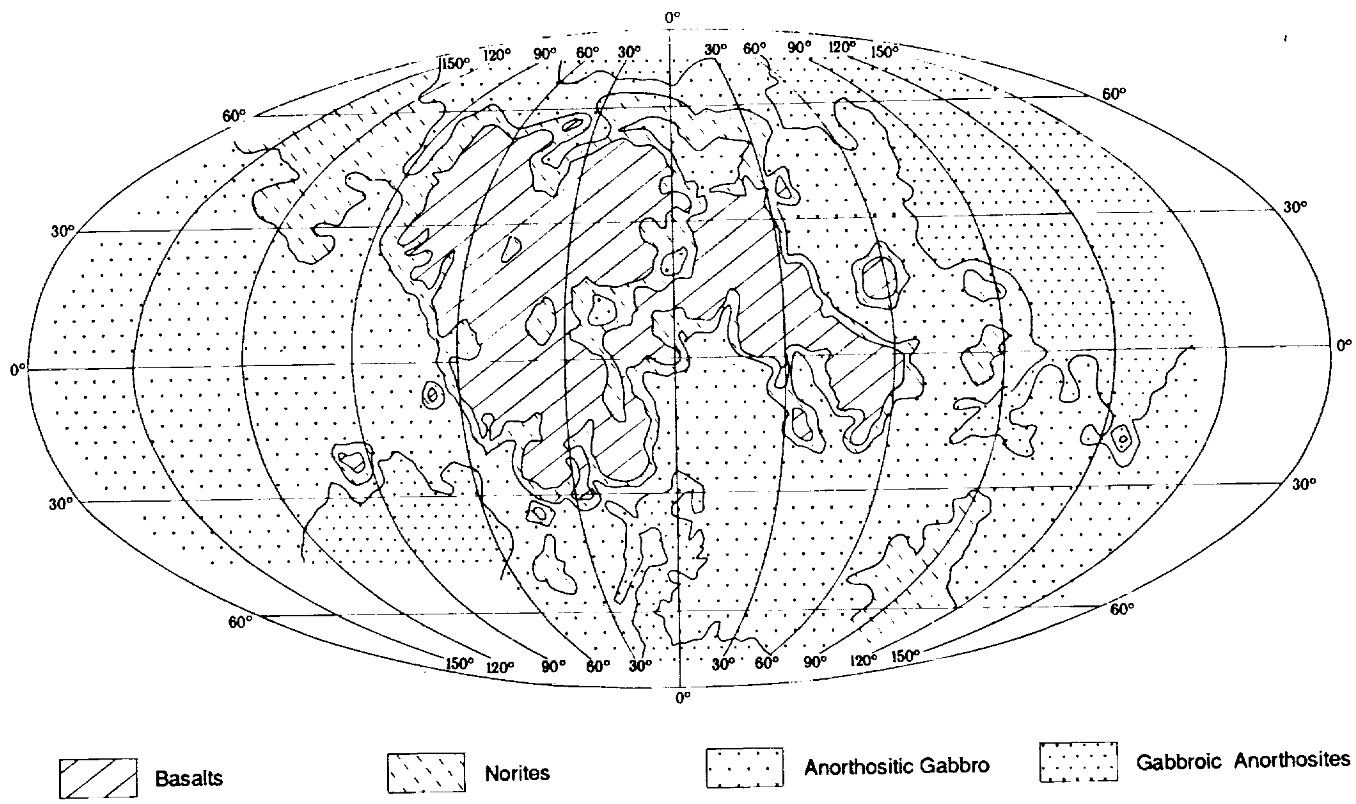

Fig. 2. A map of the distribution of lunar rocks according to photometric data.

and radio-television communications. From this point of view, let us consider primarily Oceanus Procellarum and highland regions closest to it (*Shevchenko,* 1986).

Large impact structures in this part of the Moon are of great interest. The study of samples from the lower part of the crust and ancient intrusions will help identify the age of the Moon's old rocks and determine the change in the rock composition with the bedding depth. Such samples can certainly be found in material ejected by an explosion from a depth of up to 20 km during the formation of the Moon's giant multiring basins. Apparently these basins were formed in the early stage of lunar history, 4.25-3.85 b.y. ago. If samples from the deepest layers of the crust affected by an impact explosion are identified amidst ejecta material, it will be possible to judge the time and composition of the formation of the Moon's primordial crust, the processes that took place in the era of giant impacts, and their effect on the subsequent volcanic activity during the formation of lunar maria. It is obvious that such a unique search can be carried out by a researcher in the course of long on-the-spot studies of objects and samples on the basis of preset parameters.

Of special interest in the region under consideration is the large crater Grimaldi situated within the highland near the western boundary of Oceanus Procellarum. The floor of the crater is flooded by lava rocks with a very low albedo. The outer rim is 237 km in diameter; the flooded part of the crater is about 170 km across. According to morphological classification by the degree of preservation of the crater's rim (*Rodionova et al.,* 1987), Grimaldi belongs to the pre-Imbrium period of formation. Later, approximately 3.85 b.y. ago, the crater was lavishly overlapped by ejecta from the multiring depression of Mare Orientale and apparently was destroyed to a great extent. Still later, in the Imbrian period, the dark covering of the Grimaldi floor appeared. It is also worth noting that the flooding of the crater's floor by dark lavas was multiphase, which can be traced in the photometric picture of the region obtained by processing a photograph brought back to Earth by Zond 6 space probe (*Shevchenko,* 1980). The regions of equal brightness in this picture have a sharply pronounced asymmetric structure. The center of the darkest area of the lava inside the crater is observed to the south.

The change in the reflective capacity of covering material is unambiguously correlated with the geometry of the rim's topography and the adjacent crater floor. To the north, where the rim is partly destroyed and the topography is gently sloping, the gradual brightening of the surface takes place along several dozens of kilometers making up the width of the transitional zone. In the crater's southern part the rim is fully preserved with the considerable and rather sharp change of height between the rim's crest and the floor. The sharp boundary between the darkest region of the surface of the intercrater filling and the bright highland surroundings of Grimaldi correspond to this. In all likelihood, the general form of identified regions of different brightness is the result of the dynamics of lava flows occurring at different times during the filling of the crater by mare-type material. Despite the proximity of this formation to the western

boundary of Oceanus Procellarum, photometric analysis of the structure of the crater and its surroundings belies the surface flooding of Grimaldi by lava from neighboring mare regions. The source of dark material lies inside the crater, presumably in its southern part. This conclusion is supported by the anomaly of gravity inside the crater (*Phillips and Dvorak,* 1981). The photometric structure of Grimaldi Crater is shown in Fig. 3. Albedo percentage is indicated for equal reflectivity lines (*Shevchenko,* 1986). Thus, Grimaldi Crater contains traces of different epochs of lunar history from the most ancient period to the final stage of global volcanism that resulted in the formation of lunar maria.

Some 500 km away from Grimaldi Crater is another structure that deserves close attention and detailed study. This is an albedo anomaly coinciding with the Moon's largest magnetic anomaly, Reiner Gamma (*Shevchenko,* 1980). Studying the photometric properties of this formation shows that the brightening of the surface belongs to a very thin layer, in all probability modified by mechanical processes. The polarization properties and the presence of reflection indicate an enhanced density that is not inherent in the typical lunar surface. The sites of the artificial compaction in the uppermost layer of the lunar regolith possess similar properties.

The map of the albedo of the Reiner Gamma magnetic anomaly is given in Fig. 4. The albedo of the main part of the brightness structure is 1.86 times higher than that of the surrounding surface. A similar result was obtained by photometry of the trace of Lunokhod 2's light-track-measuring wheel (*Shevchenko,* 1982). The rover's wheel compacted, without mixing, the uppermost surface layer by its pressure on the Moon's soil. This light compaction increased the brightness of the trace of the ninth wheel by 1.82 times when compared with the surrounding background. At the same time, reflection appeared on this trace, the increase in brightness at large phase angles being typical of the Reiner Gamma magnetic anomaly. A similar phenomenon can be seen, for instance, in the pictures of the light tracks of the rover used for manual cargo carriage in the Apollo 14 expedition (*Apollo 15 Preliminary Science Report,* 1972). A similar effect has been recorded when soil is subject to the impact of gas jets of sufficient density. The pictures of the Apollo 15 landing site taken before and after the spaceship's landing on the Moon show that in the latter case a bright halo around the landing site appears on the surface (*Apollo 15 Preliminary Science Report,* 1972).

The described properties of the formation of the Reiner Gamma magnetic anomaly enable us to speculate that this anomaly originated from the gas ejections of a dense coma surrounding a comet's active nucleus, which formed the albedo structure by compacting the Moon's uppermost soil layer. In this case it is possible to assess the anomaly's absolute age. The photometric properties of surface material are determined by a very thin porous soil layer. This layer is 1-2 cm thick with a density of 0.2-0.3 H/cm^2. Calculations have shown that due to the pressure of the gas jet of descent engines the value reached is about 0.689 H/cm^2 (*Apollo 15 Preliminary Science Report,* 1972), which is quite enough for destroying the high porosity of the surface layer through the compaction of soil. The natural level of porosity and, hence, the typical photometric properties can be restored as a result of the long effect of micrometeoritic erosion, the rate of which is 0.5 mm/10^6 yr. Hence, the recovery of the 2-cm thick porous layer will take not more than 40 m.y. This time is the absolute age of the Reiner Gamma formation, a very young large structure on the Moon linked perhaps with the era of global catastrophes in the internal part of the solar system.

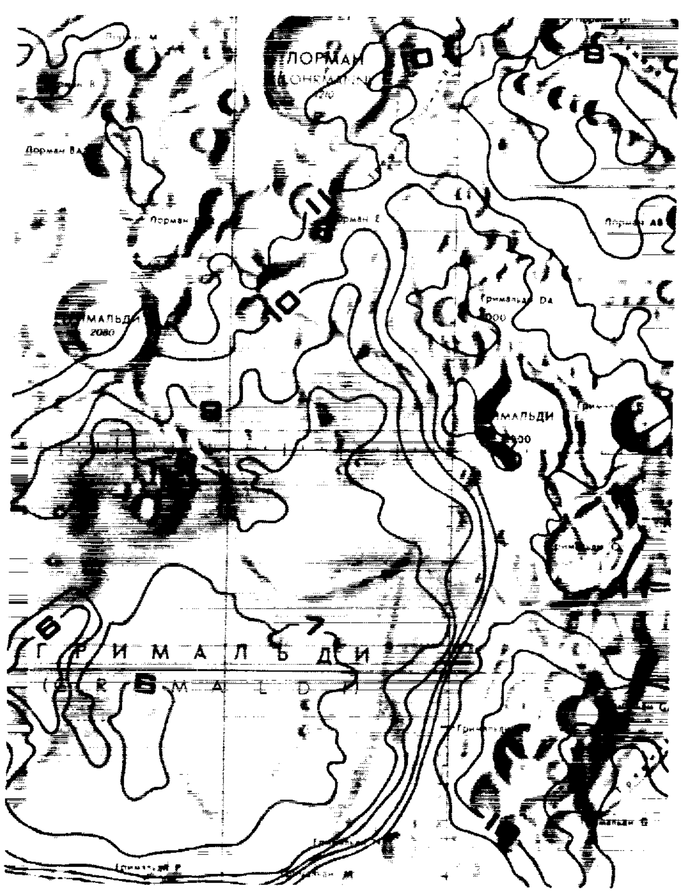

Fig. 3. A map of the albedo of Grimaldi Crater.

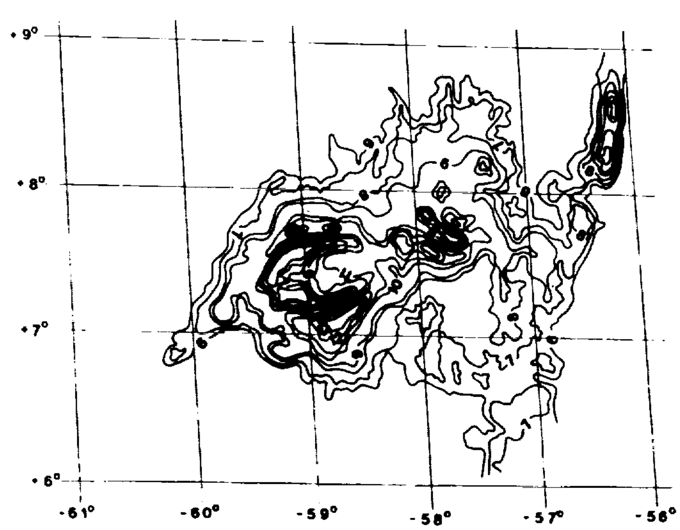

Fig. 4. A map of the albedo of the Reiner Gamma magnetic anomaly.

THE PRESENCE OF ILMENITES—OXYGEN-BEARING MINERALS

The presence of ilmenites—oxygen-bearing minerals—is essential for the choice of promising areas on the Moon for detailed studies and landing (*Shevchenko,* 1986). Ilmenite is an ore mineral and its greatest abundance (up to 20%) is recorded in highly titaniferous mare basalts. Ilmenitic basalts abound in some lunar areas whose genesis is insufficiently studied. Predicting ilmenitic basalt regions is feasible with the use of planetary astrophysics and remote sensing. According to most predictions ilmenitic basalts seem to be spread in the central and western parts of Oceanus Procellarum, in addition to the Apollo 11 and Apollo 17 landing sites, where they were detected in the samples taken back to Earth. There are relatively small areas in the northwestern part of Oceanus Procellarum that, according to the data of photography from the Earth and measurements from the Zond 6 spacecraft, have the lowest albedo on the lunar surface—less than 6%. In albedo and spectral zone characteristics, these regions have greater titanium content in the surface rocks (*Shevchenko,* 1986). Methods of remote identification of lunar rocks with enhanced titanium oxide content continue to be improved.

In accordance with the calculated values of the energy of metal-metal electron transitions in ilmenite, a spectral range of the characteristic 0.5- to 0.6-mcm absorption band has been isolated, where it is superimposed on individual absorption bands of transitions in the crystalline field in Fe^{2+} and Ti^{3+} ions. This conclusion is most clearly confirmed by the results of the laboratory measurements of cleansed samples of terrestrial ilmenite. Figure 5 presents reflection spectra of four ilmenite samples (Nos. 1 to 4) extracted from a depth of 100 to 1250 m in the northwestern Ukraine (U.S.S.R.). The chemical composition of the samples is similar, and the titanium oxide content varies within 49.28 to 49.40 wt%. Sample No. 2 contains slightly more iron oxide, which explains the specific features of its spectrum. All spectra contain a characteristic absorption band that indicates the presence of titanium in rocks in all other cases (*Busarev and Shevchenko,* 1988).

For the region northwest of the crater Lichtenberg in Oceanus Procellarum, spectra in the 0.336 to 0.758 mcm interval with resolution 0.0048 mcm were obtained. Spatial resolution corresponds to 25 km on the Moon's surface. Observations were conducted in the Crimea on the Zeiss-600 telescope in May 1987. Figure 6 presents spectra for section 1 with coordinates 72.9°W, 35.0°N, for section 2 with coordinates 75.1°W, 32.7°N, and the spectrum of the Apollo 17 landing site obtained during those observations. The comparison of the spectra shows that the characteristic absorption band in the 0.5-0.6-mcm region manifests clearly enough. Individual features of the spectra also coincide well enough to indicate the enhanced ilmenite content in the surface rocks of the region northwest of the crater Lichtenberg. The quantitative estimate based on the empirical dependence of the slope of the spectrum in the 0.400-0.565-mcm interval on the percentage of titanium oxide gives the value 5.3% for section No. 1 and 6.8% for section No. 2 (*Schevchenko and Busarev,* 1988).

The area northwest of Lichtenberg Crater is a region of presumed presence of ilmenitic basalts. (This region was preliminarily singled out in albedo and spectral zonal characteristics.) Since all these sections have similar optical properties, including the maximum degree of polarization (*Dzhapiashvili and Korol,* 1982), the estimates based on spectrophotometric data can be extended to them too as regards the enhanced ilmenite content in surface rocks.

THE MOON'S KEY REGION

It can be concluded that the western part of Oceanus Procellarum is a key region of the lunar surface whose detailed and comprehensive study is of great importance for the exploration of the Moon and for the understanding of the early history of the solar system.

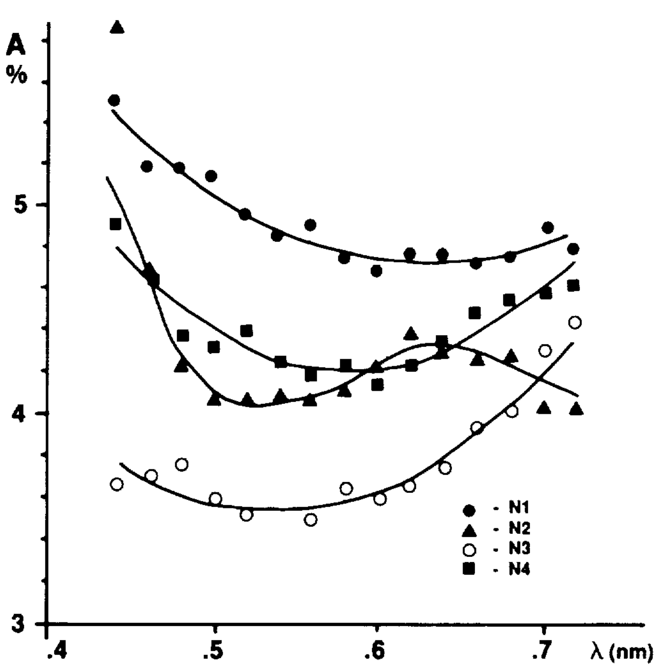

Fig. 5. Reflection spectra of the samples of cleansed ilmenite.

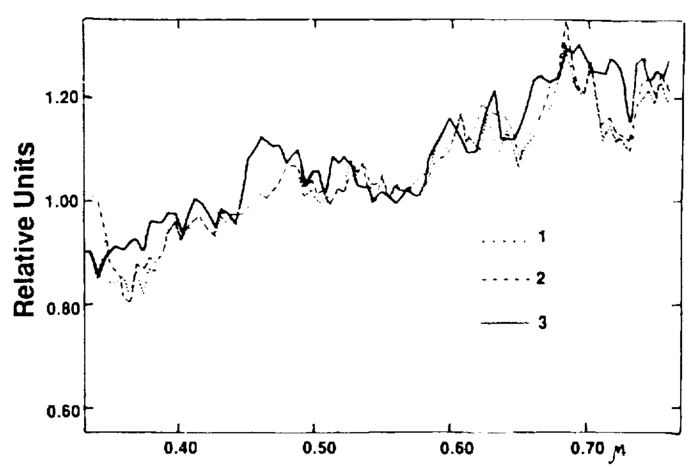

Fig. 6. Spectra of the sections of the lunar surface in the Lichtenberg Crater region (1, 2) and the Apollo 17 landing site (3).

Figure 7 presents the map-diagram of the western part of Oceanus Procellarum. The areas identified according to ground-based remote sensing data are assumed to have an enhanced titanium oxide content in surface rocks. This conclusion is drawn from the low albedo (less than 6%), increased reflectivity ("blue" rays revealed in the course of spectrozonal photography on wavelengths 0.37 and 0.61 mcm), a higher maximum degree of polarization, and lastly, spectral characteristics. Sections in the region northwest of Lichtenberg Crater for which spectra are obtained (see Fig. 6) are denoted by asterisks. Similarly, the surface rocks on the floor of the darkest southern part of Grimaldi Crater can be assumed to be ilmenitic basalts.

According to the stratigraphic diagram of Oceanus Procellarum (*Whitford-Stark and Head,* 1980) the regions in the western part of Oceanus Procellarum coincide with the sites of regional lava flows originated in small craters and with a system of rilles indicating the direction of the movement of these flows. It seems that the lava flows and their sources identified by the above authors were the entryways of ilmenitic basalts to the surface. On the other hand, these features are indicative of the traces of lunar volcanism that deserve special close study.

Maria hills situated on the territory beween the Reiner Gamma structure and Marius Crater are also such objects. As volcanic structures, these objects relate apparently to the epoch of late postmare volcanism on the Moon, the traces of which, due to the limited spread of this process, are very rare.

Mention should also be made of the fact that in the region under review there are objects near which nonstationary processes have been recorded many times. They are linked with the manifestations of modern lunar volcanism (*Shevchenko,* 1986). These sites, denoted on the diagram by the sign V, are hypothetical sources of volatiles of endogenic origin. Nonstationary phenomena on the Moon have not been studied enough because of specific features of their manifestations. Hence, a detailed study of the places where nonstationary phenomena have repeatedly been observed on the surface is important for understanding the evolution of the Moon and other minor bodies of the solar system and the nature of modern volcanism on planets and satellites. It is probable that the detection of the sources of the intensive outgassing from the Moon's interior will be of great importance for the functioning of a lunar base.

Thus, in the given region, according to preliminary data, we have natural resources in the form of ilmenitic basalts suitable for obtaining oxygen and in the form of volatiles—products of the eruption from the Moon's interior. The same sites of nonstationary phenomena and the traces of the late volcanism of the postmare period are the major aim of studies. Grimaldi Crater, lying near this region, contains numerous traces of the multistage process of the formation and evolution of lunar highlands and maria, including intercrater volcanism. Finally, this region has the youngest large formation on the Moon—the anomalous Reiner Gamma structure, the study of which opens up an opportunity to detect the traces of cometary impacts against the lunar surface that perhaps belong to one of the latest epochs of catastrophic events in the solar system.

The highland adjacent (from the west) to Oceanus Procellarum undoubtedly contains ejecta relating to different periods of impact phenomena, from the earliest, when the region of the highest

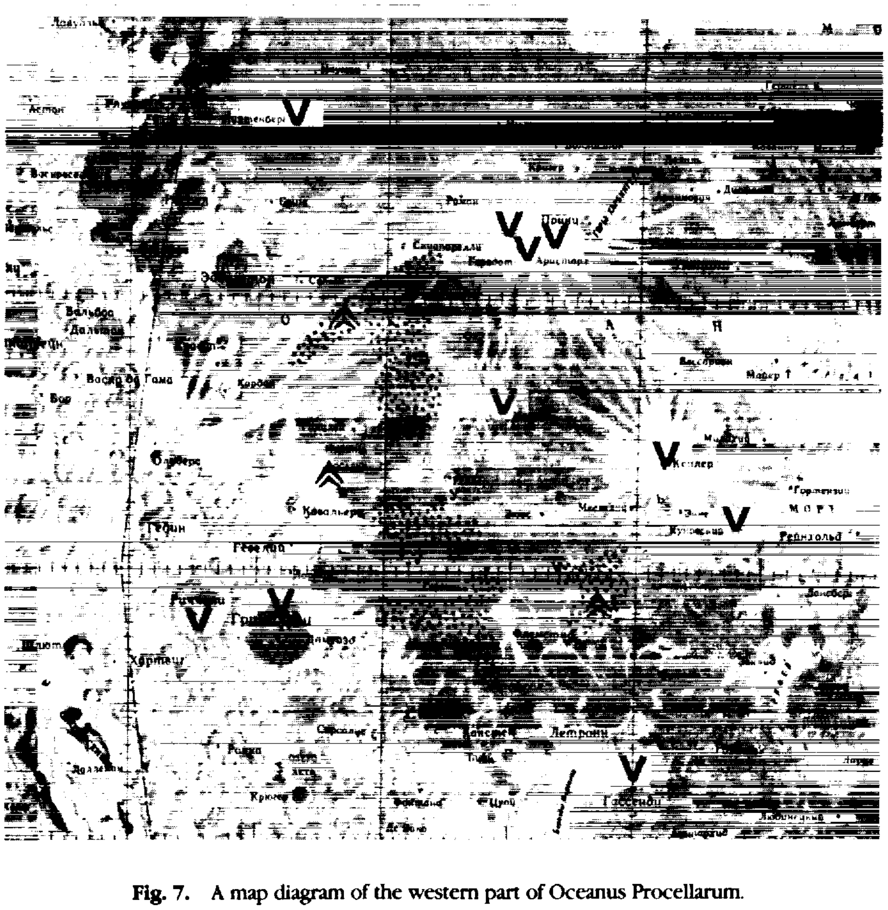

Fig. 7. A map diagram of the western part of Oceanus Procellarum.

density distribution of craters situated northwest of the region under consideration was shaped, to the latest, when the multiring basin of Mare Orientale came into existence. No doubt as a result of later impact events, these ejecta were partly transferred and scattered on the mare surface of Oceanus Procellarum. A more careful search for such fragments is needed in the overall regolith mass covering the western part of Oceanus Procellarum.

The plain-type terrain in this region provides favorable conditions for takeoff-landing transport operations of unmanned, remotely controlled, and manned spacefcraft. The typical texture of the surface is known from the pictures of the surrounding terrain taken at the landing sites of unmanned space probes—Luna 9 (Planitia Descensus), Luna 13, and Surveyor 1 (denoted on the diagram by big asterisks).

It can be concluded that expoloration by remote (ground-based) and direct (space vehicles) method should be focused on this key region on the Moon as the most suitable (as preliminary data show) for the establishment and functioning of a lunar base.

REFERENCES

Apollo 15 Preliminary Science Report (1972) *NASA SP-289*, pp. 2550-2553.

Busarev V. V. and Shevchenko V. V. (1988) TiO_2 in crater Le Monier (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, pp. 44-45. Lunar and Planetary Institute, Houston.

Dzhapiashvili V. P. and Korol A. N. (1982) *Polarimetric Atlas of the Moon*. Tbilisi. 44 pp.

Metzger A. E., Trombka J. I., Reedy R. C., and Arnold J. R. (1974) Element concentrations from lunar orbital gamma-ray measurements. *Proc. Lunar Sci. Conf. 5th*, pp. 1067-1078.

Philipps R. J. and Dvorak J. (1981) The origin of lunar mascons. In *Multi-Ring Basins, Proc. Lunar Planet. Sci. 12A* (P. H. Schultz and R. B. Merrill, eds.), pp. 91-104. Pergamon, New York.

Rodionova Zh. F., Skobeteva T. P., and Karlov A. A. (1985) A morphological catalogue of lunar craters (abstract). In *Lunar and Planetary Science XVI*, pp. 706-707. Lunar and Planetary Institute, Houston.

Rodionova Zh. F. et al. (1987) In *A Morphological Catalog of Lunar Craters* (V. V. Shevchenko, ed.), p. 174. Moscow University, Moscow.

Rodionova Zh. F. and Shevchenko V. V. (1988) The creation of maps of the density distribution of lunar craters. In *Lunar and Planetary Science XIX*, pp. 992-993. Lunar and Planetary Institute, Houston.

Shevchenko V. V. (1968) Problems of a manned lunar base. *Vestnik Akademii Nauk SSSR, 6*, 95-104.

Shevchenko V. V. (1980) *Modern Selenography*. Nauka, Moscow. 288 pp.

Shevchenko V. V. (1982) The technique and results of photometric studies in the mare-highland transition zone by Lunokhod 2. *Trudy (Proceedings) of the Sternberg Astronomical Institute, 51*, 195-207.

Shevchenko V. V. (1984) Optical properties of Reiner gamma magnetic anomaly on the Moon (abstract). In *Lunar and Planetary Science XV*, pp. 772-773. Lunar and Planetary Institute, Houston.

Shevchenko V. V. (1986) A manned lunar base project. *Vestnik Akademii Nauk SSSR, 10*, 85-98.

Shevchenko V. V. and Busarev V. V. (1988) Ilmenites in Oceanus Procellarum (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 219. Lunar and Planetary Institute, Houston.

Whitaker E. (1981) The lunar Procellarum basin. In *Multi-Ring Basins, Proc. Lunar Planet. Sci. 12A* (P. H. Schultz and R. B. Merrill, eds.), pp. 105-111. Pergamon, New York.

Whitford-Stark J. L. and Head J. W. (1980) Lunar surface morphology and stratigraphy: A remote sensing synthesis. *J. Brit. Astron. Assoc., 90*, 312-345.

GEOLOGICAL AND GEOPHYSICAL FIELD INVESTIGATIONS FROM A LUNAR BASE AT MARE SMYTHII

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Mare Smythii, located on the equator and east limb of the Moon, has a great variety of scientific and economic uses as the site for a permanent lunar base. Here a complex could be established that would combine the advantages of a nearside base (for ease of communications with Earth and normal operations) with those of a farside base (for shielding a radio astronomical observatory from the electromagnetic noise of Earth). The Mare Smythii region displays virtually the entire known range of geological processes and materials found on the Moon; from this site, a series of field traverses and investigations could be conducted that would provide data on and answers to fundamental questions in lunar geoscience. This endowment of geological materials also makes the Smythii region attractive for the mining of resources for use both on the Moon and in Earth-Moon space. We suggest that the main base complex be located at 0, 90°E, within the mare basalts of the Smythii basin; two additional outposts would be required, one at 0, 81°E to maintain constant communications with Earth, and the other, at 0, 101°E on the lunar farside, to serve as a radio astronomical observatory. The bulk of lunar surface activities could be conducted by robotic teleoperations under the direct control of the human inhabitants of the base.

INTRODUCTION

Several advanced planning studies are currently underway to identify strategies for the establishment of a permanent base on the Moon (*Mendell,* 1985). Depending upon the ultimate emphasis placed on lunar base operations, several considerations enter into the planning process, one of which includes the selection of the base site. Any lunar base site will offer something for various users. *Duke et al.* (1985) identified three separate scenarios for development of a lunar base, each having a different emphasis for ultimate base use: lunar science, resource utilization, and lunar settlement. These different thrusts are not mutually exclusive, but each could have slightly different criteria for base site selection. In fact, it is highly probable that a lunar base program will have elements of each emphasis; indeed, one of the attractions of a lunar base program is that it has so much to offer to many different users.

Although it may be premature at this stage to design detailed, site-dependent operational strategies, it is not too soon to begin considering what types of lunar base sites offer the most benefits to the most potential users. In this spirit, we here present a study of the Mare Smythii region, on the east limb of the Moon, and suggest that this location presents many advantages to all the currently identified potential base users.

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ADVANTAGES OF A BASE SITE ON THE LUNAR LIMB

A consequence of the Moon's synchronous periods of rotation and revolution is that the Earth is always visible at the same location in the sky on the nearside and always invisible from the farside. This presents both opportunities and problems. For normal lunar base operations, it may be desirable to maintain constant communication with the Earth, a condition satisfied by any nearside site. However, one of the prime advantages of the Moon as an astronomical observing platform is that the lunar farside is the only known place in the solar system that is permanently shielded from the extensive radio noise produced by our home planet. These two requirements are mutually incompatible, short of designing and operating two separate lunar base sites.

Because the Moon orbits the Earth in an elliptical path and the plane of the lunar orbit is not quite perpendicular to its rotation axis, the Moon wobbles slightly, or librates, in both latitude and longitude. Thus, the lunar limb (the great circle defined by the poles and the 90° meridians) is the only place on the Moon where the Earth is sometimes visible and sometimes occulted. It is in this region that a base could be established that may potentially satisfy both paradoxical requirements: that of radio access to the Earth and shielding from the Earth's radio noise. We emphasize at the outset that no single site accomplishes these goals at all times, but rather, several outposts or "sub-bases" in close proximity are required to make use of the lunar libration effect.

Several studies have advocated base sites at the lunar poles (e.g., *Burke,* 1985), either because of the availability of continuous solar power or because the continuous darkness of crater floors may have trapped volatiles (including water) over geologic time (e.g., *Arnold,* 1979). However, from an astronomical viewpoint, a major drawback to a polar site is that only half of the sky is ever visible. Moreover, the unique lighting conditions of the poles, where the sun is constantly near the horizon and the surface is either jet black or blazing white, would make both surface operations and geological exploration difficult.

For these reasons, we believe that a limb site located on the equator has many advantages over a polar site. First, the entire sky is visible from the lunar equator over the course of a month. Second, equatorial sites on the Moon are easily and constantly accessible in minimum energy trajectories from the LEO space station, the probable staging location for base establishment. The Mare Smythii site that we endorse as a lunar base site is not only on the limb, at the equator, but it is in a region containing evidence of a great diversity of geological processes as well as a variety of materials that occur in reasonably close proximity. This region can satisfy all potential lunar base users—geoscientists, astronomers, miners, and colonists.

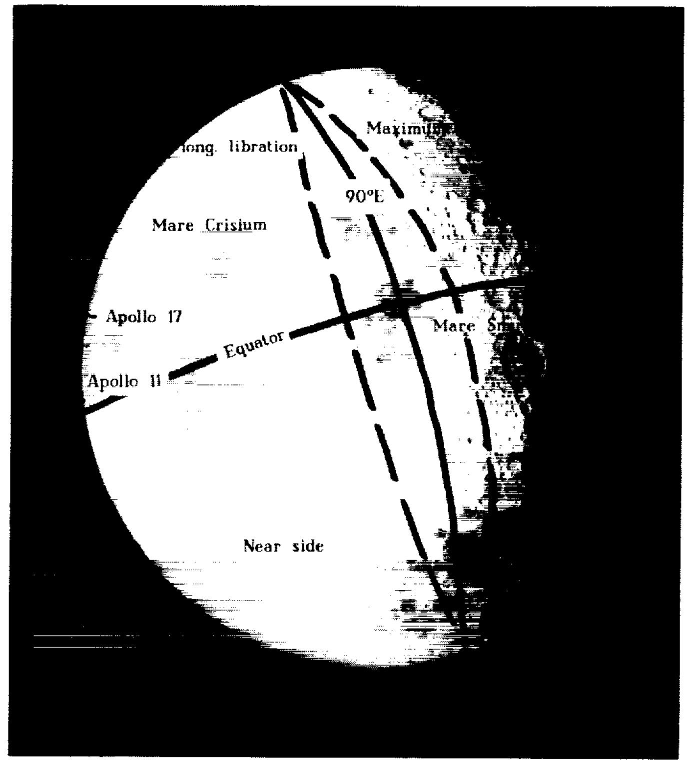


Fig. 1. Global view of the eastern limb of the Moon showing Mare Smythii and its relation to the longitudes of lunar libration. Some regional features and two Apollo landing sites are also shown. AS12-55-8226.

ADVANTAGES OF THE MARE SMYTHII SITE

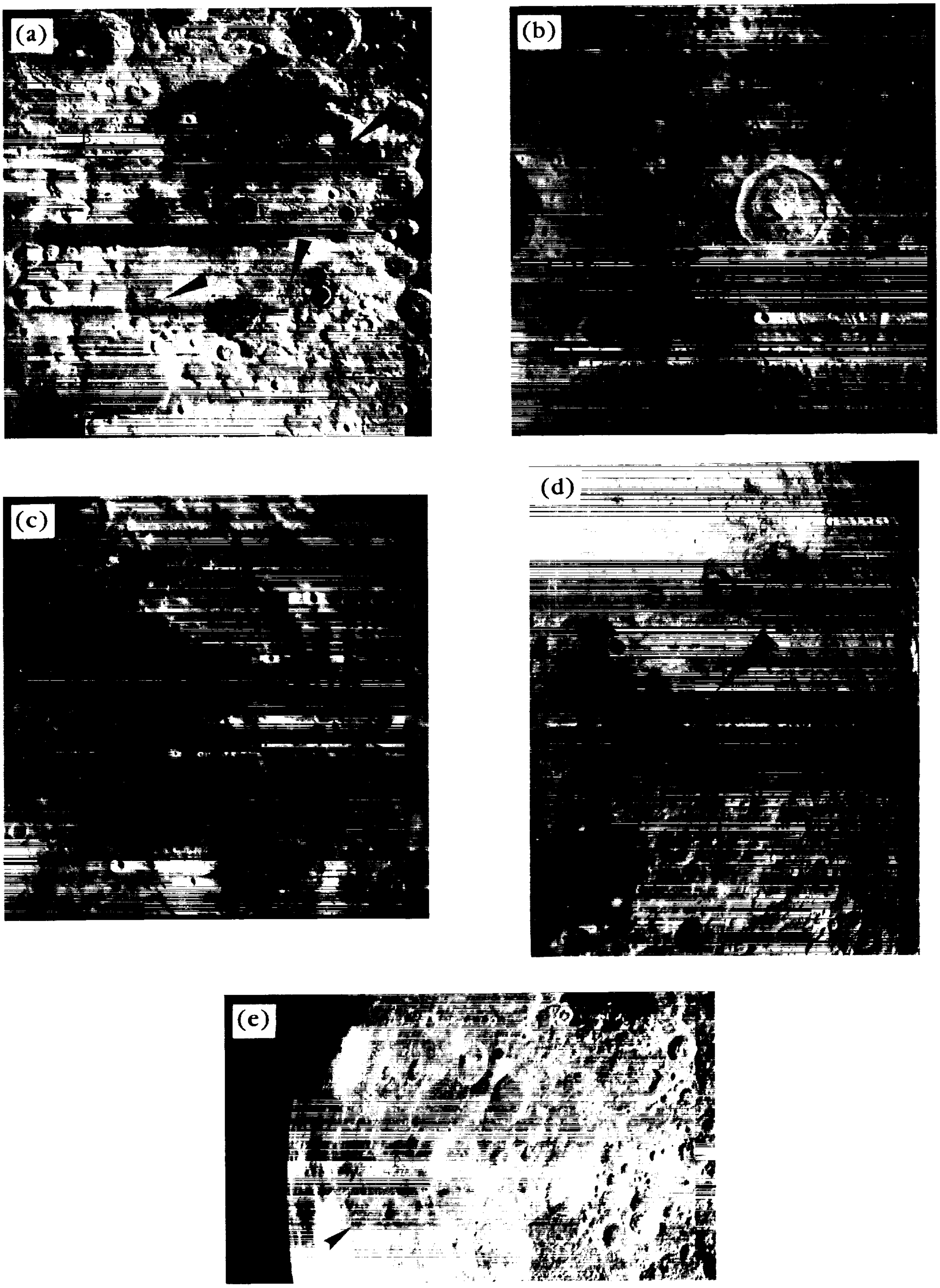
Mare Smythii is a dark lowland on the east limb of the Moon (Fig. 1). The region is well covered by orbital remote-sensing data; analysis of these data suggests that the region is probably one of the most diverse on the Moon (Figs. 2 and 3; Table 1). (For a concise summary of our current understanding of lunar geoscience, see *Lunar Geoscience Working Group,* 1986.) In the following paragraphs, we briefly discuss the advantages of the Mare Smythii region from the perspectives of several potential lunar base users.

Geological Considerations

The Mare Smythii region displays the two principal geological units found on the Moon: maria (the dark, smooth plains) and terrae (the rugged, heavily cratered highlands). Mare Smythii consists of dark lava flows that partly fill a much older, multiringed impact basin. The Smythii basin is one of the oldest lunar basins that retain recognizable ring structure; it is composed of three rings 370, 540, and 740 km in diameter. Basins were formed by the impact of asteroid-sized bodies on the Moon before about 4 b.y. ago; the study of the mechanics of their formation and their geological effects on crustal materials is one of the primary tasks of lunar geoscience.

The dark, smooth maria are known from Apollo results to consist of basaltic lava flows; the ages of mare basalt samples returned by Apollo range from 3.9 to about 3.1 b.y. The ages of mare lava flows not visited by Apollo may be estimated by examining the density of superposed impact craters. Results of this exercise for Mare Smythii are shown in Fig. 4; the astonishing result is that the lava flows of the Smythii basin are among the youngest on the Moon. The position of the crater frequency curve of Mare Smythii relative to that of dated Apollo site lava flows indicates that the Smythii basalts are probably 1 to 2 b.y. old. (A more precise estimate is impossible because the cratering history of the Moon over the last 3 b.y. is only approximately known.)

Fig. 2. Geological features of and near the Smythii basin. **(a)** Regional view of the Smythii basin. Dark smooth area (M) is Mare Smythii, consisting of high-Ti mare basalts. Smythii basin rim (B) is 370 km in diameter and is composed of anorthositic rocks. Dark mantle (arrows) is pyroclastic ash produced by fire fountain eruptions of basaltic magmas. Many floor-fractured craters (F) are visible on the basin floor. Large, mare-filled crater at upper left is Neper (N), 137 km in diameter. AS15-95-12991. **(b)** Western part of the Smythii basin floor. The prominent floor-fractured crater (F) is Schubert C (31-km diameter). Mare basalt flows (M) fill the highlands terrain (H) of the basin. Dark mantle deposits are associated with irregular volcanic vents (arrows) in this area. LO I-5 M. **(c)** Eastern part of the Smythii basin floor, showing lava flows (M) of Mare Smythii, highlands basin rim (H), and floor-fractured crater Purkyne U (F; 51-km diameter). Young rayed crater (arrow) overlies lava fill of Purkyne U. LO I-19M. **(d)** Regional view of terrain northeast of Mare Smythii (S). Basalts of Mare Marginis (M) are relatively young (about 2-3 b.y. old) and rich in KREEP. Swirls within Marginis (big arrow) are associated with large surface magnetic anomalies. Dark-halo impact craters (small arrows) are associated with buried ancient mare basalts, common in this area. Large rayed crater at top (G) is Giordano Bruno (22-km diameter), possibly the youngest large crater on the Moon. Portion of AS16-3021. **(e)** Regional view of terrain southwest of Mare Smythii (S). The mottled light plains (P) of the Balmer basin display dark-halo craters (arrow) and are KREEP-rich. Large crater near bottom center is Humboldt (207-km diameter). Portion of AS17-152-23293.


(a)
(b)
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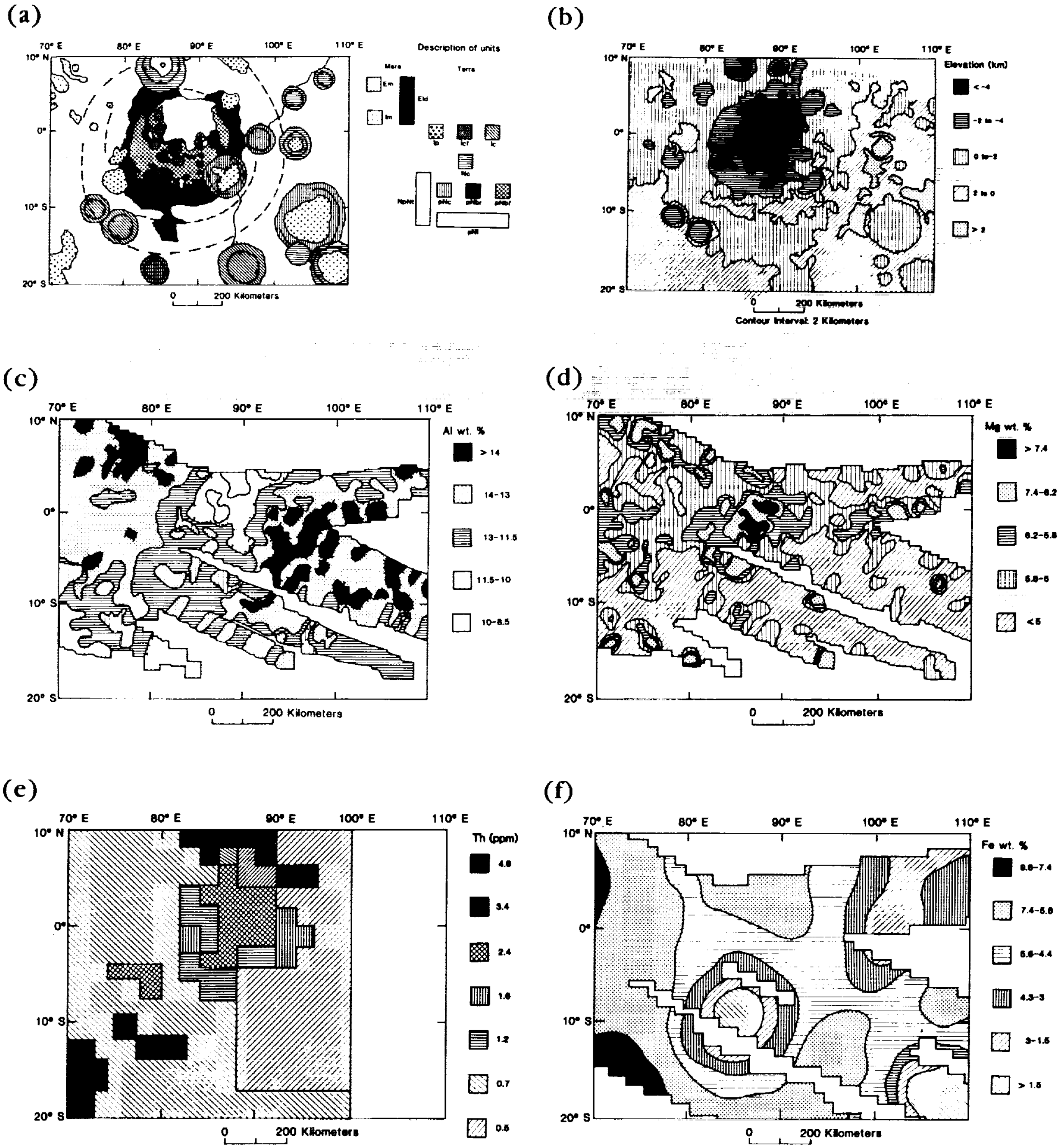

Fig. 3. Maps of geologic, topographic, and chemical remote-sensing data for the Smythii region. All geochemical data were obtained from the orbiting Apollo 15 and 16 spacecraft and, except for the thorium data, are from *La Jolla Consortium* (1977); chemical composition of major geologic units is summarized in Table 1. All maps are Mercator projection. **(a)** Geology, modified from *Wilhelms and El-Baz* (1977). Relative ages indicated by capital letters: E—Eratosthenian, I—Imbrian, N—Nectarian, pN—pre-Nectarian. Units: Em—basalts of Mare Smythii; Im—other mare basalts; Eld—pyroclastic dark mantle deposits; Ip—smooth plains, some displaying dark-halo craters; Icf—floor-fractured craters; pNbr—Smythii basin rim material; pNbf—Smythii basin floor material; Ic, Nc, pNc—impact crater materials; NpNt and pNt—undivided terra (highlands) material. Dashed lines indicate basin rings. Smaller circular features are impact craters; lines with ticks indicate crater rims. **(b)** Regional topography from Apollo metric photographs by U.S. Geological Survey (unpublished, 1982). Elevations based on global datum of a spherical Moon 1738 km in radius. **(c)** Aluminum concentration (in weight percent). **(d)** Magnesium concentration (in weight percent). **(e)** Thorium concentration (in parts per million; from *Haines et al.,* 1978). **(f)** Iron concentration (in weight percent).

(g)

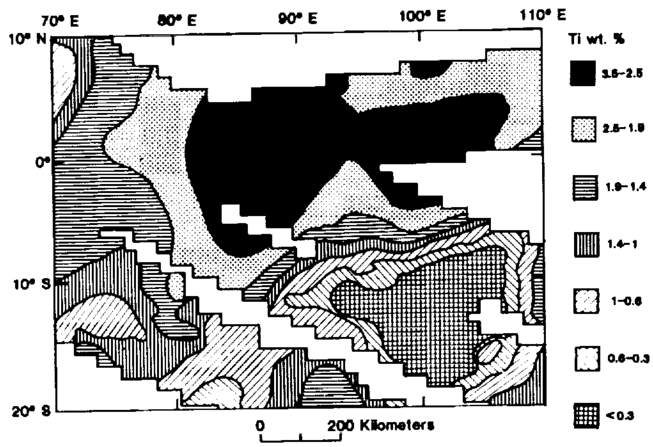

Fig. 3. (continued) **(g)** Titanium concentration (in weight percent).

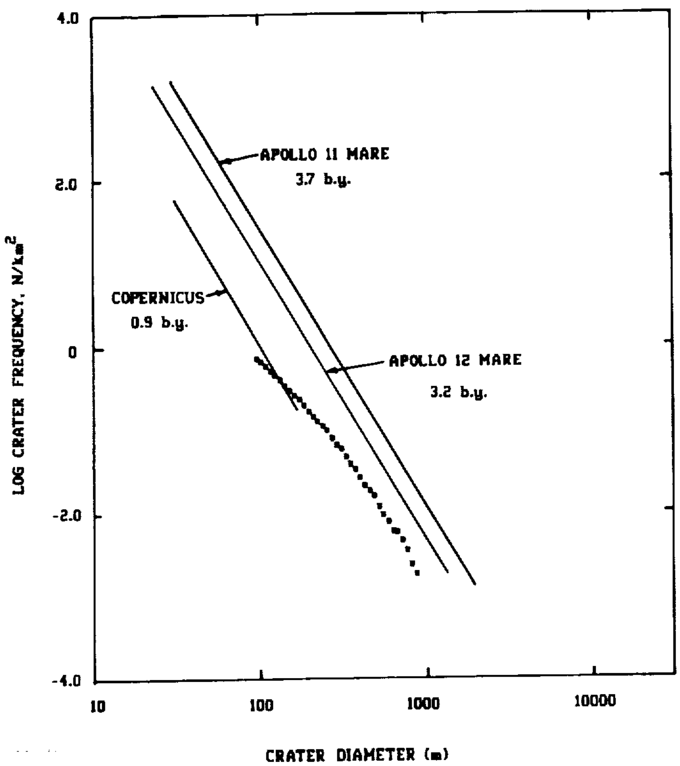

Fig. 4. Crater-frequency distribution for the mare basalt flows of Mare Smythii (squares), shown in comparison to those of the Apollo 11 and 12 landing sites and the crater Copernicus (from *BVSP,* 1981). Position of the Smythii curve indicates that these lava flows are significantly younger than those of the Apollo 12 site (the youngest sampled lunar lavas); the age of the Smythii flows is probably about 1 to 2 b.y.

Although very old by terrestrial standards, these are among the youngest lunar volcanic products; their study will greatly aid the reconstruction of the volcanic and thermal history of the Moon.

In addition to the lava flows within the basin, several localities exhibit dark mantling deposits (Fig. 2a-c). We know from the Apollo results that lunar dark mantles are composed of volcanic, pyroclastic deposits, such as the Apollo 17 orange glasses and black beads. Lunar pyroclastic glasses form in Hawaiian-type "fire fountaining" eruptions; moreover, the composition of these glasses indicates that they undergo little chemical modification during their ascent from their source regions in the lunar mantle. Thus, study of pyroclastics is important to understand lunar volcanism and the composition of the lunar mantle.

In addition to lava flows and pyroclastics, several craters inside the Smythii basin appear to be modified by internal processes (Figs. 2a-c, 3a). These features, floor-fractured craters, are not uncommon on the Moon and many are associated with the margins of the maria and other sites of volcanic activity. One hypothesis for their origin is that the subfloor zones of impact craters become sites of magmatic intrusions; the continuing injection of magma has uplifted the crater floor in a doming action that fractured them (*Schultz,* 1976). The Smythii basin, containing at least eight of these features in different states of development, is an ideal area in which to study the process of internal modification of impact craters.

The lunar terrae or highlands make up the vast bulk of the lunar crust. The crust appears to be composed largely of rocks rich in plagioclase (a silicate mineral rich in aluminum and calcium); such rocks are called anorthosites. One of the early ideas concerning the origin of the lunar crust was that the amount of plagioclase in the crust far exceeds what could be reasonably expected to be produced by partial melting and that a former "ocean" of magma existed on the Moon. The terrae surrounding the Smythii basin contain aluminum-rich terrains that are probably largely composed of anorthosites, heavily brecciated (shattered and reassembled) by impact cratering. These highlands offer an opportunity to study the rock types that make up the lunar crust as well as the effects of impact bombardment on the highlands.

One interesting and important lunar rock type contains high concentrations of potassium, rare earths, and other incompatible elements (those that do not fit well into the crystal structure of

TABLE 1. Compositional properties of selected geological units in the Mare Smythii region.

Material	Age*	Al wt%	Mg wt%	Ti wt%	Fe wt%	Th ppm	Comments
Mare Smythii	E/1-2	8.5-10	5->7.4	2.5-3.5	5.6-7.4	2.4	High-Ti mare basalts, thin with admixed highlands debris
Dark mantle	EI/3.5-1	10-11.5	5.8->7.4	2.5-3.5	4.4-5.6	1.2-2.4	Mafic, pyroclastic glasses, probably high-Ti
Basin rim	pN/~4.0	11.5->14	<5-5.8	0.6-1.9	<1.5-4.4	0.5-0.7	Anorthositic debris, breccias
Balmer plains	IN/3.9	10	6.2-7.4	1.4-1.9	7.4-9.6	4.0	Thin mantle of terra debris overlying KREEP basalts
Terra, west of basin	NpN/~4	11.5-13	<5-5.8	1-1.9	5-7.4	0.7	Anorthositic norite breccias
Terra, east of basin	pN/>4	13->14	<5	<0.3-1.4	<1.5-3	0.5	Anorthositic to pure anorthosite breccias

*Relative ages: E—Eratosthenian, I—Imbrian, N—Nectarian, pN—pre-Nectarian. Absolute ages (in billion years) are rough estimates.

common rock-forming minerals). This rock type, called KREEP, may represent the final stages of the crystallization of the original global magma system. KREEP is not uniformly distributed around the Moon, but in the Mare Smythii region, both Balmer and Mare Marginis contain KREEP-rich rocks (Figs. 2e, 3e; Table 1). At Balmer, KREEP basalts apparently underlie a thin covering of highlands debris, whereas at Marginis, the high KREEP appears to be associated with the mare lava flows. Thus, the Mare Smythii region offers the opportunity to study the occurrence and nature of KREEP-rich rocks in two entirely different geologic settings.

In short, the Smythii basin offers a wide variety of geologic units and processes for detailed investigation from a lunar base. From a centrally located base site, a series of traverses can be designed to explore this diverse terrain, as will be discussed below.

Geophysical Considerations

Structurally, the Smythii basin is of geophysical interest because (1) it is similar in size and depth to the younger Orientale basin (*Wilhelms,* 1987) and (2) it formed on a crust thought to be thicker (about 60 to 80 km thick) than that beneath the Apollo 12 and 14 landing sites (45-60 km thick). This thicker crust is inferred from the relative absence of strain-induced grabens in the Smythii region, which implies that subsidence was minimized by a thicker ancient lithosphere (*Solomon and Head,* 1980). Because the lithosphere and the more plastic asthenosphere were probably identical with the differentiated crust and mantle at the time of basin formation, a thicker crust is expected. A regional seismic network near a base in Mare Smythii would test this expectation directly and provide a determination of subsurface wave-velocity structure in a thick crustal zone to complement the Apollo results.

The Smythii and Orientale basins are also structurally similar in that they contain relatively thin mare basalt flows and strong gravity anomalies that imply the existence of subsurface mass concentrations or "mascons." The mascons are caused partly by the surficial mare basalt flows and partly by impact excavation of less dense crustal material, followed by rising of the denser mantle to compensate isostatically for the excavated crustal material. The existence of a relatively thick ancient lithosphere beneath Smythii is believed to have assisted in the preservation of a mascon beneath this basin despite its relatively thin mare fill. Geophysical characterization of the subsurface density structure under the Smythii basin (inferred from seismic and gravity surveys) will therefore provide a general test of models for the structure of mascon basins.

Mare Smythii is adjacent to a large group of swirl-like albedo markings north and east of Mare Marginis (Fig. 2d). Although the origin of these swirls is poorly understood, they are similar to markings found elsewhere on the Moon and are closely associated with strong magnetic anomalies detected from lunar orbit (Fig. 5). A base in Mare Smythii would therefore afford an opportunity to investigate the magnetic anomaly sources. In addition to establishing the nature of the swirls, such an investigation would further constrain the origin of lunar paleomagnetism, an enigma raised by the Apollo data (see *Lunar Geoscience Working Group,* 1986). The swirls have been suggested to be either surface residues of relatively recent cometary impacts or zones of the lunar surface that have been shielded from the ion bombardment of the solar wind by the associated strong magnetic fields. In the latter model, solar-wind hydrogen is considered a necessary part of the process that results in darkening with time (or "optical maturation") of lunar surface materials (*Hood and Williams,* 1988). To complicate matters further, it is likely that transient plasmas produced during hypervelocity impact are responsible for generating short-lived magnetic fields that magnetized some lunar surface materials. Geologic and magnetic investigations at a lunar base may verify this process in detail for the benefit of future paleomagnetic investigations of the Moon and similar bodies in the solar system.

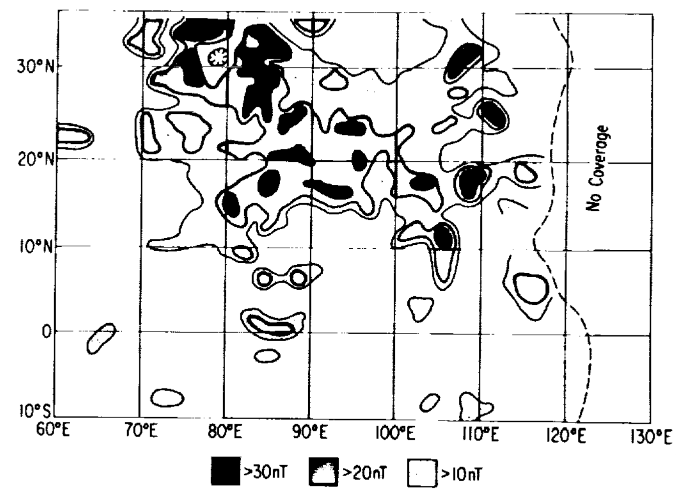

Fig. 5. Amplitude of near-surface magnetic fields in the region of Maria Smythii and Marginis as deduced from the reflection of low-energy electrons (from *Hood and Williams,* 1988).

At a lunar base, it will be important to obtain new heat flow measurements *in situ* to supplement the two Apollo measurements. A determination of the global mean heat flow is important for constraining not only the thermal state of the interior but also the bulk lunar composition (through the inferred abundance of heat-generating radioactive elements such as uranium and thorium). A major deficiency of the Apollo 15 and 17 heat flow determinations is that both were obtained near mare-terra boundaries, transitions between a surface with a thick, insulating megaregolith layer (the highlands) and a surface with a very thin insulating layer (the maria). Because heat flow at such boundaries is expected to be anomalously large, the Apollo measurements may not be representative of the Moon as a whole (*Warren and Rasmussen,* 1987). Although indirect orbital measurements of lateral variations in heat flow may be made prior to the establishment of a lunar base, direct Apollo-type heat flow determinations at additional sites around the Moon will be required to establish the absolute magnitude of global lunar heat flow. Measurements at sites in and around Mare Smythii (or any other circular mare) would allow an evaluation of heat flow as a function of megaregolith thickness. Mare Smythii is also known to be higher in radioactivity than the surrounding highlands (Fig. 3e); heat flow measurements in the Mare Smythii region, combined with orbital measurements of lateral heat flow and of surface abundance of radioactive elements, would therefore contribute ground truth for a more accurate evaluation of mean global heat flow.

Astronomical Considerations

The uses of a permanent lunar base for astronomical observations have been described in detail (*Burns and Mendell,* 1988; *Smith,* 1988). Here we will note the advantages offered by the Mare Smythii site for an astronomical observatory.

As discussed above, it is highly desirable to establish a radio astronomical observatory somewhere on the lunar farside, out of view of our electromagnetically noisy home planet. Perhaps the greatest advantage of the Smythii site is that the radio observatory could be near the main base, but out of Earth radio range. Moreover, the equatorial location of a Smythii site would ensure that the entire sky would be visible over the course of each lunar day (about 28 terrestrial days). The potential for astronomy at wavelengths other than radio is as exciting in this region as at any site on the Moon.

From a geocentric viewpoint, the Moon experiences about 8° of longitudinal libration: any point beyond 98° longitude is never in radio sight of the Earth. However, diffraction effects for very-low-frequency radio waves (*Taylor,* 1988) require that the radio observatory be located an additional 75 km east of this longitude (a degree of lunar longitude at the equator is about 30 km). Thus, the prime location for a lunar radio observatory is on the equator at any longitude greater than 100.5° east or west; at these locations, Earth radio noise does not exist. We suggest that the radio observatory for the Smythii base be an outpost, largely automated, located at 0, 101°E (Fig. 6). For routine maintenance of the observatory, a road could be bulldozed and the observatory serviced by tracked or wheeled vehicles. The observatory would be about 330 km from the main base; on a prepared road, routine speeds of at least 30 km/hr could be achieved, making the transit time to the observatory about 11 hours. Transport and servicing

Fig. 6. Relief map of the Smythii region of the Moon, showing location of the main lunar base, the farside radio astronomical observatory, and permanent communications outpost. Circles indicate a 100-km and 500-km radius of action from the main base. Three model geological traverses (I, II, and III) are described in the text. Base is portion of LOC-2, original scale 1:2,750,000.

could be largely automated or teleoperated, thus greatly minimizing surface exposure risks to the human inhabitants of the lunar base.

The Mare Smythii site offers all the advantages of lunar-based astronomical observation. Its equatorial location would make the whole sky visible and the lunar farside is in close proximity. Other properties intrinsic to the Moon (e.g., hard vacuum, stable platform; see *Smith,* 1988) are as applicable here as at any lunar base site. In conjunction with its numerous other virtues, the Smythii site easily satisfies the criteria of lunar base astronomical users.

Lunar Resource Considerations

A wide variety of potential uses for the indigenous resources of the Moon has been identified (see sections 6, 7, and 8 in *Mendell,* 1985). Although these proposed uses differ widely by process and required feedstock materials, several lunar resources appear to be common to many different schemes. In general order of decreasing usefulness, these resources are bulk regolith, ilmenite (a Ti-rich mineral), volatiles, Al-rich highlands soils, and KREEP-rich material. Each of these materials is abundant at the Mare Smythii site.

The entire surface of the Moon is covered by a mass of fragmental material, ground up by impact bombardment, called regolith. The most important identified use of bulk regolith will be to shield surface habitats from the harsh radiation environment on the Moon (e.g., *Haskin,* 1985). This is the easiest recognized use of lunar materials; loose soil can be bulldozed to cover prefabricated living modules. Moreover, expanding human presence in Earth-Moon space will require shielding at space outposts where humans will live (e.g., in geosynchronous orbit or at the Lagrangian points). Thus, bulk lunar regolith may become one of the first economical lunar exports. Both mare and highlands surrounding the Smythii site are suitable for mining bulk regolith; the old age of the highlands suggests that the regolith is extremely thick in these areas, possibly as thick as 30 m.

The mineral ilmenite is of particular importance in schemes for utilizing lunar resources. Not only would ilmenite be useful in the production of oxygen on the Moon by a reduction process (e.g., *Gibson and Knudsen,* 1985), but ilmenite-rich soils contain high concentrations of ^{3}He, implanted on the grains by the solar wind over geologic time. This ^{3}He, used in terrestrial nuclear fusion reactors, could become the most profitable lunar export resource (*Wittenberg et al.,* 1986). Ilmenite is abundant in the mare basalts from the Apollo 11 and 17 landing sites; remote-sensing data show that these high-Ti basalts are widespread within the maria, including Mare Smythii (Fig. 3g; Table 1). Photogeological evidence suggests that the mare basalts here are relatively thin, but a significant amount of the observed soil chemistry of Mare Smythii is contributed by underlying highlands debris, added to the soils by vertical impact mixing (e.g., *Rhodes,* 1977). This observation and the observed relatively high Th content of Smythii soils (Table 1) suggest that the basalts of Smythii are similar in composition to the Apollo 11 high-K subgroup of high-Ti basalts. These basalts contain about 20% by volume ilmenite and about 7 wt% Ti (*BVSP,* 1981). Thus, the Mare Smythii basalts are prime candidates for any mining process that requires large amounts of ilmenite.

Volatile elements appear to be rare on the Moon. However, notable concentrations, including zinc, sulfur, and lead, are found on the surfaces of lunar pyroclastic glasses. Not only are these materials important for what they tell us about the indigenous lunar volatile content, but they also constitute a potential resource. As noted above, dark mantle pyroclastics are abundant in the Mare Smythii region (Figs. 2 and 3a) and are present in minable quantities on the basin floor. Because they are of small extent, we cannot be certain of their composition; however, pyroclastics found at the Apollo sites appear to be broadly similar in composition to their associated mare basalts. Thus, the Smythii pyroclastics are probably also of the high-Ti variety.

The highlands surrounding Mare Smythii display some of the highest Al concentrations seen in the Apollo orbital data (Fig. 3c). This suggests the presence of nearly pure anorthositic soils, an Al-rich material that is readily usable for construction on the lunar surface and in Earth-Moon space. One proposed process, which requires such soils, involves fluorination of anorthite (the major mineral in anorthosites) to produce both oxygen and aluminum (*Burt,* 1988).

KREEP, a material rich in trace-elements, is also available at the Smythii site, and it may ultimately be needed for phosphorus to support lunar agriculture. Although its extraction is probably an element of the advanced lunar base, it is fortunate that significant KREEP deposits are near the proposed base site.

Virtually every use of lunar resources that has been thus far proposed can be accomplished at a base site within Mare Smythii. Thus, the geological diversity that makes the Smythii site such an attractive candidate for geoscience exploration also makes it a prime candidate for lunar resource exploration and utilization.

Summary

The Mare Smythii region has several attractive attributes for the siting of a permanent lunar base. Its location on the equator and limb combines the best of the nearside and farside base advantages and permits easy access to the lunar surface from the supporting LEO spaceport. The geological diversity of the region, which contains mare basalts, pyroclastics, KREEP-rich rocks, and aluminous highland soils, permits a wide variety of surface scientific exploration and resource utilization. The regional context of the Smythii site is significantly different from the Apollo sites, thus enabling detailed comparative geophysical studies. A base established in Mare Smythii has the potential to service the various scientific and engineering users of such a base from one central location.

GEOLOGICAL FIELD INVESTIGATIONS AT SMYTHII BASE

Having detailed the numerous merits of the Smythii region for a lunar base, we now briefly discuss some model studies that could be carried out from such a base. For the purposes of this discussion, we will tentatively place the main base site at exactly 0, 90°E (Fig. 6); this site is on the high-Ti basalts of Mare Smythii and is a good location to utilize the ilmenite resources of the mare floor for both oxygen production and possible mining of ^{3}He. Moreover, the smooth, flat surface of Mare Smythii will also be conducive to the ultimate construction of a mass driver, thus making the export of lunar material cheap and reliable. Because the Earth will be out of radio view from this site on some occasions during the libration cycle, we show a communications outpost on the west rim of the Smythii basin at about 0, 81°E (Fig. 6). Here, a radio installation will have a permanent view of the Earth for base communications purposes; communication

from this outpost to the main base ultimately will be established by direct optical link. For base start-up operations, either a surface relay network or a temporary comsat will provide a continuous radio link with the Earth.

We describe below three separate strategies for geological exploration based on the distance from the main base to features of interest at ranges of <100 km, 100-500 km, and >500 km from the base (Fig. 6). For extensive traverses beyond 100 km, we envision that most geological exploration will be by teleoperated robots (*Spudis and Taylor,* 1991) that would be directly controlled by geologists who remain at the main base site or, possibly, from the Earth. These robots have many advantages over human field workers and could effectively conduct most of the exploration advocated here. Follow-up visits by human geologists are assumed; these visits would largely consist of quick sorties to minimize extensive and complex life support systems and risks from radiation. Although planning for detailed traverses and field work cannot be done until a base site is selected, the following exploration plans are offered as examples that could be undertaken from a base in Mare Smythii.

Near-Base Activities (<100-km Radius)

In the early stages of base establishment, most geological work will probably be done near the base site; also, several scientific problems lend themselves well to near-base work even after the longer traverses begin. Thus, fieldwork near the base will start at the time of base emplacement and continue into the indefinite future.

The base's location on the basalt flows of Mare Smythii will provide the opportunity to study both regolith formation and lava stratigraphy. To determine the complete history of regolith formation and evolution, it will be most useful to bulldoze a pit down to bedrock (at this site, probably no more than a couple of meters, because of the young age of the Smythii lavas). The early stages of regolith growth are still almost completely unknown; within this pit, we can study the bedrock interface and address questions of grain-size evolution and soil maturity as study proceeds upsection. The sequence of lava flows and possible changes in magma composition with time can also be studied at the base site; this study of the regional bedrock unit can be done either directly (by shallow drilling and coring of the basalt flows) or indirectly (through the sampling of the ejecta from small craters in the mare to reconstruct possible subsurface layering).

Small particles of highlands rocks are found in the soils of all Apollo mare landing sites, and the aluminous composition of Mare Smythii (Table 1) suggests that this site is no exception. Most of these particles are derived from directly beneath the surface flows by vertical impact mixing of sublava highlands terrain. Thus, even though the base will be located on the mare flows, samples of the terra basin-floor materials will be available within the mare soils. Determining the composition and history of the highlands surrounding the Smythii base site will be one of the prime long-range tasks for the base geologists.

Farther afield, both the extensive dark-mantle pyroclastics and the internally modified floor-fractured craters are within 100 km of the main base site [Figs. 2a-c, 3a, 6 (I)]. The pyroclastics should be sampled to determine their place in the general volcanic history of the Smythii region, their possible compositional affinities to the mare basalt flows, the nature of their mantle source regions beneath the crust in this area, and their potential as minable resources. The floor-fractured crater Purkyne U (Fig. 2c) lies about 60 km east-southeast of the base site; this crater has an uplifted, fractured floor, partial fill by mare basalt (erupted from within the crater interior, as demonstrated by its unbreached rim), and a partial covering of dark pyroclastic material. Detailed field study of this crater could elucidate the processes of internal modification of lunar craters and contribute to our understanding of the volcanic history of the basin.

In addition to these primary studies, several smaller-scale ones will be conducted in the near-base area. These tasks will include study of a large population of small (<1-km diameter) craters to understand their formational mechanics and the regional cratering history, the study of lateral variations in both the lava flows and the subfloor basement, and investigations of the nature of crater rays. (This area is covered by rays from distant craters and it is important to establish the exact amounts of crater primary ejecta contained in ray material). These studies alone are of significant importance and complexity to provide the base geologists with challenging exploration opportunities.

Short-Range Traverse Activities (100-500-km Radius)

The middle range of exploration traverses is illustrated by the 500-km circle of Fig. 6. In this range, almost all the diverse geological features of the Smythii region are available for study. Model traverse route II (Fig. 6) could be followed by a teleoperated robot investigating the materials and processes described below. Undoubtedly, significant discoveries made along the way will perturb the actual route, but route II as shown encompasses most of the currently identified field geology goals.

In the first leg of the traverse, the lateral heterogeneity of the young Smythii lava flows north of the base will be investigated (Fig. 6). The north rim of the basin will be sampled to determine its relation to materials of the basin floor, collected near the base site (see above). Next, the traverse will continue north into the lava flows of the Mare Marginis basin (Fig. 2d). These lavas also appear to be relatively young (about 2 b.y. old); moreover, they are enriched in Th (up to 3.4 ppm; Fig. 3e). This suggests that they are a variety of KREEP-rich mare basalt, rare in the Apollo collections, and their study could shed light on the process of igneous assimilation of KREEP into mare basalt magmas.

The traverse will continue north to sample and investigate the mysterious swirl materials of northern Marginis (Fig. 2d). As described above, these swirls are associated with large surface magnetic anomalies (Fig. 5) and field studies of their composition and local environment are required to fully understand their origin. It would also be of interest to visit the crater Goddard A, as it has been proposed that this crater may be related to the Marginis swirls.

The traverse will now turn south, across Mare Marginis to determine its lateral variations, cross the mare-filled crater Neper, and return to the Smythii basin. One goal of this leg is to examine the lateral variations of the highland deposits making up the Smythii basin rim. The trip will continue south into the basin to examine and explore the floor-fractured craters Schubert C (Fig. 2b), Haldane, and Kiess. These craters display a range of modification states, and comparative studies between them and the previously studied Purkyne U (see above) will enable a resolution of the problem of their origin. In addition, this leg of the traverse covers the most abundant dark mantle deposits and local volcanic vents of the region (Figs. 2b and 3a). Field study of these features will aid in a detailed reconstruction of the volcanic history of the Smythii basin.

In the final leg of this traverse, we will study the highlands of the Smythii basin's south and west rims (Fig. 6). When this leg is completed, we will have a fairly complete knowledge of the lateral variations in basin rim deposits. We may even find evidence for large-scale compositional zoning within the basin ejecta deposits, a feature long postulated for basin geology based on incomplete and inadequate remote-sensing data, but as yet unproven on the Moon. This geological traverse provides a variety of features and processes for direct study, all within a fairly short traverse radius.

Long-Range Traverse Activities (>500-km Radius)

Beyond the 500-km limit, virtually the entire Moon beckons for detailed exploration. Indeed, one of the advantages of the teleoperated robot system is that it turns a single-site base into a "global base" by providing access to any point on the Moon (*Spudis and Taylor,* 1991). For the purpose of brevity, we here restrict our attention to a long-range traverse likely to be undertaken early in operations from the base, a mission to explore and sample the intriguing Balmer basin (Fig. 6, III).

As noted previously, Balmer is an old multiring basin apparently filled with light plains materials of Imbrian age (Figs. 2e and 3a). This otherwise unremarkable basin is worth investigating for two reasons: (1) the light plains that fill Balmer display dark-halo craters (Fig. 2e), indicating the presence of a subsurface basalt unit at least 3.9 b.y. old; and (2) orbital gamma-ray data suggest that this area is rich in KREEP (Fig. 3e), having a local Th concentration of 4 ppm. Moreover, this Th enrichment is coincident with the plains displaying dark-halo craters, suggesting that the KREEP component is associated with the underlying, ancient lava flows. In combination, these observations suggest the presence of ancient KREEP-rich basalt flows; flows of this composition have long been postulated in the lunar literature, but thus far we have identified only one example, the planar Apennine Bench formation near the Apollo 15 landing site. Because the concept of KREEP volcanism is so important to models of lunar evolution and because of the controversy over its existence, we have specifically planned this traverse to examine and characterize the volcanic fill of the Balmer basin.

The traverse begins by exploring the southwestern floor and rim of the Smythii basin, previously unvisited, to determine more completely the nature of the highlands around Mare Smythii and to provide comparative data for the previous traverses. This route includes a complete traverse of the crater Ansgarius; not only can we investigate the geology of this large, complex crater of Imbrian age, but this location also demarcates the crest of the outermost ring of the Smythii impact basin. The internal structure of this basin ring may be exposed within the walls of Ansgarius, thus making the detailed geologic structure of the ring available for study.

The traverse next proceeds to the plains of the Balmer basin. The goals in this area include characterization of the Imbrian-age light plains to determine their provenance and study of the dark-halo craters to understand their internal structure and ejecta. It is within the ejecta of these craters that we hope to find the long-sought KREEP basalts; through study of the ejecta volumes and their distribution around the craters, we can estimate the thickness of the overlying highlands debris mantle and, possibly, the thickness of the buried ancient basalts. Another important goal at this stop is study of regolith developed on the ejecta blankets of the dark-halo craters to understand how they form the strong photometric contrast seen in orbital photographs. These tasks involve intensive fieldwork; an advantage of using robots here rather than human field geologists is that as much time as is required can be spent in the field area to completely understand and solve these problems.

On the return trip to base, we will investigate the west basin rim and the light plains fill of the craters Gilbert and Kastner G (Fig. 6). At these two craters, an important question is the possible relation of their plains fill to that in the Balmer basin. If these light plains are related to the Crisium basin to the north (Fig. 1), these stops will test the concept of lateral variation in basin debris blankets and could also address the vexing question of primary basin ejecta vs. locally reworked material in highland plains materials. On the final leg, we will continue previously started field studies of the Smythii basin floor material, dark-mantle deposits and vents, and a previously unvisited floor-fractured crater, Runge (Figs. 3a and 6).

Summary

These three strategies of geological exploration demonstrate the amazing variety of geological units and processes that are available for direct exploration at the Smythii base site. The units represent the range of lunar geologic processes and absolute ages, from the ancient brecciated highlands crust to the youngest, rayed craters. Many additional traverses could be described; moreover, after a short time of base operations, many significant new discoveries will undoubtedly be made, thus altering the order of exploration priorities and planning of the actual routes. The total potential of a lunar base for geologic study is of such magnitude that it is impossible to predict the exact schedule and order of surface operations.

GEOPHYSICAL FIELD INVESTIGATIONS AT SMYTHII BASE

Following the order outlined above for geological exploration, we divide the discussion of geophysical exploration into categories depending upon the maximum radial traverse distance from the base.

Near-Base Activities (<100-km Radius)

After base establishment, the first priority for geophysical studies should be the deployment of an Apollo-type geophysical station containing such instruments as a seismometer, heat flow probe, magnetometer, and solar-wind spectrometer. These instruments should be emplaced near enough to the base to allow easy access for maintenance and recalibration but far enough away so that base activities do not add an undue amount of artificial noise to the measurements. The structure of near-surface seismic wave velocities can be determined using active sources, perhaps in conjunction with construction or mining activities. To deduce the structure at greater depths, using a single-station seismometer, will require active energy sources of increasing magnitude, comparable to those produced by the planned crashes of LM ascent modules and S-IVB stages during the Apollo program. Measurements from a single heat flow probe should be monitored for at least a year to establish the thermal properties of the surrounding regolith, which are needed for heat flow determination. The final value, if obtained away from the periphery of the mare, will provide a valuable benchmark for comparison with the Apollo values. A single magnetometer and solar-wind spectrome-

ter will define the local crustal magnetic strength at the base site and determine the extent of deflection by this magnetic field of ions in the solar wind.

The next order of priority after establishing the base geophysical station is to conduct field geophysical measurements during the surface geological traverses. A local area network of seismic stations should be emplaced to allow passive seismic studies using meteoroid impacts and shallow moonquake sources. Active seismic sounding using artificial sources will also be very effective using this local array. Heat flow probes can be deployed at a series of sites on different megaregolith thicknesses to obtain a first determination of the dependence of lunar heat flow on this quantity. During exploratory traverses, it will be desirable to obtain direct surface gravity and elevation measurements at specific points along the route to constrain later modeling studies of subsurface density structure. These measurements will provide a ground-truth supplement to Apollo and LO orbital gravity and topography data. Also, magnetic field and solar-wind flux measurements along the traverse will provide the first direct measurements of solar-wind deflection as a function of surface magnetic field intensity and direction. The surface magnetic field measurements, combined with orbital magnetic data, will also facilitate modeling of the bulk magnetization properties of large-scale geologic units (e.g., mare basalt flows) in order to constrain the nature and origin of lunar paleomagnetism.

Short-Range Traverse Activities (100-500-km Radius)

Several important geophysical investigations can be made during the medium-range traverse discussed above (II in Fig. 6). This traverse will enable the deployment of one or more geophysical stations that will become part of a regional network designed to determine the subsurface structure and thermal state of the Smythii region. The primary instruments to be deployed at these stations will be seismometers and heat flow probes. In order to resolve basin structure, individual seismic stations should be no more than about 150 km (5°) apart, requiring at least 8 regional stations in addition to the base station. Active seismic sounding near at least one of the highland stations will allow the first direct crustal thickness determination at a highland site on the Moon. As noted previously, the crust is expected to be substantially thicker in this region than at the Apollo sites. Crustal thickness peripheral to the basin will be larger still because of the expected isostatic raising of the crust-mantle boundary beneath the basin center. Following establishment of both mare and terra seismic velocity and thickness benchmarks using active methods, the passive network will be capable of a first-order determination of the velocity structure beneath the entire basin. In combination with gravity and topography surveys, the two-dimensional velocity model will provide strong constraints on the subsurface composition and density structure of this mascon basin. Heat flow measurements will likewise establish the lateral variation of surface heat flow and probable subsurface thermal state as a function of radial distance from the basin center.

A traverse to the Mare Marginis swirl belt will make possible direct surface magnetometer and solar-wind spectrometer measurements at the site of one of the largest magnetic anomalies on the Moon (Fig. 5). As stated above, simultaneous geologic investigation and sampling of the swirls should establish their origin. As a by-product of these investigations, solar-wind spectrometer measurements will determine the lateral variation of the implantation rate of solar-wind gases (mostly hydrogen and helium) into the uppermost regolith. For example, the strongest lunar magnetic anomalies are probably capable of completely deflecting bombardment by ions of the solar wind (*Hood and Williams*, 1988). This process will lead to zones of relatively low implantation rates near the centers of large surface anomalies and zones of high implantation rates in complex, curvilinear areas peripheral to the same anomalies. Measurement of these fluxes will be helpful for evaluating the volatile resource potential (i.e., the extraction efficiency of trapped solar wind gases; *Haskin*, 1985) of different source regions. Furthermore, the strongest magnetic anomalies are characterized by surface field amplitudes that probably exceed several hundredths of a Gauss (for comparison, the Earth's field near the equator at the surface is about 0.3 G). Depending upon their horizontal scale, these relatively strong crustal fields may be capable of significantly deflecting a part of the solar cosmic ray flux during flare events. If such deflection is beneficial in reducing the hard radiation environment for human activities, then it may even be desirable to locate outposts or bases within the shelter of strong magnetic anomalies.

Long-Range Traverse Activities (>500-km Radius)

When the robotic field explorations are extended to greater distances, identified geologic targets can be characterized using geophysical methods. For example, along the suggested route III of Fig. 6, small-scale seismic sounding and surface gravity measurements may be useful in delineating the thickness of subsurface basalt units in the Balmer basin and in identifying the crest of the outermost ring of the Smythii basin. More generally, remote geophysical stations may be deployed to allow seismic and electromagnetic sounding of the deeper lunar interior. These stations would be part of a global-scale network that should be established in the course of continuing field investigations. Among the major objectives of large-scale seismic and electromagnetic sounding network studies are determinations of the seismic-velocity profile of the lunar mantle and of the existence and size of a possible metallic core. Although core detection may be achieved earlier through alternative approaches, a detailed characterization of the size, mass, and physical properties of the core will probably require long-term seismic measurements using a large number of stations. Similarly, a more accurate appraisal of mantle structure and thermal state will need both long-term and large-scale seismic, electromagnetic, and heat flow measurements. Thus, the geophysical stations deployed in the course of lunar base activities and traverses will contribute significantly to an eventual accurate determination of the structure, composition, and thermal state of the deep lunar interior. Because the bulk composition of the Moon (including core size and mass) is a basic constraint of lunar origin models, such a determination will lead to a much improved understanding of the origin of the Moon.

CONCLUSIONS

We have demonstrated that the Mare Smythii region holds great promise as a lunar base site from a scientific, operational, and resource utilization viewpoint. This site enables enough flexibility to satisfy any potential lunar base user. Among its attributes are the following:

1. Its location on the lunar limb permits the establishment of a base complex that combines the benefits of a nearside base (for ease of initial and routine base operations) and a farside base (to

shield the radio astronomical observatory from the electromagnetic interference produced by the Earth).

2. Its equatorial location allows for easy base access from the LEO space station and also permits a clear view of the entire sky for astronomical observations.

3. The Smythii region abounds in a diversity of both geologic features and natural resources. This diversity permits a wide range of geological and geophysical investigations to be performed and it also provides almost the entire known range of potential lunar resources to be mined, processed, and used on the Moon and in Earth-Moon space.

Nearly all the identified lunar geoscience problems can be addressed at a base located in Mare Smythii. Some of these problems are the origin and evolution of the lunar crust and mantle, the cratering history of the Moon, the formational mechanics of large craters and basins, the nature and evolution of the lunar regolith, the origins of lunar paleomagnetism, and lunar volcanic history. Geologic and geophysical field studies conducted from the Smythii base will provide data applicable to all these problems.

Based on our study of the Smythii region, we have tentatively identified the following operational requirements for base establishment and initial operations:

1. We propose that the main lunar base be located at 0, 90°E, in Mare Smythii. This location will provide high-Ti mare regolith as a feedstock for oxygen production and possibly ^{3}He mining, and it allows easy access to a variety of important geological and geophysical exploration targets.

2. At least two installations will be required in addition to the main base. The first is a communications outpost on the west rim of the basin at about 0, 81°E. This site is in constant radio view of the Earth; the outpost will be needed as a relay station when the main base is out of contact with Earth during minimum libration cycles. The outpost will be connected to the main base by an optical link cable emplaced during base start-up; interim Earth-Moon communications can be provided by a temporary lunar comsat.

3. The second outpost could be a lunar very low frequency radio astronomy observatory. It should be located on the equator east of 100.5°E; we suggest an intercrater area at 0, 101°E, where the observatory will be permanently shielded from radio noise from the Earth. The suggested site is about 330 km from the main base; a road can be constructed to allow easy rover access to service the outpost.

4. As we envision base operations, most geological field work and emplacement of geophysical instruments can be done by teleoperated robots. Some visits by humans to sites distant from the base will be required.

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REFERENCES

Arnold J. R. (1979) Ice in the lunar polar regions. *J. Geophys. Res., 84,* 5659-5668.

Burke J. D. (1985) Merits of a lunar polar base location. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 77-84. Lunar and Planetary Institute, Houston.

Burns J. O. and Mendell W. W., eds. (1988) *Future Astronomical Observatories on the Moon.* NASA CP-2489. 134 pp.

Burt D. M. (1988) Lunar mining of oxygen using fluorine (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 43. Lunar and Planetary Institute, Houston.

BVSP (Basaltic Volcanism Study Project) (1981) *Basaltic Volcanism on the Terrestrial Planets.* Pergamon, New York. 1286 pp.

Duke M. B., Mendell W. W., and Roberts B. B. (1985) Strategies for a permanent lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 57-68. Lunar and Planetary Institute, Houston.

Gibson M. A. and Knudsen C. W. (1985) Lunar oxygen production from ilmenite. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 543-550. Lunar and Planetary Institute, Houston.

Haines E. L., Etchegaray-Ramirez M. I., and Metzger A. E. (1978) Thorium concentrations in the lunar surface. II: Deconvolution modeling and its application to the regions of Aristarchus and Mare Smythii. *Proc. Lunar Planet. Sci. Conf. 9th,* pp. 2985-3013.

Haskin L. A. (1985) Toward a spartan scenario for the use of lunar materials. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 435-444. Lunar and Planetary Institute, Houston.

Hood L. L. and Williams C. R. (1988) The lunar swirls: Distribution and possible origins. *Proc. Lunar Planet. Sci. Conf. 19th,* pp. 99-113.

La Jolla Consortium (1977) Global maps of lunar geochemical, geophysical, and geologic variables. *Proc. Lunar Sci. Conf. 8th,* frontispiece, 25 plates.

Lunar Geoscience Working Group (1986) *Status and Future of Lunar Geoscience.* NASA SP-484, Washington, DC. 54 pp.

Mendell W. W., ed. (1985) *Lunar Bases and Space Activities of the 21st Century.* Lunar and Planetary Institute, Houston. 865 pp.

Rhodes J. M. (1977) Some compositional aspects of lunar regolith evolution. *Philos. Trans. R. Soc. London, A285,* 293-301.

Schultz P. H. (1976) Floor-fractured lunar craters. *Moon, 15,* 241-273.

Smith H. J. (1988) Why the Moon is the best place in the solar system from which to do astronomy (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 266. Lunar and Planetary Institute, Houston.

Solomon S. C. and Head J. W. (1980) Lunar mascon basins: Lava filling, tectonics, and evolution of the lithosphere. *Rev. Geophys. Space Phys., 18,* 107-141.

Spudis P. D. and Taylor G. J. (1992) The roles of humans and robots as field geologists on the Moon. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Taylor G. J. (1989) Site selection criteria. In *A Lunar Far-Side Very Low Frequency Array,* pp. 61-63. NASA CP-3039.

Warren P. H. and Rasmussen K. L. (1987) Megaregolith insulation, internal temperatures, and bulk uranium content of the Moon. *J. Geophys. Res., 92,* 3453-3465.

Wilhelms D. E. (1987) *The Geologic History of the Moon.* U.S. Geol. Surv. Prof. Paper 1348. 302 pp.

Wilhelms D. E. and El-Baz F. (1977) Geologic map of the east side of the Moon. *U.S. Geol. Surv. Map I-948.*

Wittenberg L. J., Santarius J. F., and Kulcinski G. L. (1986) Lunar source of ^{3}He for commercial fusion power. *Fusion Technol., 10,* 167-178.

A LUNAR POLAR EXPEDITION

N93-17433

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Advanced exploration and development in harsh environments require mastery of basic human survival skills. Expeditions into the lethal climates of Earth's polar regions offer useful lessons for tomorrow's lunar pioneers. In Arctic and Antarctic exploration, "wintering over" was a crucial milestone. The ability to establish a supply base and survive months of polar cold and darkness made extensive travel and exploration possible. Because of the possibility of near-constant solar illumination, the lunar polar regions, unlike Earth's, may offer the most hospitable site for habitation. The World Space Foundation is examining a scenario for establishing a five-person expeditionary team on the lunar north pole for one year. This paper is a status report on a point design addressing site selection, transportation, power, and life support requirements.

POLAR EXPLORATION AND LUNAR OBJECTIVES

In March 1899, almost one hundred years ago, the explorer Carsten E. Borchgrevnik established the first winter camp on the "white continent," Antarctica. Unlike the north polar regions, Antarctica had never been inhabited by man. Though marine birds and animals visit the coastal regions, only primitive moss and lichen can survive the polar deserts of ice and snow. In the winter, no creature lives on the ice cap. Louis Bernacchi, a young Australian physicist who had joined the expedition, called it "... a land of unsurpassed desolation."

The International Geographical Congress, meeting in London during July, 1895, determined to make Antarctica the target of new exploration, launching an era of government-sponsored national expeditions. The British, especially, were planning a large-scale scientific expedition under the aegis of the Royal Geographic Society. But the Norwegian-born Australian, Borchgrevnik, was determined to get to the Antarctic and spend a winter there before anyone else. He sought private funding and found it in the fourth estate, a wealthy British publisher, Sir George Newnes.

Landing operations from their ship, Southern Cross, took 10 days to complete, including assembly of 2 prefabricated huts joined by a center section for ease of movement. During their 10-month winter camp, and before their relief on 28 January, 1900, expedition members collected specimens, made meteorological observations, and determined that the south magnetic pole was much farther north and east than previously supposed. Perhaps more importantly, they produced the first reliable charts of the Great Ice Barrier and conducted the first explorations by dog sled on the Ross Ice Shelf (*Kirwin,* 1960; *Huntford,* 1984; *Allen et al.,* 1985).

The expedition, despite its limited resources, achieved a lot. The scientific observations and collections, though not extensive, comprised a useful addition to detailed knowledge of Antarctica. But the expedition's true role was that of a reconnaissance team for the large, well-equipped expeditions then being planned. It proved that a party could winter ashore with comparative safety and carry out routine scientific work. Borchgrevnik's winter camp laid the groundwork and tested the techniques for a "golden age" of polar exploration in the following decades. These explorations would culminate in Admiral Robert Peary's attainment of the North Pole on 6 April, 1909, and Roald Amundson's magnificently planned expedition reaching the South Pole on 14 December, 1911. Today there are permanent residents in both the Arctic and Antarctic pursuing commercial and scientific activities. Indeed, the International Antarctic Treaty may prove a useful example for those trying to determine who "owns" the Moon.

Unlike Earth's polar regions, the lunar poles may be the most hospitable locations for early long-term human habitats, so we do not wish to press the polar exploration analogy too far. However, the critical importance of wintering over does apply to lunar exploration. The demonstrated ability to survive on the Moon for long periods is essential for advanced exploration, development, and settlement. The balance of this paper will address the requirements and advantages of an early long-duration mission to the lunar north pole.

DESIGN PHILOSOPHY

We propose a one-year stay on the lunar surface by five people as a thorough demonstration of our ability to live and work on the Moon. While a number of approaches to establishing a lunar base have been proposed, it is our intent to show that very little in the way of new technology or overly complex systems are required to demonstrate our capability to live and work on the Moon for extended periods. Specific transportation systems, base structures, life support, communications, and other techniques described may be very different from those employed for the first lunar base. However, a plausible approach is described here that would permit the next major step in lunar development within a decade for a reasonable expenditure (compared to Apollo or the space station), given our understanding of space systems to be developed for other purposes.

Several assumptions were made to guide this approach: (1) the poles afford the most hospitable environment; (2) no artificial gravity is required; (3) maximal use is made of flight-proven systems to minimize development risk and cost (sometimes with greater operations cost); (4) an abort or rescue option (except during lunar ascent) always exists; (5) an Earth-based rescue vehicle is available with a few weeks' notice; (6) the first base is a nucleus for subsequent operations; and (7) it must be kept simple.

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These somewhat arbitrary constraints are reasonable in terms of the engineering required. Air, food, and water requirements would be the same anywhere on the Moon. Near-constant illumination at a polar site may provide a decisive advantage in terms of power subsystem mass by eliminating the 14-day storage requirement and corresponding need for extra power generation during the day, or for nuclear power. While not necessarily indicative of future experience on the Moon, attempts to use nuclear power in Antarctica have been less than satisfactory. In 1962, a 1500-kW nuclear power plant was installed at McMurdo Base. It was decommissioned 10 years later after a history of fire, radiation leakage, and shutdown. The site was quarantined for six years and 11,000 cu m of contaminated rock had to be shipped back to the U.S. (*Allen et al.,* 1985).

Other advantages of a polar site include the ease of heat rejection, unobstructed astronomy, volatile resources evaluation and prospecting, and possible phenomena related to the unique thermal regime.

LUNAR ORBITER DATA

Our knowledge of the Moon's polar regions is derived from lunar orbiter photography. Five spacecraft were launched at three-month intervals between August 10, 1966, and August 1, 1967. The primary mission of the lunar orbiters was to identify safe landing sites on the Moon's nearside for the Apollo program. Target areas were photographed on film, which was then processed on board the spacecraft. These film frames were scanned for transmission to Earth where video signals were reconverted into photographic images.

Orbiters I-III successfully met all mission objectives, and consequently Orbiters IV and V were retargeted to provide photography of general scientific interest. They were placed in near-polar orbits from which virtually any part of the Moon could be photographed. Lunar Orbiter IV returned 13 high-resolution frames of the north polar region during its 70-day mission. Polar viewing was generally from an altitude of 2500-3500 km and resolution was approximately 100 m (*Hansen,* 1970). Maps prepared from this data in 1981 (U.S.G.S. 1:5,000,000 map I-1326-A) are based on the Apollo control system of 1973 and may contain positional discrepancies at the poles of several kilometers (*Kosofsky and El-Baz,* 1970; *Kuiper et al.,* 1967; *Hall,* 1981).

The Moon's axis of rotation is inclined 1.5° off the normal to the ecliptic. Using the standard lunar reference datum radius of 1738 km, were the Moon a perfect sphere, an object 595 m in elevation at the pole would be in sunlight even during extremes of libration. This is a modest elevation for crater features comparable to Peary Crater, but the elevation of the surrounding terrain is unknown. If it is typical of other highland regions, elevations 2-3 km higher than the datum are not unusual. Nonetheless, the area is illuminated in the photographs we have now. The radar altimeter planned for the Lunar Geoscience Orbiter (LGO) will provide useful new topographic data and improved geodesy will refine positional errors. In the meantime, stereometric information is available in the overlapping Lunar Orbiter IV frames and investigations are underway to estimate the heights of local features relative to the floor of Peary Crater.

It is possible that solar illumination may be periodically interrupted by shadows cast from nearby features. If these can be identified, the difficulty may be surmounted by careful choice of position or by increasing the height of the support tower for the solar arrays. Illumination will certainly be interrupted for a few hours during eclipse cycles, and this eventuality is anticipated in the reserve power subsystem design.

BASE CHARACTERISTICS

Features of this lunar base unique to its polar location have been considered in some detail, while more generic features such as habitat design, architectural layout, and equipment design have been better described by other investigators. Masses have been estimated for generic and site-specific equipment, to arrive at a total of 30,000 kg of cargo brought by four autolanders to support the crew of five for one year (Table 1).

TABLE 1. Polar base mass summary.

Category	Mass (kg)	Remarks
Habitat	7,500	40 m^3/person, 2 modules (approx. size of Spacelab double segment)
Safe Haven	2,500	4 m^3/person, isolated
Power	3,800	3 kWe/person, includes heat rejection
Consumables	8,200	only water recycled
General Equipment	2,700	science, technology, loader, miscellaneous
Rovers	600	two units, two-person each
EVA Suits	1,300	3 per person, not including flight
Personal Effects	400	85 kg per person
Reserve (10%)	3000	
Total	30,000	

Includes all cargo carried aboard autolanders. Does not include crew and their flight spacesuit/backpacks that are brought aboard the crew lander.

Power

The crew's first order of business after landing will be to verify that the autolander payload integrity is sufficient to permit an extended stay on the surface. (Initial verification will have been provided via telemetry long before the crew leaves Earth.) Within a few hours of landing, the rovers will be deployed and loaded with solar panel packages and the packaged erectable power tower. A check of stellar, solar, and topographic positions will verify that the location chosen before launch for the power tower is adequate. Three crewmembers will erect the Astromast-type tower, attaching solar panels and guy wires as the tower is motor- or hand-driven to its approximately 100-m height. The other two crewmembers, in one of the rovers, will lay up to 2 km of cable between the tower and base equipment.

The site tentatively chosen based on available lunar orbiter imagery lies at the intersection of the rim of Peary Crater and the smaller crater containing the north pole. (We refer to this as Polaris Crater. In fact, this crater may or may not contain the pole. It is about 8 km in diameter, which is about the same as the geodetic uncertainty in the polar region.) This intersection is toward the west limb (as viewed from Earth) at about the eight o'clock position on Polaris' rim (Figs. 1 and 2). Three ridges formed by the rims of Peary and Polaris afford steep slopes facing in three different directions down which the solar blankets may be rolled in the event no suitable location for the tower is found immediately. At any one time, it appears that one of these slopes will be illuminated, at least during the landing period of "high" sun elevation when the sun is 1.5° above the ideal horizon.

Batteries permit up to 15 hr of energy storage for discharge at a standby level of 500 W per person. Additional power may be possible by salvaging the autolander batteries, which are not a part

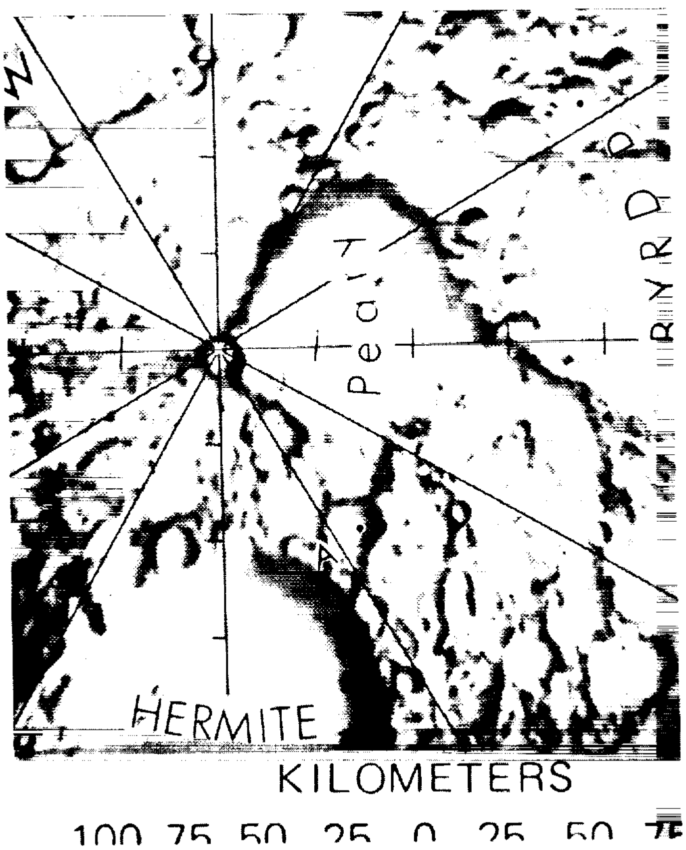

Fig. 1. Enlarged segment of U.S.G.S. Map I-1326 showing north polar region. The polar position shown here appears inside Polaris Crater on the rim of Peary. Actual position of the pole is uncertain.

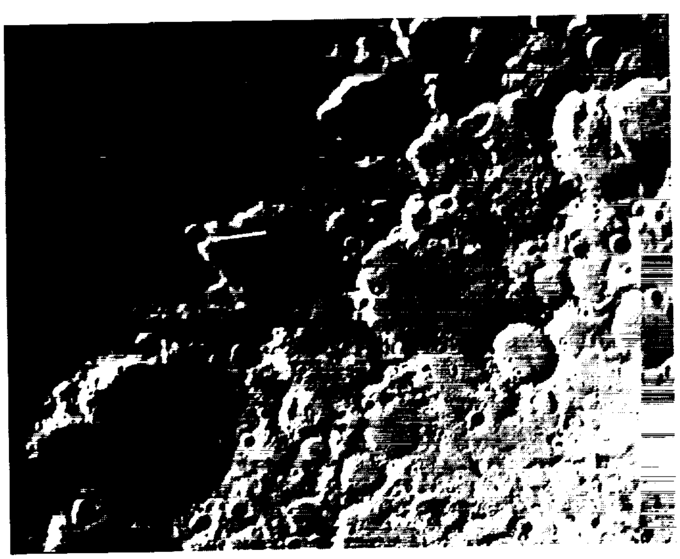

Fig. 2. Lunar Orbiter IV image showing nearside north polar region. Peary is the large crater containing the center of the frame, while Polaris is near the center of the top edge. Photo: NASA and U.S.G.S.

of the base mass budget. This energy reserve is sufficient to handle eclipses and solar obstruction by terrain features up to 8° in angular width as seen from the power tower (Table 2).

TABLE 2. Power subsystem mass summary.

Component	Mass (kg)	Remarks
Solar Panels/Blankets	1500	Si, 25% illumination factor (250 W/kg)
Cable and Connectors	1500	Depends on voltage, AC/DC?
Power Tower	250	Up to 100 m
Secondary Batteries	375	Eclipse, shadow power (37 kWhr)
Conditioning and Controls	175	External to habitat
Total	3800	

Not including equipment integral to habitats, or batteries aboard rovers, loader, or autolanders.

Habitat

No specific habitat design was considered because many different configurations and construction techniques have been proposed. The mass was estimated by comparison of published sources. One possible approach will be to have two cylindrical modules with airlocks carried by two of the autolanders in condition ready for occupation. Another autolander will carry a half-ton loader/crane that can be loaded with rock and soil for ballast. The crew-operated loader will dig trenches in an appropriate location large enough to contain the two modules. In one scenario, the modules are equipped with a wheeled cradle on which they are lowered to the ground and towed into the trench. First one module is moved, buried, and all operations verified, while the crew lives in the other module. Then the crew moves to the newly buried module. The second module is positioned and connected with the first.

Systems utilizing inflatable living quarters or other techniques may be superior. Whatever technique is employed, the living quarters are likely to be buried under 2 m of soil for protection from powerful solar flares and cosmic rays. The safe haven may then be maintained in an isolated location in case of catastrophic destruction of the two connected living modules.

Consumables

Advanced lunar operations will become dependent on highly regenerative life support systems, especially for carbon, hydrogen, and nitrogen (oxygen will be available from processing rock). To keep operation simple and development risk as low as possible for this first base, however, only water is recycled, and no "living off the land" is assumed, even in the event that ices are discovered in nearby permanently shadowed craters (*Watson et al.,* 1961; *Staehle,* 1983). Water recycling has been demonstrated in numerous live-in ground tests, while carbon dioxide and trace odor removal may be performed as aboard Skylab. A portion of the water allocation may be brought in food to permit a more palatable menu. Lunar gravity will permit more conventional food preparation and presentation than on past space missions. Pilot production of resources may be attempted in the course of technology experiments carried out by the crew. Operationally useful gases or liquids could be stored in empty autolander tanks for later use (Table 3).

TABLE 3. Consumable mass allocations.

Category	Mass (kg)	Remarks
Oxygen	2975	7.0 × body mass/year
Food	1910	4.5 × body mass/year
Water	955	9.0 × body mass/year (75% recycling)
Airlock and leakage	2360	includes 2 airlock cycles/day
Total	8200	

Assumes 85 kg body mass and safety factor of 0.2. These figures could be substantially improved by recycling of oxygen and pumping out an airlock during egress for EVA.

Equipment

A small variety of equipment is possible within the autolander mass allocation. If available, an additional autolander could be devoted entirely to added equipment, permitting much more in the way of scientific investigations and pilot production (Table 4).

Rovers and Space Suits

Two-person rovers are allocated in the mass budget at 300 kg each, compared with 210 kg for the Boeing/Apollo lunar roving vehicle (LRV). The LRV had a total of 1 hp with a top speed of 14 km/hr. A similar configuration is anticipated, somewhat ruggedized for longer use and with rechargeable batteries. A range of about 120 km would be desirable, but may be impractical from a mass, speed, and portable life support system (PLSS) endurance standpoint. If achievable, a 120-km range would allow reconnaissance of the entire floor of Peary Crater, and possible emplacement of a communications relay on the far rim of Peary. Suit/PLSS combinations are allocated 85 kg each, about the same as Apollo. However, the Apollo suits would be quite inadequate for the heavy usage expected at the lunar polar base. Three complete suits with PLSS are allocated per crewmember. In addition, each crewmember lands in a flight suit capable of surface EVA. Two surface suits would presumably be alternated for each crewmember, with the third held in reserve. Considerable development and heavy testing on the ground will be required to qualify suits for the base. Current work on zero prebreathe hardsuits suggests likely configurations (*AW&ST,* 1988).

MISSION PROFILE

At least one year before the crew is to arrive, two or three automated precursor missions are launched to survey the base site. If the LGO (*Phillips et al.,* 1986) has already flown, its imaging data will have allowed selection of a site having close to continuous illumination to within about 100-200 m. If the LGO carries an infrared thermal radiometer, a polar-temperature map with illumination from different longitudes will be available, but is not essential. If necessary, a Ranger-style imaging impact probe (IIP) will be targeted to the center of the selected base area to return imagery with resolution substantially better than 1 m. To obtain a 500 × 500-m image with 1-m resolution and 256 grey levels requires 2.25 million bits. Assuming a 2-km/sec impact velocity, if the last image is taken at a 4-km altitude and transmitted while the spacecraft is at least 2 km above the surface, a data rate of 1.1 Mbps is required, which is supportable over the Deep Space Net at lunar distance.

If the LGO has not flown, a somewhat simpler lunar imaging polar orbiter (LIPO) may be required to perform a one-month-long polar lighting survey over one full lunar day when the sun is at its minimum elevation angle of about 1.5° below the ideal horizon as seen from the north pole. If flown, LIPO would follow the original LGO mission profile carrying only the 11-kg Geodesic Imager or a similar instrument and associated support subsystems. We do not think that existing lunar orbiter imagery is adequate to select a north polar base site. Existing data could be adequate only if these images prove sufficient to target the IIP and to assure that a properly located solar panel tower of a given height will be in shadow for no more than about ten hours per lunar day.

The IIP carries with it four hard-landing navigation beacons carrying plastic sheet impact markers and strobe lights synchronized to the IIP cameras. These heavily cushioned navigational impactors (NIs) are patterned after similar devices on the first Rangers, which carried a seismometer in a balsa wood sphere designed (and successfully tested on Earth) to survive impact and transmit results for an extended period after impact. (The Rangers carrying these instrumented impactors failed for reasons unrelated to the instrument packages.) About 20 min before impact, NIs are ejected ahead of the IIP with about 10-m/sec along-track and 0.25-m/sec crosstrack velocity radially away from the IIP path. This assures their impact points will be visible within several images of the IIP. Upon impact they begin transmitting navigation test signals that are measured during the last seconds of the IIP flight. The results are relayed to Earth. The NIs then shift to a low-power standby mode, to be activated by an incoming autolander. Two or three operating beacons are required to guide the autolanders.

A rudimentary rover having a range of perhaps 10 km could supplement or replace the NIs. The rover would be landed within the target area to perform a detailed imaging survey and place several navigation beacons at locations correlated with the IIP imagery. While it would be helpful, such a rover is not essential. Two redundant IIP/NI missions are the recommended alternative, being much less costly than a rover.

TABLE 4. Equipment mass allocations.

Category	Mass (kg)	Remarks
Science Experiments	800	Selection to be made
Technology Experiments	800	Oriented towards expanded base, self-sufficiency
Loader	500	Excavation and towing
Miscellaneous Tools	400	Repairs, manual excavation
Film	100	
Communication Repeaters	100	Ridgetop relay to Earth
Total	2700	

Not including items integral to habitation modules.

Within a year after the IIP precursor, a communications relay orbiter (RO) is placed in polar orbit to verify the navigation beacons and monitor the autolander landings and the integrity of their cargo. Shortly afterward the first of four cargo missions is launched on a heavy lift vehicle (HLV), with subsequent loads landed at three-month intervals. Approximately 30,000 kg of base equipment, shelters, consumables, etc. are landed in four autolanders prior to the crew's arrival. Each autolander is guided to within 50-100 m of its chosen landing site using its onboard control system and the navigation beacons. In this manner they may be dispersed in the base area to eliminate interference, especially in the event of a failure.

When all four cargo loads are down and their integrity verified using onboard sensors, the base crew of five is launched aboard the shuttle or other crew transport into a low Earth parking orbit. Then an automated HLV launches the crew's Earth departure stage (EDS) with its payload of the command service module (CSM) and crew lander (CL) into a nearby orbit. The shuttle executes a final rendezvous and docking. The crew transfers to the command module and is ready to deport. No other mission-specific equipment need be carried by the shuttle. The crew's EDS superficially resembles the larger Apollo Saturn V S-IVB stage with the CL carried in a shroud beneath the CSM. Translunar injection (TLI) begins when the EDS cryogenic engines are ignited. After shutdown, the CSM turns around and docks with the CL. Finally, the docked CSM/CL is separated from the EDS to begin four days' transit to the Moon.

Targeted for 185-km altitude, the CSM executes its lunar orbit insertion (LOI) maneuver, placing it in polar orbit with a 2-hr period. Final observations verify the landing site, after which the CSM is put into a semidormant mode under ground control. The entire crew boards the CL, separates from the CSM, and fires the descent stage (CL-DS) engine to enter a 185 × 15-km orbit, with pericynthion about 500 km up-range from the landing site. From here, the landing essentially follows the Apollo profile (*MET,* 1971; *Staehle,* 1980), but with the navigation beacons assisting the pilots to a precision landing. With its landing close to Surveyor 3, Apollo 12 demonstrated adequate landing accuracy without navigation beacons but with ground tracking.

Following landing and a brief system check with the opportunity to abort to orbit, the crew immediately transfers to one of the autolanders with the storm shelter, habitation module, and solar panel packages on board. Power collectors are set up as the next order of business, followed by other base activities described later. The relay orbiter can provide a periodic communications link with Earth when Earth is below the horizon and with rovers or crewmembers some distance from the base site.

For the return trip, the crew reboards the CL and flies to orbit in the ascent stage (CL-AS), executes a rendezvous and docks with the CSM, transfers to the CM, and discards the CL. The CSM fires its propulsion system in the trans-Earth injection (TEI) maneuver, and carries the crew back to Earth. A few hours before entering the atmosphere the CM separates from the SM and orients for entry, and then enters, splashes down, and waits for the recovery ship just as with Apollo. Aerobraking and rendezvous with a shuttle can be used as well. All the base equipment and remaining consumables are left intact on the Moon, perhaps already in use by relief crews following the first to live and work over a year on the Moon.

Figure 3 shows the lunar mission profile and maneuver velocity roadmap. A summary of the spacecraft and vehicles to be used in the lunar mission is given in Table 5.

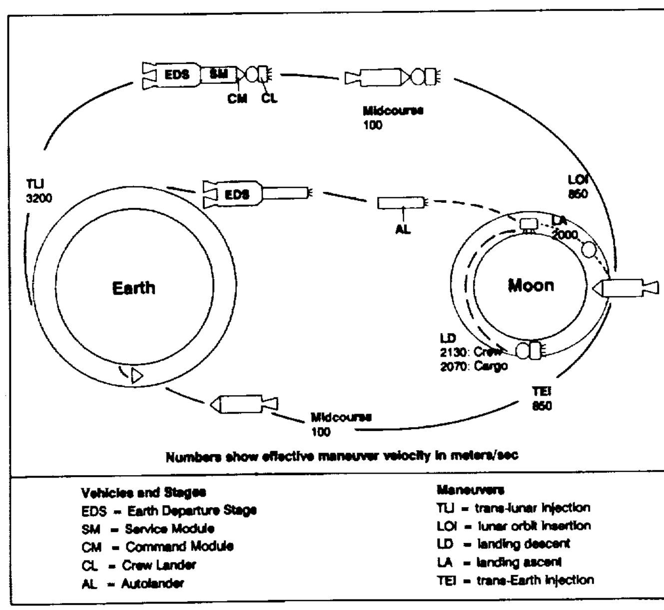

Fig. 3. Lunar mission profile and maneuver velocity roadmap.

TABLE 5. Spacecraft and vehicle summary.

# Req.	Vehicle or Spacecraft	Initial Mass (kg)	Final Mass Payload (kg)	Prop. Mass (kg)	Isp (sec)
1	Lunar Geoscience Orbiter	TBD	100-130	TBD	TBD
1	Lunar Imaging Polar Orbiter	TBD	10-20	TBD	TBD
2	Imaging Impactor Probe	TBD	100-150	TBD	TBD
1	Relay Orbiter	TBD	50-200	TBD	TBD
5	Heavy Lift Vehicle	TBD	78,000	TBD	TBD
5	Earth Departure Stage	74,000	29,000	40,000	444
1	Command Service Module	18,000	7,000	9,000	342
1	Crew Lander Descent Stage	9,900	3,300	5,200	304
1	Crew Lander Ascent Stage	3,100	1,500	1,400	342
4	Autolander	28,000	7,500	18,000	322

The number of each type of vehicle is shown for the mission description, followed by initial mass and full propellant loading. Final mass is shown excluding tankage, plumbing, and engines. Propellant mass and specific impulse are shown in the last two columns.

VEHICLE DESCRIPTION

Heavy Lift Vehicle (HLV)

All mission vehicles and base equipment are launched from Earth by an HLV. The reference mission described above uses five HLVs: four to launch the autolanders and their cargo, and one to launch the CSM/CL combination. All HLV launches are without a crew; the crew is brought up by a shuttle. The HLV payload requirement is dependent on the specific impulse of the EDS as follows (assumes HLV/EDS 5% adapter mass):

EDS I_{sp}	HLV Capability to LEO	
sec	kg	lb
420	83,000	183,000
444*	78,000	172,000
460	75,000	166,000
480	72,000	160,000

*Design point.

Each of the five lunar base flights constitutes a single launch package integrated and tested on the ground. Payload capabilities are within those contemplated for different versions of the USAF/NASA Advanced Launch System, NASA Shuttle C, Boeing/Hughes Jarvis, and of course the Saturn V and Energia. While the lunar base could be established using smaller expendables and the shuttle, this would result in a very different mission plan entailing more complex ground integration and testing and probably orbital assembly. In this case, the space station might be the logical assembly site, but the required assembly operations could interfere with laboratory and observational science contemplated for the station.

Earth Departure Stage (EDS)

As with the Saturn S-IVB stage and the Centaur, cryogenic liquid hydrogen and oxygen are the propellants of choice because of the substantial mass saving compared with storable propellants. The EDS was sized by the requirement to perform the 3200-m/sec TLI with the 27,900-kg CSM/CL combination as payload. The same EDS configuration also performs TLI for the autolanders. Adapter masses of 5% of dry weight between stages are used throughout.

Command Service Module (CSM)

Several configurations of vehicles for carrying the crew to and from the Moon were considered. The lunar orbit rendezvous technique that drove the Apollo configuration still appears the best within the guidelines of simplicity. Though a little cramped, the Apollo CM was capable of carrying five people and was configured for the eventuality of a Skylab rescue in 1974. Electrical and electronic subsystems would be totally redone today with a substantial mass saving, but the configuration, mechanical construction, attitude control, and recovery subsystems could remain essentially unchanged with a mass of 5400 kg or less upon entry. For all except the last few hours of its mission, the CM depends on the SM for utilities, consumables, and stabilization.

Fuel cells would be replaced by solar panels and nickel-hydrogen batteries for electrical power. The service propulsion system engine could be retained or replaced by a cluster of four Rocketdyne XLR-132 engines for redundancy and somewhat better performance (the latter is assumed). About nine days of consumables (plus reserves) are required to support the five-member crew during the time they are aboard the CSM. A smaller SM than used for Apollo is contemplated, with a dry mass of 2700 kg carrying 9400 kg of propellant. Provisions are made for storing the CSM in lunar orbit under ground control. Occasional maneuvers are required to maintain the orbit and be ready for an early return of the crew.

Crew Lander (CL)

Unlike the Apollo LM, the CL is not required to support the crew on the lunar surface or to carry supplies. Before undocking from the CSM, the surface navigation beacons will have been activated and tested, and the integrity of the autolander payloads (including all base equipment and provisions) will have been verified. The CL can be set down within easy walking distance of the autolanders. Consequently, the CL will be almost an "open cockpit" vehicle, which the crew will board wearing their spacesuits and PLSS. The unpressurized CL will afford thermal and sunlight protection, and possibly an oxygen and coolant supply. It is used only to carry the crew between the orbiting CSM and the lunar base. (Note that the Apollo astronauts were fully suited during landing, and that the LM could complete a landing in the event of a pressure failure.)

A single-stage CL would weigh 19,000 kg to be able to land, relaunch immediately in the event of an emergency, and return to the orbiting CSM carrying its five crewmembers with a mass of 200 kg each (including suits, PLSS, and other crew-specific equipment). Instead, the CL consists of the ascent and descent stages (AS and DS), which permits an abort at any point in the descent in the event of a descent propulsion failure. There is also a savings of 9000 kg from the single-stage version. The dry AS is 1700 kg with crew and burns 1400 kg of nitrogen tetraoxide and monomethyl hydrazine in its single XLR-132 engine. The fully loaded AS, of course, forms the payload of the DS, which is powered by a single lunar module descent engine (LMDE) burning nitrogen tetraoxide and Aerozine 50 (50% N_2H_4 and 50% unsymmetrical dimethyl hydrazine). All CSM and CL engines are hypergolic. The XLR-132 is pump fed, while the LMDE is pressure-fed (*Rockwell*, 1984; *Elverum et al.*, 1967). The dry DS is 1500 kg, with 5200 kg of propellant.

Autolander (AL)

All cargo is carried to the base site by the 28,000-kg AL, which is sized for the EDS's translunar injection capability. Total landed dry mass is 10,100 kg, of which 7500 kg is base payload. Clever use can presumably be made of much of the nonpayload mass, such as propellant tankage, batteries, and other equipment, but this is not yet factored into base design. There will be at least 500 kg of unburned propellant that could be scavenged from each AL. The tanks should be vented some time after landing to prevent corrosion and possible rupture or explosion as has happened on discarded Δ stages left in Earth orbit.

To attain the required thrust:weight ratio, throttleability, and improved performance, each AL employs two XLR-132 engines for midcourse, lunar-orbit insertion, and descent orbit insertion maneuvers. For powered descent, these engines are accompanied by a single lunar module descent engine (LMDE). Slightly under half of total maneuver velocity is executed with the higher specific impulse XLR-132s, with the rest by the LMDE. Putting the LMDE back in production may or may not be practical, but the tooling

is said to be stored by TRW, the engine manufacturer (D. Lee and W. Reynolds, personal communication, 1988). There could be considerable difficulty obtaining critical valves from subcontractors, some of whom may not be in business. Nearly all the original engineering and shop talent is dispersed, so either redevelopment or a derivative from nonthrottleable engines now in production might be more practical. Similar uncertainty surrounds possible production of other Apollo-derived equipment described here, such as the CM.

In keeping with the design philosophy, storable propellants and Apollo inheritance were chosen for the ALs. If a modest departure were taken from this philosophy, a substantial performance improvement could be realized, reducing the number of ALs from four to three, and the number of EDSs and HLVs from five to four. One uprated Pratt & Whitney RL-10 engine has the thrust required for the AL. Using LOX/LH_2, its performance is significantly higher than that of storable propellants. The engine can be operated up to 89 kN (20,000 lb) thrust, and has been throttled over a 10:1 range (J. Brown, personal communication, 1988). With a better insulated hydrogen tank, the fuel could be kept cold for the four days required from launch through landing, giving the AL a 10,000-kg payload, or one-third greater than the selected AL concept with storable propellants.

A more ambitious concept proposed in the 1984 Johnson Space Center Lunar Surface Return Study calls for expendable landers capable of placing 38,600 lb of useful payload on the lunar surface. Only two such landings would be sufficient to deliver all polar base elements discussed, including the crew and AS (*Roberts et al.,* 1984).

CREW SELECTION AND BASE ACTIVITIES

The single overriding objective of the proposed polar base is to successfully live on the Moon for a year. Productivity is of secondary importance, because it is felt that the harsh environment will tax the crew and system resources. During their stay, the crew can extensively explore the local area and perform limited scientific and technological experiments. Given appropriate training and interests, the crew can remain "productively entertained" for a year near the pole. By analogy, an accountant might become seriously depressed if confined to a desolate region of mountainous terrain on Earth for a year, while a geologist might go to great lengths to secure the opportunity to explore and map virgin territory. Much of the crew's waking time will be spent in subsistence activities such as maintaining equipment, preparing meals, housekeeping, exercising, etc. More interesting activities such as base construction and training for the return flight will occupy more time. What time remains will be available for exploration, setting up and performing experiments, and "just taking it all in."

For such a long stay, the crew must have the sort of autonomy enjoyed by terrestrial explorers. Priorities set on Earth with endless timelines and requests for "one more sample" will not succeed. Crewmembers will have their own desires, objectives, and pet projects, and must also function as a cohesive team for each other's survival. Mission control in the traditional space mission sense will not work. Instead, a very few engineers, technicians, and personal assistants on the ground will need to function at the service of the crew.

The most successful terrestrial expeditions have had a single leader respected by a crew of his or her own choosing. Morale is exceedingly important and has been a critical factor on both polar and space missions of long duration. In this regard many polar expedition leaders feel a good cook is the most important crewmember! Two pilots thoroughly familiar with the vehicles will be required to fly the CSM and CL. A physician/dentist is almost a necessity, while the other two crewmembers might be accomplished in the science and technology related to the base. All crewmembers should probably have specific system responsibilities, though this is a matter of organization best left to the leader.

CONCLUSIONS

In summary, the described lunar polar expedition ranks among the more modest lunar base missions currently being discussed in terms of transportation requirements, life support, and power system design. Yet it yields major advances in our knowledge of how to live and work on the Moon. Whether this is a first mission or an adjunct to a major national or international program, it offers a threshold achievement by visiting a uniquely interesting region of the Moon and testing long-term habitation techniques. It can create a nucleus of expanding human presence beyond Earth.

A successfully established polar camp can be augmented dramatically with each additional landing, exploring new capabilities in selenogical research, astronomy, industry, biomedicine, and life support. Once operational, a solar power system can be enlarged and extended to serve other nearby regions. It could serve as the starting point for a comprehensive power distribution grid reaching southward toward the equator. Indeed, initial base expansion can contribute to virtually every aspect of lunar development: surface transport systems; closed-loop life support systems and food production; production of oxygen, ceramics, and metals; bioscience; astronomy; and architectural systems.

Earth's Moon has been called a "seventh continent," a sister planet, a "stepping stone to the stars." It is all of these. Learning to live and work productively on the Moon, our nearest celestial neighbor, will provide invaluable knowledge, resources, and experience for bolder missions into deep space.

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REFERENCES

Allen K. R. et al. (1985) *Antarctica.* Reader's Digest Services, Surrey Hills, NSW. 319 pp.

AW&ST (1988) NASA Explores New Space Suit Concepts. *Aviation Week and Space Technology,* Jan. 11, p. 36.

Elverum G. W. et al. (1967) The descent engine for the lunar module. In *Papers Presented to the AIAA 3rd Joint Propulsion Specialist Conference,* Paper #67-51. Washington, DC. 15 pp.

Hall B. J. (1981) Shaded relief map of the lunar polar regions. *U.S. Geol. Surv. Map I-1326-A/B.*

Hansen T. P. (1970) *A Guide to Lunar Orbiter Photographs.* NASA SP-242. 125 pp.

Huntford R. (1984) *Scott and Amundson: The Race to the South Pole.* Atheneum, New York. 565 pp.

Kirwin L. P. (1960) *A History of Polar Exploration.* Norton, New York. 374 pp.

Kosofsky L. J. and El-Baz F. (1970) *The Moon as Viewed by Lunar Orbiter.* NASA SP-200. 152 pp.

Kuiper G. P. et al. (1967) *Consolidated Lunar Atlas—Supplements 3 and 4 of the USAF Photographic Lunar Atlas.* Lunar and Planetary Laboratory, Univ. of Arizona, Tucson. 24 pp.

MET (Mission Evaluation Team) (1971) *Apollo 11 Mission Report*, pp. 19-20, 29-52. Mission Evaluation Team, NASA SP-238.

Phillips R. et al. (1986) *Contributions of a Lunar Geoscience Observer Mission to Fundamental Questions in Lunar Science.* Southern Methodist Univ., Dallas. 86 pp.

Roberts B. et al. (1984) *Lunar Surface Return.* NASA Johnson Space Center Publ. No. 84-0761. 185 pp.

Rockwell (1984) *XLR-132*, pp. 1-2. Publication 574-F-27 Rev. 10-84. Rockwell International, Rocketdyne Division, Canoga Park, California.

Staehle R. L. (1983) Finding "paydirt" on the Moon and asteroids. *Astronaut. Aeronaut.*, Nov., 44-49.

Watson K., Murray B. C., and Brown H. (1961) On the possible presence of ice on the Moon. *J. Geophys Res., 66,* 1598.

ASTRONOMY ON THE MOON: GEOLOGICAL CONSIDERATIONS

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The Moon is an excellent site for astronomical observations. This paper descibes two geological aspects related to astronomy from the Moon. First it evaluates the sources of gases near a lunar base as input to calculations reported in a separate paper on the growth of an artificial lunar atmosphere. The results suggest that mining for ^{3}He could produce the most gas (1 kg/sec), but rocket exhaust (0.1 kg/sec) and habitat venting (0.5 kg/sec) are also important. Second, the paper discusses criteria that need to be considered when determining the site of a lunar astronomical facility. These are longitude and latitude (equatorial sites are favored), topography (important to be relatively flat for ease of installation), distance from a lunar base (to be free of seismic noise, dust, and gases), the site's value to lunar geoscience (other factors being equal, a geologically diverse site is better), and its value as a materials resource (mining and observatories are incompatible).

INTRODUCTION

The Moon may be the best place in the inner solar system from which to make astronomical observations (*Burns,* 1988). As part of our University of New Mexico and BDM Corporation study of concepts for astronomical observatories on the Moon, we have investigated several geological aspects of the problem. This paper describes our initial results, in two broad categories. First, it evaluates quantitatively the sources of gases near a lunar base, which provides essential information for our calculations concerning the evolution of an artificial lunar atmosphere (*Burns et al.,* 1991; *Fernini et al.,* 1990). Second, the paper develops criteria for the selection of sites for lunar astronomical facilities.

SOURCES OF ATMOSPHERIC GASES

There will be both natural and artificial sources of gases on the Moon. Although the present tenuous nature of the lunar atmosphere indicates that natural sources are too low to allow a significant atmosphere to develop, I list them here both for completeness and for comparison with artificial sources. As a rule of thumb, if an artificial source is of the same order as a natural one, it will not lead to a significantly enhanced lunar atmosphere. Results of the evaluations appear in Table 1 and are discussed below.

Solar Wind

The solar wind is a major source of gas to the lunar atmosphere. Most of the gases trapped in the regolith are derived from solar wind implantation. Using fluxes given by *Vondrak* (1974), the total amount of solar wind input to the lunar atmosphere is 5×10^{-2} kg/sec, almost all of which is H (40 g/sec) and He (8 g/sec). These gases are delivered uniformly to the sunlit part of the Moon.

TABLE 1. Sources of gas near a lunar base.

Source	Rate (kg/sec)
Solar wind	5×10^{-2}
Meteoritic volatilization	2×10^{-3}
Internal degassing	$<3 \times 10^{-4}$
Rocket exhaust	10^{-1}
Habitat venting	0.5
Mining and Manufacturing	1
^{3}He mining	1
Oxygen production	10^{-3}
Glass production	10^{-5}

Meteorite and Comet Volatilization

Many of the micrometeoroids that hit the Moon are rich in volatile substances. Because meteoroids are vaporized when they impact the lunar surface, their volatiles are released to the atmosphere. *Gault et al.* (1972) estimate a flux of 2×10^{-3} g/cm^2/10^6 yr for masses smaller than 1 g. If these contain on average 10% H_2O (appropriate for CI carbonaceous chondrites, though possibly low for comets), then meteoroids contribute 2×10^{-3} kg/sec to the atmosphere. Using a flux estimated by *Hartmann* (1980) reduces this estimate by a factor of 2. Like the solar wind, this is distributed globally. Of course, a single, large impact of a comet or hydrated meteorite could inject a considerable amount of volatiles near the point of impact. For example, impact of a body 20 m across containing 10% H_2O would release 10^6 kg of water vapor instantaneously. Fortunately, such events are rare, happening once every 10^4 yr.

Internal Degassing

The Moon continuously outgasses, as shown by the release of radon (e.g., *Gorenstein et al.,* 1973). Reports of lunar transient phenomena (e.g., *Middlehurst,* 1967) suggest occasional large releases. However, *Vondrak* (1977) argues that such releases

would need to be $>10^4$ kg to be detected by the SIDE (Superthermal Ion Detector Experiment) carried by Apollo 12, 14, and 15. Since no such release was observed during its eight years of operation, we can place an upper limit of 10^4 kg/yr, or $<3 \times 10^{-4}$ kg/sec.

Rocket Exhaust

Vondrak (1974) sounded the first alarm about the possibility that the Moon's fragile atmosphere could be modified significantly by rocket exhaust, habitat venting, and mining, pointing out that each Apollo mission temporarily doubled the lunar atmosphere. The amount of gas released by rocket exhaust is difficult to estimate, as one must assume spacecraft capabilities and frequency of flights. If we assume each landing or ascent uses 20 times the Apollo lunar module capacity (3000 kg; *Johnson,* 1971) and that there are 18 trips per year, then on average only 0.1 kg/sec will be released into the lunar environment.

Habitat Venting

This is also difficult to guess, principally because we must assume some critical size for the lunar base. Structural leakage accounts for 0.2 mg/m^2-sec (*Vondrak,* 1991), about the same as expected for habitats on Mars (*Duke and Keaton,* 1986). Assuming each habitat has an area of 239 m^2 (cylinders 14.3 m long $\times$ 4.6 m diameter; i.e., the nominal size of space station modules), then each would release 4.8×10^{-2} kg/sec. If there are 10 such habitats, then they would release 4.8×10^{-1} kg/sec. Air lock venting would also allow gases to escape. Assuming 0.6 kg/use (*Duke and Keaton,* 1986) and 10 uses per day, on average 7×10^{-5} kg/sec would escape to the lunar atmosphere, insignificant compared to leakage.

Mining

Several lunar resources have been identified as promising. I consider three of them here: ^{3}He for use in nuclear fusion reactors (*Wittenberg et al.,* 1986), oxygen production from ilmenite ($FeTiO_3$) for use a propellant (e.g., *Gibson and Knudsen,* 1985), and glass production from lunar regolith (*Blacic,* 1985). Extraction of each of these releases gases bound in the regolith and thus poses a threat to the lunar atmosphere.

Maximum concentrations of gases in the regolith are given in Table 2. Concentrations of CH_4, CO, and CO_2 were calculated from gas-release curves measured by *Gibson and Johnson* (1971) and from data reported by *DesMarais et al.* (1973). Although sulfur is an important constituent of the lunar regolith (about 0.2 wt%; *Gibson and Moore,* 1974), I have not included it because it is not released as readily as are the other gases, though the precise kinetics of sulfur release ought to be worked out.

TABLE 2. Maximum concentrations (ppm) of gases in lunar regolith.

Molecule	Concentration	Reference
H_2	100	1,2
He	70	3
Ne	5.4	6
C	280	4,5,6
CH_4	8	
CO	232	
CO_2	39	
N_2	80	6,7

References: (1) *DesMarais et al.* (1974); (2) *Gibson et al.* (1987); (3) *Walton et al.* (1973); (4) *Moore* (1974); (5) *Pillinger* (1979); (6) *Gibson* (1974); (7) *Mueller* (1979).

Helium mining. Mining for ^{3}He is by far the worst case because the low abundance of ^{3}He necessitates mining and processing huge quantities of regolith. In calculating the amount of gases that will be released, I assume that regolith containing the maximum amount of gas will be mined and that we will need to produce 20 metric tons (20,000 kg) of ^{3}He per year, which would supply U.S. energy needs for one year (*Wittenberg et al.,* 1986). This requires mining 8.3×10^8 metric tons of regolith per year. I assume mining takes place continuously and that 10% of the gas is lost to the atmosphere; losses might be less than 10%, but *Carrier et al.* (1973) report that several percent of trapped gases are lost merely by moving regolith, so to calculate the maximum gas released, I use 10%. Consider two cases: (1) heating the regolith to 700°C, which will release >90% of the He (*Pepin et al.,* 1970) and (2) heating to 1200°C, which might enhance extraction of other species. The second case also allows us to place a better defined upper limit of gas release. Results are shown in Table 3.

TABLE 3. Gas loss (kg/sec) during ^{3}He mining.

	700°C	1200°C
H_2	0.13	0.13
H_2O	0.06	0.06
He	0.18	0.18
Ne	0.01	0.01
CO_2	0.05	0.1
CO	0.03	0.6
N_2	0.01	0.21
Total	0.47	1.29

For case 1, all the H is released, 10% of it as H_2O, all the He, 60% of the Ne, 50% of the total CO_2, and about 5% of the total CO and N_2. This results in a total of about 0.5 kg/sec of gases released. For case 2, all the H (10% of it as H_2O), He, and Ne, 90% of the CO_2, and 80% of the CO and N_2 are released. This causes the emission of 1.3 kg/sec. It appears that He mining would release about 1 kg/sec into the lunar atmosphere, much greater than the other sources listed in Table 1.

Oxygen production. To produce 1000 metric tons per year from regolith containing 5% ilmenite at 5% efficiency, 2×10^6 metric tons of regolith must be mined. Although oxygen extraction will take place at 1000°C (e.g., *Gibson and Knudsen,* 1985), I use the gas release values for 1200°C to place a firmer upper limit on the amount of gas released. The amount of gas will be proportional to the amount of regolith mined, and since oxygen requires mining about 10^{-3} as much regolith, only about 10^{-3} kg/sec will be released, much less than for ^{3}He mining. If an enormous commercial market develops for lunar oxygen and requires 10^3 times more, then the amount of gas released would equal that from ^{3}He.

Glass production. If a lunar base could use 1000 metric tons per year of glass of feldspar composition, and the mined soil is in the highlands (average 70% plagioclase feldspar), then only 1.4×10^3 metric tons of regolith need be processed. In this case the regolith would be melted, releasing all the gas in it. As above, I assume 10% of that gas will escape. Using the gas contents in Table 2, this results in release of 10^{-5} kg/sec, clearly not a threat to the integrity of the lunar atmosphere. Even if a million tons of glass were needed, only about 10^{-2} kg/sec would be released.

In summary, it appears that rocket exhaust, habitat venting, and ^{3}He mining have the most potential for disturbing the tenuous lunar atmosphere (Table 1). These calculations will need to be improved as plans for lunar base development become clearer and as mining and processing equipment is designed.

SITE SELECTION CRITERIA

We have identified several criteria that must be considered when determining where to locate an astronomical observatory on the Moon: longitude and latitude, topography, distance from a lunar base, value of the site to lunar geoscience, and value as a materials resource.

Longitude and Latitude

The high background of very low frequency (VLF; <30 MHz) radio waves emanating from Earth requires that a VLF array be located on the lunar farside. Because of librations of the Moon, only sites with longitudes >98° (east and west) are permanently shielded from Earth. However, because of growing radio-frequency interference on Earth, it is in general desirable to place all radio telescopes on the farside. An exception is the Moon-Earth radio interferometer (MERI), which employs one or more radio antennas on Earth, hence must tolerate radio interference. On the other hand, even this would benefit by location on the farside because it might afford a way to distinguish interference from the signals of interest.

To view the entire sky, telescopes must be deployed over a wide range of latitudes. However, a complex VLF (or optical) array is almost certain to be a unique facility, so an optimum latitude must be chosen. Because objects of interest occur in both northern and southern skies, it seems sensible to locate a VLF array within 20° of the lunar equator. Also, polar sites are weak for viewing the planets in our solar system as all the objects of interest would be at the horizon.

Topography

The Moon's surface is divided into two distinct terrains, the highlands and the maria. The highlands compose the oldest lunar crust and are densely cratered. Relief differences are large over relatively short distances. For example, central peaks and walls of large craters can rise 3-4 km above their floors. Some large basins (craters >100 km across) have floors that are relatively smooth and light colored; the floor materials represent either volcanic flows different in composition from darker mare flows or are impact-generated, fluidized materials (which is the case of the smooth plain on which Apollo 16 landed). Because of the highlands' great age, they are covered with a thick regolith of impact-generated debris, hence tend to contain fewer large blocks of rocks. The maria are younger than the highlands and formed when lavas erupted onto the lunar surface and filled low-lying regions. Mare surfaces tend to be much smoother than highland surfaces and are much less cratered. However, they also have thinner regoliths, so crater ejecta blankets tend to contain numerous blocks of rocks.

Topography enters into the selection of a site for an observatory more for ease of deployment and operation of the facility than for scientific reasons. The rugged terrain in the highlands makes it difficult for elements of an array to communicate by line-of-sight with a single central processing station. Also, deployment vehicles would need to maneuver around many hills and valleys created by old, degraded craters. On the other hand, blocks of rocks would be less of a hazard than in the maria. Overall, the optimum site would be a relatively old (>3.5 b.y.) mare surface. The old age would permit a relatively thick, unblocky regolith and the presence of mare basalt flows would create relatively low relief across a large region.

Distance from a Lunar Base

An observatory needs to be isolated from an active lunar base, especially if the base is the site of extensive mining operations. Several factors must be taken into account when estimating how far an observatory needs to be located from a lunar base. These are the distance from a lunar base located on a limb (90° longitude), seismic noise, atmospheric contamination, and dust.

Distance from a limb site. It might be desirable to locate a lunar base close to a nearside limb. For example, the Mare Smythii region holds great promise for lunar geoscience investigations and for lunar resource extraction (*Spudis and Hood,* 1991). To keep Earth in view continuously (for both psychological and operational reasons), the base could be no farther than about 90°E. However, lunar librations cause sites up to 98° to sometimes have Earth in view. Consequently, a radio array would need to be at least 240 km east of a lunar base located at 90°E longitude (1° equals 30 km at the lunar equator). Furthermore, radio waves from Earth would be diffracted. Assuming a perfectly spherical Moon, the diffraction region for very low frequency radio waves (300-m wavelength) is 75 km (see, e.g., *Jackson,* 1975, p. 447). Thus, this distance must be added to that caused by librations: a radio telescope must be located at least 315 km from 90° longitude.

Seismic noise. Lunar base activities will increase the general seismic background on the Moon. This might affect radio telescope antennas, especially dishes, and would almost certainly affect an array of optical telescopes. Using data from the signal strengths generated by charges placed on the lunar surface by astronauts and from impacts of the Apollo 17 lunar module, *Cooper and Kovach* (1975) developed an empirical relation between ground motion and seismic energy, $A = kE^{0.5}/r$, where A is the amplitude (nm), E is the energy (ergs), and r is the distance (km). K is an empirical constant, 2×10^{-5}. To estimate the effect of lunar base activity, let us assume that surface mining takes place continuously, and calculate the ground motion (amplitude) generated by dropping 1 m^3 of soil from a height of 2 m. This generates about 6×10^{10} ergs, assuming soil density of 2×10^3 kg/m^3. This produces the following ground motions:

Distance (km)	Ground Motion (nm)
1	5
10	0.5
100	0.05

The lunar seismic background produces ground motions on the order of 1 nm, so it is clear from the above that even an optical-telescope array will not be affected if it is located more than 10 km from a mining operation. This analysis does include the additive effects of each mining scoop. This would seem to be important because seismic waves are not attenuated rapidly on the Moon; for example, a signal damped out in minutes on Earth lasts hours on the Moon (*Lammlein et al.,* 1974). The above

analysis also does not consider more potent sources of energy such as blasting operations. A detailed analysis of artificial lunar seismicity needs to be done. Nevertheless, it seems safe to conclude that artificial seismic disturbances will not affect radio observations on the Moon.

Artificial atmosphere. The Moon's tenuous atmosphere makes it ideal for astronomical observation. However, lunar base operations could lead to a significant increase in atmospheric density, as was first pointed out by *Vondrak* (1974). This problem has been addressed recently by *Burns et al.* (1991). Even considering the worst case, mining for ^{3}He (which might contribute as much as 1 kg/sec into the lunar atmosphere; Table 1), *Burns et al.* (1991) concluded that no significant growth of the atmosphere occurs beyond 10-100 km from a lunar base, roughly the range at which seismic pollution becomes negligible. However, if lunar base activities contributed $\geqslant$10 kg/sec, significant damage to the environment might occur.

Dust contamination. In principle, this could be a serious problem within 1-10 km of a lunar base because of dust accelerated by rocket landings and lift-offs. However, this could be mitigated by construction of landing pads, so we do not consider it to be a serious problem, but a quantitative analysis needs to be made. It is almost certainly of little concern for radio telescopes.

Value to Lunar Geoscience

The site for any astronomical observatory on the Moon ought to be chosen for its suitability for that purpose. Nevertheless, other factors, including operational considerations such as communications, being equal, it seems reasonable to propose choosing the site that has the greatest interest to lunar geoscientists. If this is done, any visit by a crew to repair or expand the facility could include geologic sampling as well. Even during deployment by automated vehicles, geophysical instruments could be deployed as well, although this might add to the cost and complexity of the deployment.

Value as a Materials Resource

Mining and astronomy are probably incompatible, so sites that hold obvious resource potential ought to be avoided. An alternative would be to designate areas for astronomy (and other sciences) within regions possessing resources, keeping in mind the criteria for distance from a lunar base.

Candidate Site for a VLF Array: Tsiolkovsky

The large crater Tsiolkovsky (Fig. 1) is an excellent candidate for the site of a VLF radio array. The crater is 180 km across, rim to rim, and its floor is 113 km across, providing ample space for even an advanced array. It is located on the lunar farside at 20°S latitude and 130°E longitude. Thus, Tsiolkovsky is in the equatorial zone and far from any base established on the nearside: Even a base on the eastern limb at 90°E would be 1200 km away.

The crater's floor is covered by high-Ti mare basalt (*Wilbur,* 1978) with an age similar to that of the Apollo 11 landing site, ~3.6 b.y. The floor is smooth, except where punctuated by craters. Based on its age, a thick regolith ought to be present, thereby lessening hazards from boulder fields near small craters. The central peak rises 3 km above the smooth plains. A central station located on the highest point could receive signals from

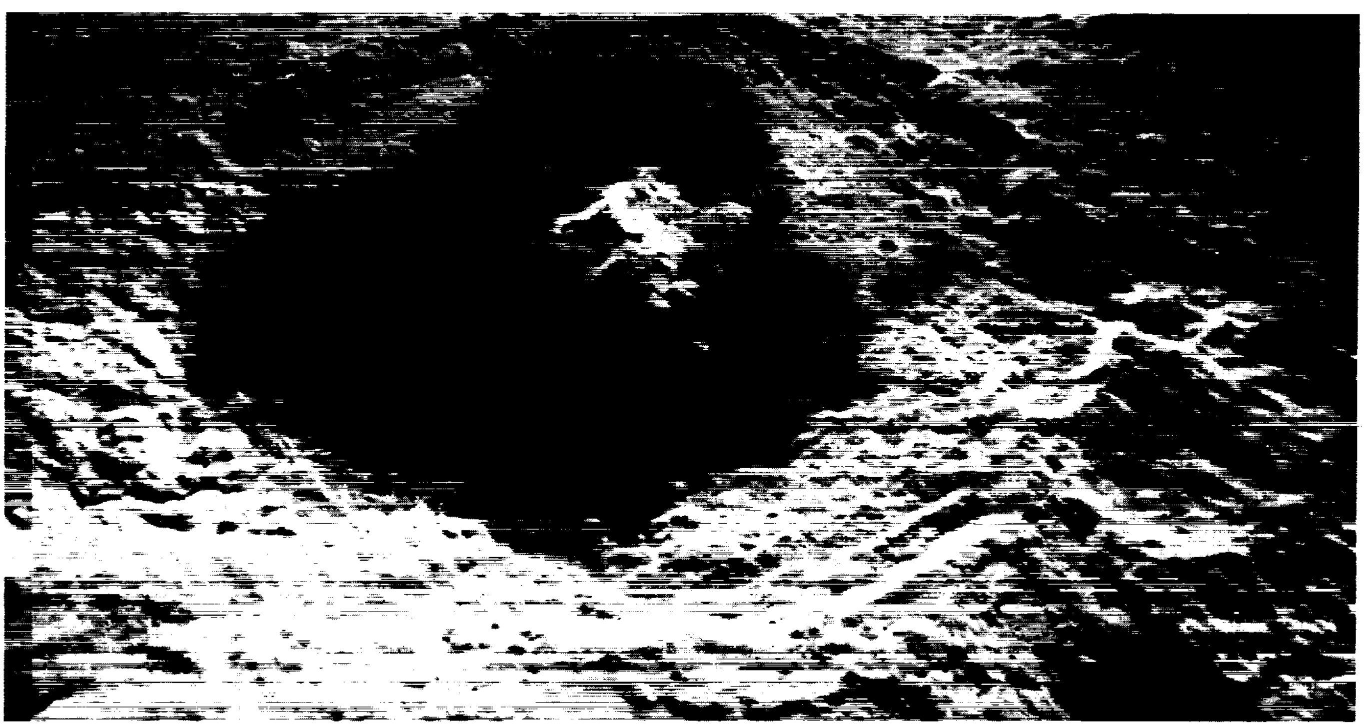

Fig. 1. The crater Tsiolkovsky, located at 20S 130E on the lunar farside, would be an excellent site for a very low frequency array. The crater is 180 km across and is floored by dark, smooth mare basalt flows.

anywhere on the floor; on a sphere with the Moon's radius, the horizon would be 102 km away when viewed from a mountaintop 3 km high.

Tsiolkovsky is also interesting geologically (*Guest and Murry,* 1969; *Guest,* 1971; *Wilbur,* 1978). It represents an opportunity to study a relatively well-preserved large crater. The central peak probably represents uplifted, deep-seated materials, an ideal place to study the field relations of highland crustal rocks. It also provides an opportunity to study eruption mechanisms and post-volcanic tectonic processes.

There are two drawbacks to Tsiolkovsky as the site for the VLF array, although neither is a fatal flaw. One is that the walls rise 4 km above the floor, thus limiting the view of the horizon to >6° above the horizontal (if the array is centered 40 km from the crater wall). The second problem is that the mare basalts that help make the floor smooth are of the high-Ti variety. This makes the regolith in the crater a potential source of ^{3}He, which is found in greater abundance in high-Ti materials. However, development of ^{3}He-based fusion reactors for commercial power production is far in the future and Tsiolkovsky represents only a few percent of the total amount of high-Ti basalt on the Moon, so it would not need to be exploited. Furthermore, although high-Ti basalts are the richest source of He, all lunar soils, mare and highland, contain He in extractable quantities. If Tsiolkovsky turns out to be the best site for the VLF array, it ought to be declared a scientific preserve, closed to resource exploitation.

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REFERENCES

Blacic J. D. (1985) Mechanical properties of lunar materials under anhydrous, hard vacuum conditions: Applications of lunar glass structural components. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 487-495. Lunar and Planetary Institute, Houston.

Burns J. O. (1988) Prolog. In *Future Astronomical Observatories on the Moon,* p. 1-4. NASA CP-2489.

Burns J. O., Fernini I., Sulkanen M., Duric N., and Taylor J. (1992) Artificially-generated atmosphere near a lunar base. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Carrier W. D. III, Bromwell L. G., and Martin R. T. (1973) Strength and compressibility of returned lunar soil. *Proc. Lunar Sci. Conf. 3rd,* pp. 3223-3234.

Cooper M. R. and Kovach R. L. (1975) Energy, frequency, and distance of moonquakes at the Apollo 17 site. *Proc. Lunar Sci. Conf. 6th,* pp. 2863-2879.

DesMarais D. J., Hayes J. M., and Meinschein W. G. (1973) The distribution in lunar soil of carbon released by pyrolysis. *Proc. Lunar Sci. Conf. 4th,* pp. 1543-1558.

DesMarais D. J., Hayes J. M., and Meinschein W. G. (1974) The distribution in lunar soil of hydrogen released by pyrolysis. *Proc. Lunar Sci. Conf. 5th,* pp. 1811-1822.

Duke M. B. and Keaton P. W., eds. (1986) *Manned Mars Missions.* NASA M001. 85 pp.

Fernini I., Burns J. O., Taylor G. J., Sulkanen M., Duric N., and Johnson S. (1990) Dispersal of gases generated near a lunar outpost. *J. Spacecr. Rockets, 27,* 527-538.

Gault D. E., Hörz F., and Hartung J. B. (1972) Effects of microcratering on the lunar surface. *Proc. Lunar Sci. Conf. 3rd,* pp. 2713-2734.

Gibson E. K. Jr. (1975) Distribution, movement, and evolution of the volatile elements in the lunar regolith. *Moon, 13,* 321-326.

Gibson E. K. Jr. and Johnson S. (1971) Thermal analysis—inorganic gas release studies of lunar samples. *Proc. Lunar Sci. Conf. 2nd,* pp. 1351-1366.

Gibson E. K. Jr. and Moore G. W. (1974) Sulfur abundances and distributions in the valley of Taurus-Littrow. *Proc. Lunar Sci. Conf. 5th,* pp. 1823-1837.

Gibson E. K. Jr., Bustin R., Skaugset A., Carr R. H., Wentworth S. J., and McKay D. S. (1987) Hydrogen distributions in lunar materials (abstract). In *Lunar and Planetary Science XVIII,* pp. 326-327. Lunar and Planetary Institute, Houston.

Gibson M. A. and Knudsen C. W. (1985) Lunar oxygen production from ilmenite. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 543-550. Lunar and Planetary Institute, Houston.

Gorenstein P., Golub L., and Bjorkholm P. J. (1973) Spatial features and temporal variability in the emission of radon from the Moon: An interpretation of results from the alpha particle spectrometer. *Proc. Lunar Sci. Conf. 4th,* pp. 2803-2809.

Guest J. E. (1971) Geology of the farside crater Tsiolkovsky. In *Geology and Physics of the Moon* (G. Fields, ed.), pp. 93-103. Elsevier, Amsterdam.

Guest J. E. and Murry J. B. (1969) Nature and origin of Tsiolkovsky Crater, lunar farside. *Planet. Space Sci., 17,* 121-141.

Hartmann W. K. (1980) Dropping stones in magma oceans: Effects of early lunar cratering. In *Proceedings of the Conference on the Lunar Highlands Crust* (J. J. Papike and R. B. Merrill, eds.), pp. 155-171. Pergamon, New York.

Jackson J. D. (1975) *Classical Electrodynamics.* Wiley, New York.

Johnson F. S. (1971) Lunar atmosphere. *Rev. Geophys., 9,* 813-823.

Lammlein D. R., Latham G. V., Dorman J., Nakamura Y., and Ewing M. (1974) Lunar seismicity, structure, and tectonics. *Rev. Geophys. Space Phys., 12,* 1-21.

Middlehurst B. M. (1967) An analysis of lunar events. *Rev. Geophys., 5,* 173-179.

Moore C. B. (1974) *A Check List of Total Carbon and Sulfur Analyses.* Center for Meteorite Studies Contrib. No. 94. 29 pp.

Mueller O. (1979) Solar wind nitrogen and indigenous nitrogen in lunar material. In *Origin and Distribution of the Elements* (L. Ahrens, ed.), pp. 47-59. Pergamon, New York.

Pepin R. O., Nyquist L. E., Phinney D., and Black D. C. (1970) Rare gases in Apollo 11 lunar material. *Proc. Apollo 11 Lunar Sci. Conf.,* pp. 1435-1454.

Pillinger C. T. (1979) Solar wind carbon chemistry as revealed by lunar sample analysis. In *Origin and Distribution of the Elements* (L. Ahrens, ed.), pp. 61-72. Pergamon, New York.

Spudis P. and Hood L. L. (1992) Geological and geophysical field investigations from a lunar base in Mare Smythii. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Vondrak R. (1974) Creation of an artificial lunar atmosphere. *Nature, 248,* 657.

Vondrak R. (1977) Upper limits to gas emission from lunar transient phenomena sites. *Phys. Earth Planet. Inter., 4,* 293-298.

Vondrak R. (1992) Lunar base activities and the lunar environment. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Walton J. R., Lakatos S., and Heymann D. (1973) Distribution of inert gases in fines from the Cayley-Descartes region. *Proc. Lunar Sci. Conf. 4th,* pp. 2079-2095.

Wilbur C. L. (1978) Volcano-tectonic history of Tsiolkovsky (abstract). In *Lunar and Planetary Science IX,* pp. 1253-1255. Lunar and Planetary Institute, Houston.

Wittenberg L. J., Santarius J. F., and Kulcinski G. L. (1986) Lunar source of ^{3}He for commercial fusion power. *Fusion Technol. 10,* 167.

HELIUM MINING ON THE MOON: SITE SELECTION AND EVALUATION

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The feasibility of recovering helium (He) from the Moon as a source of fusion energy on Earth is currently being studied at the University of Wisconsin. Part of this study is selection and evaluation of potential sites for lunar He mining. Selection and evaluation of potential mining sites are based on four salient findings by various investigators of lunar samples: (1) Regoliths from areas underlain by highland materials contain less than 20 wppm He; (2) Certain maria regoliths contain less than 20 wppm He, but others contain 25 to 49 wppm; (3) The He content of a mare regolith is a function of its composition; regoliths rich in Ti are relatively rich in He; and (4) He is concentrated in the <100-μm size fractions of regoliths. The first three findings suggest that maria are the most promising mining sites, specifically, those that have high-Ti regoliths. Information on the regional distribution and extent of high-Ti regoliths comes mainly from two sources: direct sampling by various Apollo and Luna missions, and remote sensing by gamma-ray spectroscopy and Earth-based measurements of lunar spectral reflectance. Sampling provides essential control on calibration and interpretation of data from remote sensing. These data indicate that Mare Tranquillitatis is the principal area of high-Ti regolith of the eastern nearside, but large areas of high-Ti regolith are indicated in the Imbrium and Procellarum regions. Recovery of significant amounts of He-3 will require mining billions of tonnes of regolith. Large individual areas suitable for mining must therefore be delineated. The concentration of He in the finer size fractions and considerations of ease of mining mean that mining areas must be as free as possible of sizable craters and blocks of rock. Pending additional lunar missions, information regarding these features must be obtained from lunar photographs, photogeologic maps, and radar surveys. The present study is decidedly preliminary; available information is much too limited to permit even a close approach to final evaluation. As a prelude to recovery of He from the Moon, systematic exploration and sampling of high-Ti maria regoliths should therefore have a high priority in future lunar missions.

INTRODUCTION

Part of the University of Wisconsin study of the feasibility of recovering He-3 from the Moon is the selection and evaluation of potential mining sites. First it is necessary to identify areas in which the regolith is enriched in He, preferably those containing 30 wppm or more. The occurrence of He in the regolith must then be examined, and consideration given to physical characteristics of terrain and regolith that could affect the feasibility of mining in the areas selected.

This paper summarizes the information pertinent to site selection and evaluation that is currently available from Apollo and Luna lunar samples and from remote sensing of the lunar surface. The use of this information in locating minable He-rich areas of the Moon is discussed, and preliminary conclusions as to favorable sites for mining are presented. Further work needed for site selection and evaluation is outlined.

BASIS OF SITE SELECTION

Of prime importance to site selection and evaluation are the following salient findings by the various investigators of lunar samples:

1. Regoliths from areas underlain by highland materials contain less than 20 wppm He, and many contain less than 10 wppm.

2. Regoliths of some maria or parts of maria contain less than 20 wppm He, but others have He contents ranging from 25 to nearly 50 wppm.

3. The He content of a mare regolith is a function of its composition. In particular, the He content appears to be a function of the Ti content of the regolith.

4. Helium is concentrated in the <100-μm size fractions of regoliths.

These findings are used directly in site selection. Moreover, they serve for calibration of data from remote sensing and as controls on interpretation of such data.

TITANIUM AND HELIUM IN LUNAR REGOLITHS

The relationship between the He contents of regoliths and their Ti contents is shown in Fig. 1, in which He content is plotted against TiO_2 content for samples of highland and mare regoliths. Highland regoliths are all low in both He and Ti. Mare regoliths fall into two groups, one with high He and Ti contents, the other with low contents of the two elements. It is generally accepted that the compositions of mare regoliths are controlled by the nature of underlying basaltic rocks. More than a dozen different types of basalts have been described from various maria, being distinguished on the basis of mineral and chemical composition (*Basaltic Volcanism Study Project,* 1981a; *Wilhelms,* 1987). In terms of Ti content, however, these basalts are assigned to three principal groups: (1) very-high-Ti basalts (VHT) sampled by Apollo 11 and Apollo 17; TiO_2 content 8 to 14 wt%; (2) low-Ti basalts (LT) sampled by Apollo 12, Apollo 15, Apollo 17, Luna

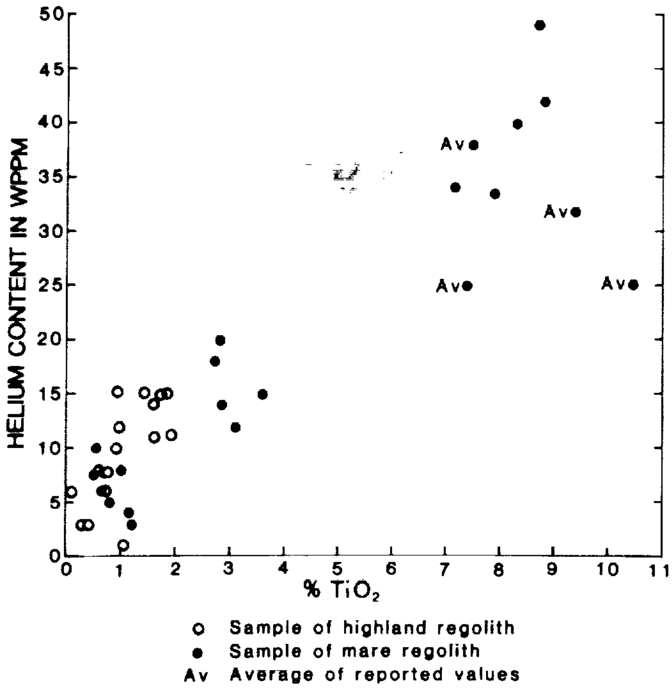

Fig. 1. Relation between He contents and TiO_2 contents of lunar regolith samples. Data from *Bogard and Hirsch* (1978a), *Bogard and Nyquist* (1972), *Criswell and Waldron* (1982), *Cuttitta et al.* (1971, 1973), *Eberhardt et al.* (1972), *Eugster et al.* (1985), *Funkhauser et al.* (1970), *Haskin et al.* (1973), *Heymann et al.* (1970, 1972a,b, 1973, 1978), *Hintenberger et al.* (1970, 1971, 1974, 1975), *Hintenberger and Weber* (1973), *Hübner et al.* (1973, 1975), *Kirsten et al.* (1972), *Laul et al.* (1974), *Laul and Papike* (1980a,b), *Laul and Schmitt* (1973), *Ma et al.* (1978), *Marti et al.* (1970), *Nava* (1974), *Pepin et al.* (1970), *Pieters et al.* (1980), *Pieters and McCord* (1976), *Rose et al.* (1974), *Wakita and Schmitt* (1971), *Wänke et al.* (1971), *Wilhelms* (1987), and *Willis et al.* (1972).

16, and Luna 24; TiO_2 content 1.5 to 5 wt%; and (3) very-low-Ti basalts (VLT) sampled by Apollo 17 and Luna 24; TiO_2 content less than 1.5 wt%.

In Fig. 1, the mare regolith samples fall into three distinct clusters corresponding rather closely to the above groupings of basalts on the basis of Ti content. The gap between VHT regoliths and LT regoliths is conspicuous. The diagram shows a broad correlation of He content with Ti content, but there is a considerable scatter, especially at the high TiO_2 end of the range. The samples highest in TiO_2 content are not all correspondingly high in He content. This is not unexpected. In lunar basalts, much of the Ti is in the form of ilmenite, and ilmenite fractions from Apollo 11 and Apollo 17 mare regoliths have been found to be enriched in He (*Eberhardt et al.*, 1970, 1972; *Hintenberger et al.*, 1974). However, ilmenite grains in high-Ti basalts range from a millimeter to less than a micrometer in size (see, for example, *Cameron*, 1970, Figs. 1 and 2), and variations in grain size are bound to affect the degree of exposure of ilmenite to the solar wind even in particles of the finer size fractions of regoliths. Moreover, TiO_2 can be dissolved in part in agglutinate glass or in volcanic, pyroclastic Ti-rich glass particles. The latter particles in Apollo 11 regolith have been shown by *Kirsten et al.* (1970) to carry higher concentrations of He than associated particles of olivine, pyroxene, and plagioclase, but are less enriched than ilmenite grains. Finally, the He content of regolith is a function of the length of time of exposure to the solar wind, the ultimate source of lunar He.

Considering all the factors that have affected the absorption of He by regolith, a linear relation between TiO_2 content and He content is not to be expected. However, it seems clear that the Ti content of regolith can be used as a general guide in selection of areas where the regolith contains at least 20 wppm He and hopefully areas where the regolith contains at least 30 wppm. This is of critical importance in site selection. Only minute fractions of a few maria have yet been sampled. For information on the extent and distribution of the more He-rich maria and portions of maria, we must presently rely on information from remote sensing of the Ti contents of mare regoliths.

INFORMATION FROM REMOTE SENSING

Broad reviews of both methods and results of remote sensing have been given by *Moore et al.* (1980) and the *Basaltic Volcanism Study Project* (1981b). Two general types of remote sensing have furnished information on the Ti contents of lunar regoliths, namely, gamma-ray spectroscopy performed by Apollo 15 and Apollo 16 orbiters, and Earth-based telescopic measurement of lunar reflectance. The results of both types of measurements have been calibrated, as far as possible, against returned lunar samples of known Ti content, but Fig. 1 indicates that there can be no calibration for the intermediate range, i.e., for the gap between VHT and LT basalts. Whether this gap is real or is due to incomplete sampling of the maria is discussed in a subsequent section of this paper.

Gamma-ray spectroscopy makes use of radiation produced mainly by cosmic-ray bombardment of the lunar regolith. Its advantage is that it measures a property that is uniquely related to Ti content. However, there are serious deficiencies in gamma-ray data presently available. Resolution is very low, about 100×100 km. Coverage by the Apollo orbiters is limited to two bands lying between 30°N and 15°S. There are problems in interpreting the data, in part due to interference with Ti lines by Fe and O in the gamma-ray spectra. Nonetheless, the gamma-ray surveys are valuable because they indicate broad variations in Ti content of regolith over the equatorial region of the lunar nearside.

Figure 2 shows variations in the Ti content of lunar regolith as interpreted by *Metzger and Parker* (1980) from gamma-ray spectroscopy. Two principal areas of high-Ti regolith are indicated, one the area of Mare Tranquillitatis, with its extension northward into Mare Serenitatis (sampled at the Taurus-Littrow region of Apollo 17), the other a part of Oceanus Procellarum. Two smaller areas are also shown. Certain areas are shown as having intermediate Ti content (2.0-2.5% Ti), but none of these has been sampled by lunar missions, except possibly the one in Mare Fecunditatis. A soil sample recovered by Luna 16 contains 3.53% TiO_2 (*Criswell and Waldron*, 1982), but the He content of the bulk sample can only be approximated from data of *Vinogradov and Zadorozhny* (1972) for the <83-μm fraction, and it may be as high as 30 wppm. In view of this uncertainty, the sample is not plotted in Fig. 1. *Davis* (1980) has used the orbital gamma-ray data to produce a map on which variations in Ti content are

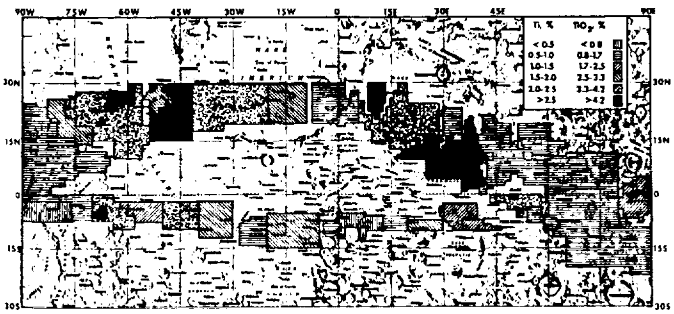

Fig. 2. Map of the Ti content of the lunar regolith covering nearside regions overflown by Apollo 15 and 16. From *Metzger and Parker* (1980), by permission of the authors and Elsevier Publishing Company.

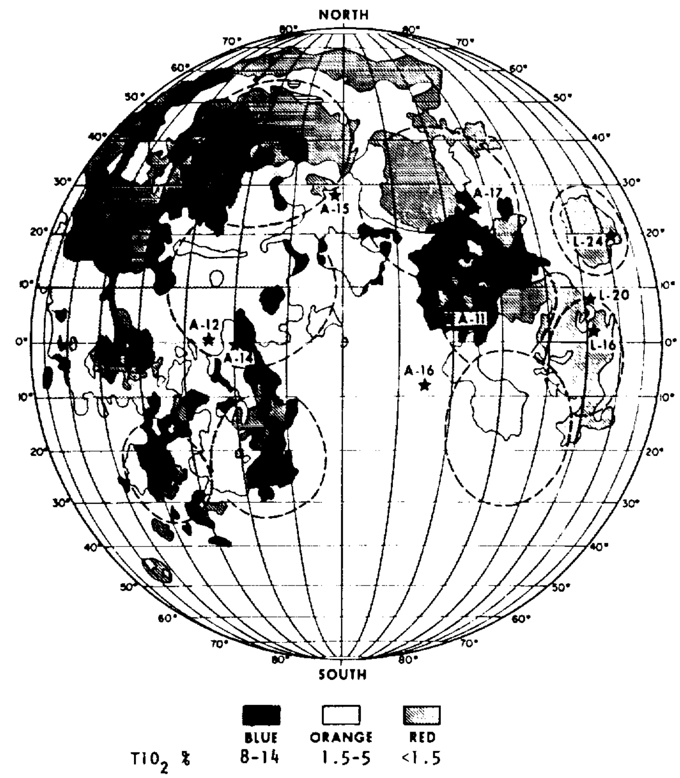

Fig. 3. Color groups of mare regoliths and TiO_2 values thought to be represented by the groups. From *Basaltic Volcanism Study Project* (1981a), by permission of the Lunar and Planetary Institute, Houston. Modified to show blue areas in solid black.

shown in image format. There are differences between his map and that of Fig. 2, but the broad picture of Ti variation is much the same.

There is a considerable variety of measurements of lunar reflectance. Figure 3 is a map of the entire lunar nearside prepared from superposed ultraviolet negatives and near-infrared positives. It shows the color groups of basaltic regoliths, with TiO_2 values thought to be indicated by the colors. Again, the only sizable area of high-Ti regolith shown in the eastern hemisphere is that of Mare Tranquillitatis, with its extension northward into the Apollo 17 area, but the map shows large areas of high-Ti regolith in the western hemisphere. Like Fig. 2, Fig. 3 shows large areas thought to be of intermediate Ti content.

Figure 3 is actually based on one form of spectral ratio mapping. Quantitative spectral ratio mapping is based on use of the 0.38 μm:0.56 μm ratio, the 0.38 μm:0.58 μm ratio, or the 0.38 μm:0.62 μm ratio, all of them UV:VIS ratios, or on the 0.38 μm:0.95 μm ratio (UV:IR). Compared to gamma-ray spectroscopy, spectral ratio mapping has the advantage of higher resolution (1 to 3 km) and broader coverage of the lunar surface. *Johnson et al.* (1977) prepared a map of a large part of the lunar nearside using the 0.38 μm:0.56 μm ratio. On it, Mare Tranquillitatis appears once again as a high-Ti area, but the higher resolution of this method shows that the mare is not uniform in Ti content. Spectral ratio mapping appears to be a good indicator in the high-Ti and low-Ti ranges, but it is ambiguous in the intermediate-Ti range, as indicated in Fig. 4. This is unfortunate, because it is in this range that sampling of the lunar surface is lacking.

Figure 3 is actually a version of a remarkable color difference photograph (Fig. 5) prepared by *Whitaker* (1965). High-Ti regoliths appear dark in this photograph, and it shows that high-Ti regolith in Mare Tranquillitatis is mainly in an irregular belt extending northward along the west side of the mare and thence eastward across it as shown in Fig. 6. The area south of the belt is a complex of high-Ti and lower-Ti regoliths, whereas the area north of the belt is apparently one of low-Ti regolith. Comparable variations in Mare Imbrium and Oceanus Procellarum are also suggested by the photograph.

Figure 7 shows the distribution of a number of petrographic types of basaltic regoliths as recognized by *Pieters* (1978) on the

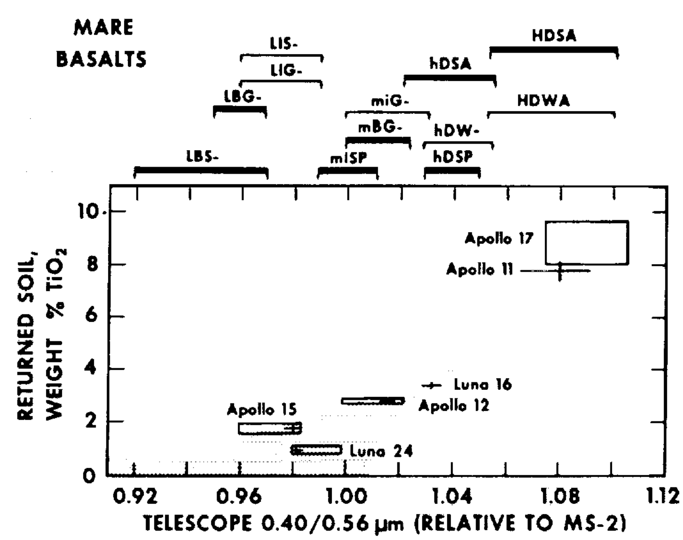

Fig. 4. Relationship between percent TiO_2 in lunar mare regoliths and the 0.40:0.56-μm reflectance ratio for telescopic spectra relative to MS2 (after *Charette et al.*, 1974, and *Pieters and McCord*, 1976). The stippled area is the estimated range of TiO_2 that can be derived from a 0.40:0.56 μm ratio measurement of mature mare regions. Shown above the plot are the ranges of 0.46:0.56 μm ratio observed for each of the basalt types discussed by *Pieters* (1978). The heavy lines indicate unsampled basalt types. From *Basaltic Volcanism Study Project* (1981a), by permission of the Lunar and Planetary Institute, Houston.

Fig. 5. Color difference photograph (from *Whitaker,* 1966), made by subtracting a photograph taken at 0.31 μm from one taken at 0.61 μm. Courtesy of E. A. Whitaker.

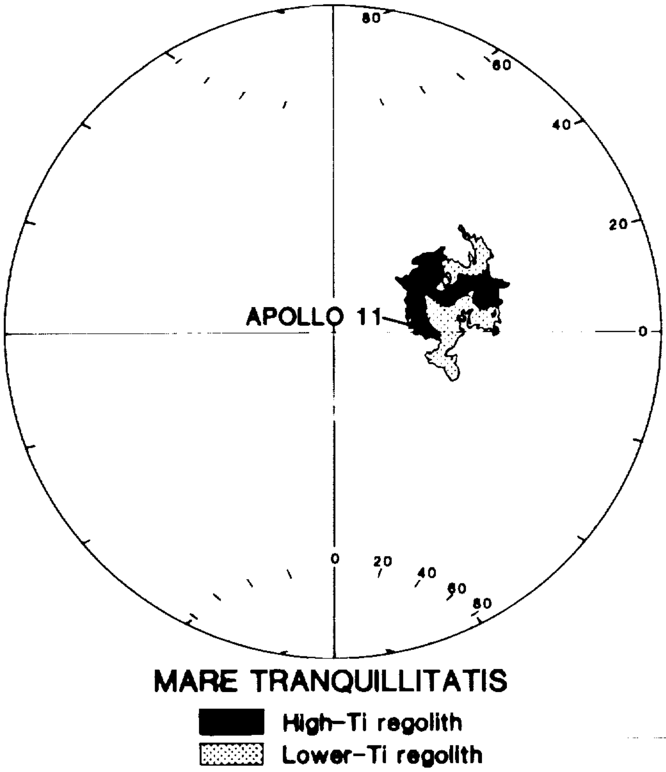

Fig. 6. Broad variations in TiO_2 content of regolith in Mare Tranquillitatis. Based on *Wilhelms* (1987, plate 4A).

basis of albedo, UV:VIS ratio, strength of the 1-μm band, and strength of the 2-μm band. Only Tranquillitatis and the Flamsteed area are regarded as occupied by high-Ti regolith.

There is considerable agreement among various maps that are based on reflectance measurements. There is agreement on certain high-Ti areas. There is also agreement that regoliths of intermediate or low-Ti content occupy considerably more area than those identified as high-Ti regoliths. However, there is disagreement on the location and extent of low-Ti and intermediate-Ti areas. *Metzger et al.* (1979) compared the results of gamma-ray spectroscopy with the work of Pieters expressed in Fig. 7. They found agreement for 10 of 14 map regions. However, this required a rather liberal intepretation of the data. In addition, they assume a uniformity of maria that probably does not exist and is, in fact, contradicted by the spectral ratio map of *Johnson et al.* (1977) and the photograph of Fig. 5. As noted above, there is a lack of intermediate-Ti samples that have been analyzed for both He and Ti. Given the poor resolution of most reflectance measurements, intermediate-Ti areas shown on maps may therefore be due to averaging high-Ti and low-Ti regoliths. The gap between the two suggested by Fig. 1 may therefore be a real one, not an apparent one that is due to incomplete sampling of the maria. Lack of samples from areas that are indicated as intermediate in Ti content is particularly unfortunate inasmuch as those areas occupy a substantial fraction of the total area of the maria.

It is important that there is substantial agreement among the maps on the locations of certain areas of high-Ti regolith, Mare Tranquillitatis in particular. Tranquillitatis has an area of at least 190,000 sq km and seems likely to contain substantial amounts of

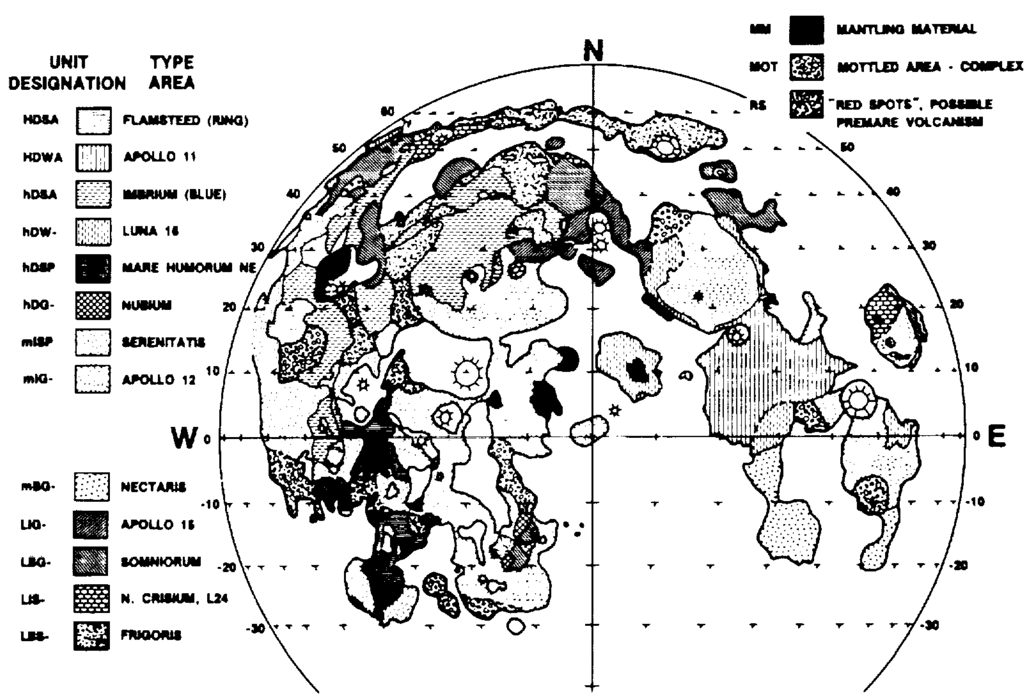

Fig. 7. Major basalt types for the front side of the Moon as derived from the current spectral reflectance data. The unit designations represent values for four measurable parameters (UV:VIS ratio, albedo, strength of the 1-μm band, and strength of the 2-μm band). By permission of the Lunar and Planetary Institute and Basaltic Volcanism Study Project.

regolith with 30 wppm or more He. If only 30% of the total area is minable at 30 wppm He, if mining is carried to an average depth of 3 m, and if recovery from mining and processing is 60%, roughly 1700 tonnes of He-3 could be recovered from this mare.

In summary, remote sensing has yielded a substantial amount of information on the location and broad distribution of high-Ti regoliths. The information serves as a general guide to the selection of potential mining areas. Beyond this, however, the use of remote sensing is limited by the low resolution of the methods.

REQUIREMENTS FOR MINING AREAS

The mining scenario envisioned by the University of Wisconsin group calls for the schedule of production shown in Table 1. In Fig. 8, line A shows the area in square kilometers having the amount of He-3 necessary to match the requirements for successive periods between 2015 and 2050, assuming a mining depth of 3 m, an average He content of 30 wppm, and 100% recovery of He. Line B shows square kilometers that must be mined to a depth of 3 m to meet the schedule if recovery from mining and processing is 80%. Line C shows square kilometers that must be mined to a depth of 3 m if recovery is 60%, probably a more realistic figure. At 10% recovery, the mined area would be 123 sq km by the year 2030, and 8110 sq km by the year 2050, when the full production rate of 20,000 kg He-3/yr would be achieved. Thenceforth, the mining area required per year would be 665 sq km. Thus, large mining areas must be delineated if the mining scenario is to be fulfilled.

The He content of regolith, recovery percentage, and depth of mining are not the only factors determining the areas over which mining operations will have to be extended. Knowledge of the maria, including Mare Tranquillitatis, is far from complete, but it is enough to indicate that there will be areas that cannot be mined, at acceptable costs, owing to the presence of large craters or abundant large blocks of rock. It is therefore to be expected that mining will have to extend over larger areas than indicated by Fig. 8, and that the pattern of mining will be complicated by the necessity of avoiding unminable areas.

Regolith high in Ti was found in the mare-filled valley of the Apollo 17 landing site. However, the site is one of complex geology and marked heterogeneity. It does not appear attractive as a potential mining site, but photogeologic maps (*Scott et al.*, 1972) suggest that there may be minable areas in the Taurus-Littrow region west of the landing site.

DISTRIBUTION OF HELIUM IN MARE REGOLITHS

As indicated earlier, investigations of lunar samples have shown that He is concentrated in the <100-μm size fractions of regoliths. This stems from the fact that absorption of He from the solar wind is proportional to particle surface area per unit of mass. The distribution of He in Apollo 11 regolith is indicated in Fig. 9, which is based on weight percentages of various size fractions given by *Criswell and Waldron* (1982) for sample 10084,853, and on He contents of a series of size fractions given by *Hintenberger et al.* (1970) for sample 10084,18. The latter authors did not give the weight percentages of the bulk soil represented by the various fractions, and the percentages had to be estimated from a size distribution curve plotted from the data of Criswell and Waldron. Calculations show that not all the He content of the bulk soil is accounted for in the size fractions, hence Fig. 9 should be taken as indicating only the pattern of He distribution in Apollo 11 regolith in relation to particle size.

TABLE 1. Schedule of He-3 production.

Period	Average annual production (mt)	Cumulative production (mt)
2015-2020	0.014	0.070
2020-2025	0.066	0.400
2025-2030	0.656	3.680
2030-2035	4.572	25.540
2035-2040	9.312	73.100
2040-2045	14.880	147.500
2045-2050	19.300	244.000

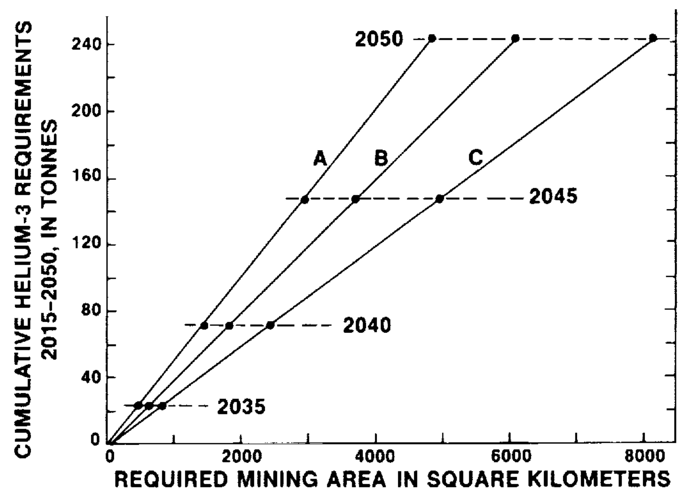

Fig. 8. Relation between required mining areas and cumulative requirements for He-3 for a mining depth of 3 m. See text for explanation.

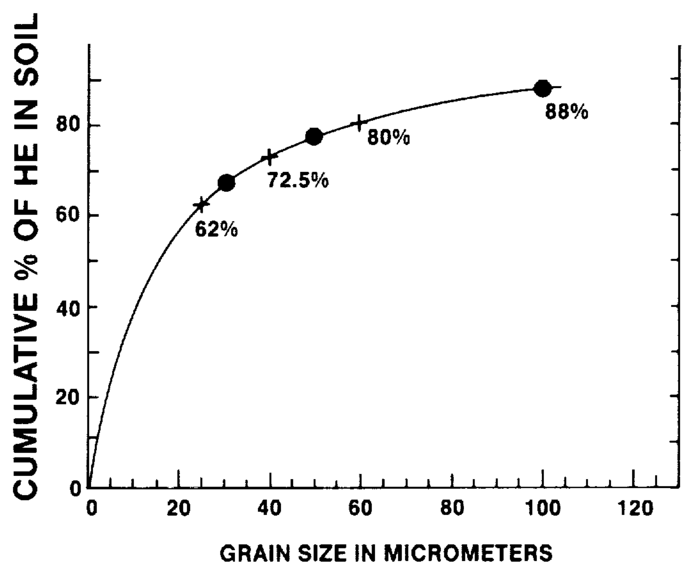

Fig. 9. Percentage of total He in Apollo 11 regolith in relation to grain size. Based on data of *Criswell and Waldron* (1982) and *Hintenberger et al.* (1970).

The significance of the above relations for site selection is that coarse material in the regolith is not important as a source of He. This and considerations of ease of mining and processing mean that mining areas should be as free as possible of blocks of rock and sizable craters. Information on these and other physical features of the regolith must be obtained from lunar photographs and photogeologic maps, and from radar surveys that indicate surface roughness at various scales (*Basaltic Volcanism Study Project,* 1981b; *Moore et al.,* 1980; *Zisk et al.,* 1974, 1987). Photogeologic maps should also aid in studying variations in the composition of lunar regoliths and in delineating suitable mining areas.

DARK-MANTLE MATERIALS AS AN ALTERNATIVE SOURCE OF HELIUM

Dark-mantle materials, distinguished on the basis of their radar and spectral reflectances, occupy certain areas of the lunar nearside. The largest such areas are along the southwest and south sides of Mare Serenitatis, including the Sulpicius Gallus and Taurus-Littrow regions, and in the Rima Bode region (*Lucchitta,* 1974; *Head,* 1974; *Lucchitta and Schmitt,* 1974; *Wilhelms,* 1987). Dark-mantle materials typically occur along the margins of maria, extending over parts of adjacent highlands. The deposits range up to tens of meters in thickness. Consisting of glass droplets with a mean size in the neighborhood of 40 μm, dark-mantle materials could be mined and processed for He much more easily that normal mare regolith.

From studies of spectral reflectance, *Adams et al.* (1974) concluded that droplets of black glass are the essential ingredient of dark-mantle material associated with Mare Serenitatis. The black glass has formed by devitrification of droplets of orange glass, now considered to be of pyroclastic origin. *Haggerty* (1974) and *Heiken and McKay* (1978) have shown that black ash droplets consist of ilmenite and olivine with varying amounts of residual glass.

Orange glass was sampled at the Apollo 17 site in the trench at Shorty Crater and in drive tubes put down beside the trench. Orange ash is reported to contain 8.09% to 8.9% TiO_2 (*Wänke et al.,* 1971; *Nava,* 1974). Its He content is reported as 2.6 wppm (*Hintenberger et al.,* 1974). The drive tubes show a progression downward from nearly pure orange glass at the top to material with 73.4% black glass at the bottom (*Heiken and McKay,* 1974).

In view of the ilmenite content of the black glass, it might be expected to show a higher He content than the orange glass, but *Bogard and Hirsch* (1978b) found He contents of 0.9 wppm or less in the black glass. However, petrographic studies of the drive tube section show that only the top 5.5 cm of the section has been gardened, and it has been exposed to the solar wind for only the past 10 to 30 m.y. We still do not know, therefore, the He potential of black glass in areas where it has been gardened and exposed to the solar wind over long periods of time. Such areas might prove to be important sources of helium. Sampling them should be a part of future missions to the Moon.

CONCLUDING COMMENTS

The study now in progress and reported here is preliminary. Study of photographs, photogeologic maps, and radar surveys of the lunar nearside should advance the process of site selection, but final selection of mining sites will not be possible with data presently available. Only very small fractions of a few of the maria have been sampled, and no area has been systematically sampled. Information on thickness of regolith and on variation of He content with depth is limited. Remote-sensing maps, both those based on gamma-ray spectroscopy and on reflectance measurements, have insufficient resolution for purposes of site selection. All remote-sensing maps show large areas of intermediate-Ti regolith, but no such regolith has yet been sampled. The He potential of dark-mantle materials cannot currently be appraised. These significant deficiencies in present information must be remedied by systematic exploration and sampling of regolith, definition of minable portions of regolith, and estimation of tonnage and He content. Given the enormous potential of lunar He as a source of energy, such work should have a high priority in future lunar missions.

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REFERENCES

Adams J. B., Pieters C., and McCord T. B. (1974) Orange glass: Evidence for regional deposits of pyroclastic origin on the Moon. *Proc. Lunar Sci. Conf. 5th,* pp. 171-180.

Basaltic Volcanism Study Project (1981a) Lunar mare basalts. In *Basaltic Volcanism on the Terrestrial Planets,* pp. 236-267. Pergamon, New York.

Basaltic Volcanism Study Project (1981b) Remote sensing of basalts in the solar system. In *Basaltic Volcanism on the Terrestrial Planets,* pp. 439-490. Pergamon, New York.

Bogard D. D. and Hirsch W. C. (1978a) Noble gases in Luna 24 core. In *Mare Crisium: The View from Luna 24* (J. J. Papike and R. B. Merrill, eds.), pp. 105-116. Pergamon, New York.

Bogard D. D. and Hirsch W. C. (1978b) Depositional and irradiational history and noble gas content of orange-black droplets in the 74001/2 core from Shorty Crater. *Proc. Lunar Sci. Conf. 9th,* pp. 1981-2000.

Bogard D. D. and Nyquist L. E. (1972) Noble gas studies on regolith materials from Apollo 14 and 15. *Proc. Lunar Sci. Conf. 3rd,* pp. 1797-1820.

Cameron E. N. (1970) Opaque minerals in certain lunar rocks from Apollo 11. *Proc. Apollo 11 Lunar Sci. Conf.,* pp. 221-245.

Charette M. P., McCord T. B., Pieters C. M., and Adams J. B. (1974) Application of remote spectral reflectance measurements to lunar geology classification and determination of titanium content of lunar soils. *J. Geophys. Res., 79,* 1605-1613.

Criswell D. R and Waldron R. D. (1982) Lunar utilization. In *Space Utilization,* Vol. II (B. O'Leary, ed.), pp. 1-53. CRC, Boca Raton, Florida.

Cuttitta F. Jr., Rose H. J. Jr., Annell C. S., Carron M. K., Christian R. P., Ligon D. T. Jr., Dwornik E. J., Wright T. L., and Greenland L. P. (1973) Chemistry of twenty-one igneous rocks and soils returned by the Apollo 15 mission. *Proc. Lunar Sci. Conf. 4th,* pp. 1081-1086.

Cuttitta H., Rose H. J. Jr., Annell C. S., Carron M. K., Christian R. P., Dwornik E. J., Greeland L. P., Helz A. W., and Ligon D. T. Jr. (1971) Elemental composition of some Apollo 12 lunar rocks and soils. *Proc. Lunar Sci. Conf. 2nd,* pp. 1217-1229.

Davis P. A. (1980) Iron and titanium distribution in the Moon from orbital gamma-ray spectrometry with implications for crustal evolutionary models. *J. Geophys. Res., 85,* 3209-3224.

Eberhardt P., Geiss J., Graf H., Grögler N., Krähenbuhl U., Schwaller H., Schwarzmuller J., and Stettler A. (1970) Trapped solar wind noble gases, exposure age and K/Ar age in Apollo 11 lunar fine material. *Proc. Apollo 11 Lunar Sci. Conf.,* pp. 1037-1070.

Eberhardt P., Geiss J., Graf H., Grögler N., Mendia M. D., Mörgeli M., Schwaller H., and Stettler A. (1972) Trapped solar wind noble gases in Apollo 12 lunar fines 12002 and Apollo 11 breccia 10046. *Proc. Lunar Sci. Conf. 3rd,* pp. 1821-1856.

Eugster O., Eberhardt P., Geiss J., Grögler N., Jungck M., and Mörgeli M. (1975) Solar-wind-trapped and cosmic-ray-produced noble gases in Luna 20 soil. *Proc. Lunar Sci. Conf. 6th*, pp. 1989-2007.

Funkhouser J., Jessberger E., Müller O., and Zähringer J. (1970) Active and inert gases in Apollo 12 and Apollo 11 samples released by crushing at room temperature and by heating at low temperatures. *Proc. Lunar Sci. Conf. 2nd*, pp. 1381-1396.

Haggerty S. E. (1974) Apollo 17 orange glass: Textural and morphological characteristics of devitrification. *Proc. Lunar Sci. Conf. 5th*, pp. 193-206.

Haskin L. A., Helmke P. A., Blanchard D. F., Jacobs J. W., and Telander K. (1973) Major and trace element abundances in samples from the lunar highlands. *Proc. Lunar Sci. Conf. 4th*, pp. 1275-1296.

Head J. W. (1974) Lunar dark-mantle deposits: Possible clues to the distribution of early mare deposits. *Proc. Lunar Sci. Conf. 5th*, pp. 207-222.

Heiken G. H. and McKay D. S. (1974) Petrography of Apollo 17 soils. *Proc. Lunar Sci. Conf. 5th*, pp. 843-860.

Heiken G. H. and McKay D. S. (1978) Petrology of a sequence of pyroclastic rocks from the valley of Taurus-Littrow (Apollo 17 landing site). *Proc. Lunar Planet. Sci. Conf. 9th*, pp. 1933-1943.

Heymann D. and Yaniv A. (1970) Inert gases in the fines from the Sea of Tranquility. *Proc. Apollo 11 Lunar Sci. Conf.*, pp. 1247-1259.

Heymann D., Yaniv A., and Lakatos S. (1972a) Inert gases in 12 particles and one dust sample from Luna 16. *Earth Planet. Sci. Lett.*, *13*, 400-406.

Heymann D., Yaniv A., and Lakatos S. (1972b) Inert gases from Apollo 12, 14, and 15 fines. *Proc. Lunar Sci. Conf. 3rd*, pp. 1857-1863.

Heymann D., Lakatos S., and Walton J. R. (1973) Inert gases in a terra sample. *Geochim. Cosmochim. Acta*, *37*, 875-885.

Heymann D., Jordan J. L., Walker A., Dziczkaniec M., Ray J., and Palma R. (1978) Inert gas measurements in the Apollo 16 drill core and an evaluation of the stratigraphy and depositional history of this core. *Proc. Lunar Planet. Sci. Conf. 9th*, pp. 1885-1912.

Hintenberger H. and Weber H. W. (1973) Trapped rare gases in lunar fines and breccias. *Proc. Lunar Sci. Conf. 4th*, pp. 2003-2020.

Hintenberger H., Weber H. W., Voshage H., Wänke H., Begemann F., and Wlotzka F. (1970) Concentrations and isotopic abundances of the rare gases, hydrogen, and nitrogen in lunar matter. *Proc. Apollo 11 Lunar Sci. Conf.*, pp. 1269-1282.

Hintenberger H., Weber H. W., and Takaokoka N. (1971) Concentration and isotopic abundances of rare gases in lunar matter. *Proc. Lunar Sci. Conf. 2nd*, pp. 1617-1626.

Hintenberger H., Weber H. W., and Schultz L. (1974) Solar, spallogenic, and radiogenic rare gases on Apollo 17 soils and breccias. *Proc. Lunar Sci. Conf. 5th*, pp. 2005-2022.

Hintenberger H., Schultz L., and Weber H. W. (1975) A comparison of noble gases in lunar fines and soil breccias. *Proc. Lunar Sci. Conf. 6th*, pp. 2261-2270.

Hübner W., Heymann D., and Kirsten T. (1973) Inert gas stratigraphy of Apollo 15 core sections 15001 and 15003. *Proc. Lunar Sci. Conf. 4th*, pp. 2021-2036.

Hübner W., Kirsten T., and Kiko J. (1975) Rare gases in Apollo 17 soils with emphasis on analysis of size and mineral fractions of soil 74241. *Proc. Lunar Sci. Conf. 6th*, pp. 2009-2026.

Johnson T. V., Saunders R. S., Matson D. L., and Mosher J. L. (1977) A TiO_2 abundance map for the northern maria. *Proc. Lunar Sci. Conf. 8th*, pp. 1029-1036.

Kirsten T., Müller O., Steinbrunn F., and Zähringer J. (1970) Study of distribution and variations of rare gases in lunar material by a microprobe technique. *Proc. Apollo 11 Lunar Sci. Conf.*, pp. 1331-1343.

Kirsten T., Deubner J., Horn P., Kaneoka I., Kiko J., Schaeffer O. A., and Thio S. K. (1972) The rare gas record of Apollo 14 and 15 samples. *Proc. Lunar Sci. Conf. 3rd*, pp. 1865-1890.

Laul J. C. and Papike J. J. (1980a) The lunar regolith: Comparative chemistry of the Apollo sites. *Proc. Lunar Planet. Sci. Conf. 11th*, pp. 1307-1340.

Laul J. C. and Papike J. J. (1980b) The Apollo 17 drill core: Chemistry of size fractions and the nature of the fused soil component. *Proc. Lunar Planet. Sci. Conf. 11th*, pp. 1395-1413.

Laul J. C. and Schmitt R. A. (1973) Chemical composition of Apollo 15, 16, and 17 samples. *Proc. Lunar Sci. Conf. 4th*, pp. 1349-1367.

Laul J. C., Hill D. W., and Schmitt R. A. (1974) Chemical studies of Apollo 16 and 17 samples. *Proc. Lunar Sci. Conf. 5th*, pp. 1047-1066.

Lucchitta B. K. (1973) Photogeology of the dark material in the Taurus-Littrow region of the Moon. *Proc. Lunar Sci. Conf. 4th*, pp. 221-274.

Lucchitta B. K. and Schmitt H. (1974) Orange material in the Sulpicius Gallus region of the Moon. *Proc. Lunar Sci. Conf. 4th*, pp. 221-274.

Ma M. S., Schmitt R. A., Taylor G. J., Warner R. D., Langer D. E., and Keil K. (1978) Chemistry and petrology of Luna 24 lithic fragments and $<250\ \mu m$ soils: Constraints on the origin of VLT basalts. In *Mare Crisium: The View from Luna 24* (J. J. Papike and R. B. Merrill, eds.), pp. 569-592. Pergamon, New York.

Marti K., Lugmair G. W., and Urey H. C. (1970) Solar wind gases, cosmic-ray spallation products and the irradiation history of Apollo 11 samples. *Proc. Apollo 11 Lunar Sci. Conf.*, pp. 1357-1367.

Metzger A. E. and Parker R. E. (1980) The distribution of titanium on the lunar surface. *Earth Planet. Sci. Lett.*, *45*, 155-171.

Metzger A. E., Johnson T. V., and Matson D. L. (1979) A comparison of mare surface concentrations obtained by spectral reflectance and gamma-ray spectroscopy: An early assessment. *Proc. Lunar Planet Sci. Conf. 10th*, pp. 1719-1726.

Moore H. J., Boyce J. M., Schaber G. G., and Scott D. H. (1980) Lunar remote sensing and measurements. *U.S. Geol. Surv. Prof. Pap. 1046-B*, pp. B71-B78.

Nava D. F. (1974) Chemical compositions of some soils and rocks from the Apollo 15, 16 and 17 lunar sites. *Proc. Lunar Sci. Conf. 5th*, pp. 1087-1096.

Pepin R. O., Nyquist L. E., Phinney D., and Black D. C. (1970) Rare gases in Apollo 11 lunar material. *Proc. Apollo 11 Lunar Sci. Conf.*, pp. 1435-1454.

Pieters C. (1978) Mare basalt types on the front side of the Moon: A summary of spectral reflectance data. *Proc. Lunar Planet. Sci. Conf. 9th*, pp. 2825-2849.

Pieters C. and McCord T. B. (1976) Characterization of lunar mare basalt types: I. Remote sensing study using reflectance spectroscopy of surface soils. *Proc. Lunar Sci. Conf. 7th*, pp. 2677-2790.

Pieters C., Head J. W., Adams J. B., McCord T. B., Zisk S., and Whitford-Stark J. (1980) Late high-titanium basalts of the western maria: Geology of the Flamsteed region of Oceanus Procellarum. *J. Geophys. Res.*, *85*, 3913-3938.

Rose H. J. Jr., Cuttitta F., Berman S., Brown F. W., Carron M. K., Christian R. P., Dwornik E. J., and Greenland L. P. (1974) Chemical composition of rocks and soils at Taurus-Littrow. *Proc. Lunar Sci. Conf. 5th*, pp. 1119-1134.

Scott D. H., Lucchitta B. K., and Carr M. H. (1972) Geologic maps of the Taurus-Littrow region of the Moon. *U. S. Geol. Surv. Map I-800.*

Vinogradov A. P. and Zadorozhny I. K. (1972) Rare gases in the regolith from the Sea of Fertility. *Space Res.*, *13*, 23-31.

Wakita H. and Schmitt R. A. (1971) Bulk elemental composition of Apollo 12 samples: Five igneous rocks and one breccia, and four soils. *Proc. Lunar Sci. Conf. 2nd*, pp. 1231-1236.

Wänke H., Wlotzka F., Baddenhausen H., Balacescu A., Spettel B., Teschke F., Jagoutz E., Kruse H., Quijano M., and Rieder R. (1971) Chemical composition and its relation to sample locations and exposure ages, the two-component origin of the various soil samples, and studies on lunar metallic particles. *Proc. Lunar Sci. Conf. 2nd*, pp. 1187-1208.

Whitaker E. A. (1965) The surface of the Moon. In *The Nature of the Moon* (W. N. Hess, D. H. Menzel, and J. A. O'Keefe, eds.), pp. 79-98. Johns Hopkins, Baltimore.

Wilhelms D. E. (1987) *The Geological History of the Moon.* U.S. Geol Surv. Prof. Pap. 1347. 302 pp.

Willis J. P., Erlank A. J., Gurney J. J., Theil H. H., and Ahrens L. H. (1972) Major, minor, and trace element data for some Apollo 11, 12, 14, and 15 samples. *Proc. Lunar Sci. Conf. 3rd*, pp. 1269-1273.

Zisk S. H., Pettengill G. H., and Catura G. W. (1974) 3.8 cm atlas of the Moon. *Proc. Lunar Sci. Conf 6th,* pp. 1653-1672.

Zisk S. H., Mouginis-Mark P. J., Pettengill G. H., and Thompson T. W. (1987) New very-high resolution lunar radar measurements at 3.0 cm wavelength: Initial maps of the Hadley Rille/Apollo 15 area (abstract). In *Lunar and Planetary Science XVIII,* pp. 1030-1031. Lunar and Planetary Institute, Houston.

THE LUNAR ORBITAL PROSPECTOR

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The establishment of lunar bases will not end the need for remote sensing of the lunar surface by orbiting platforms. Human and robotic surface exploration will necessarily be limited to some proximate distance from the support base. Near real-time, high-resolution, global characterization of the lunar surface by orbiting sensing systems will continue to be essential to the understanding of the Moon's geophysical structure and the location of exploitable minerals and deposits of raw materials. The Lunar Orbital Prospector (LOP) is an orbiting sensing platform capable of supporting a variety of modular sensing packages. Serviced by a lunar-based shuttle, the LOP will permit the exchange of instrument packages to meet evolving mission needs. The ability to recover, modify, and rotate sensing packages allows their reuse in varying combinations. Combining this flexibility with robust orbit modification capabilities and near real-time telemetry links provides considerable system responsiveness. Maintenance and modification of the LOP orbit are accomplished through use of an onboard propulsion system that burns lunar-supplied oxygen and aluminum. The relatively low performance of such a system is more than compensated for by the elimination of the need for Earth-supplied propellants. The LOP concept envisions a continuous expansion of capability through the incorporation of new instrument technologies and the addition of platforms.

INTRODUCTION

Human and robotic exploration of the Moon during the last two decades has greatly increased our knowledge of the Moon's geophysical and geochemical nature. The Ranger, Surveyor, Lunar Orbiter, Luna, and Apollo missions returned vast amounts of data by means of orbit-based photography, remote sensing, and lunar surface samples. Notwithstanding these efforts, coverage of the Moon by remote sensing remains woefully inadequate. Data from the Ranger and Surveyor missions were limited to a few isolated sites, and the Lunar Orbiter missions provided only photographic information. Although the Apollo command modules carried remote-sensing instruments, their data were obtained only at low latitudes.

One of the primary reasons for establishing a manned lunar presence is the possibility of exploiting the Moon's resources. In addition to a known abundance of lunar oxygen and various metals, undiscovered resources, possibly large deposits of lunar ores, probably exist in permanently shadowed polar craters (*Watson et al.,* 1961), and early lunar volcanic activity may have provided a mechanism for forming large ore and mineral deposits near the lunar surface (*Vaniman and Heiken,* 1985). Given the known resource potential of only a few explored lunar sites (high concentrations of oxygen, titanium, and aluminum were found in many Apollo surface samples), the existence of large deposits of these and other yet undiscovered lunar ores seems highly likely.

Although a manned presence is required for the recovery of lunar resources, a global search for these resources requires highly advanced remote-sensing satellites.

A remote-sensing orbital mission, such as the planned Lunar Geoscience Observer (LGO), is a necessary precursor to the development of a manned lunar base. The need for a mission of this nature, however, does not end with the establishment of the base. Several of the most fundamental geophysical and geochemical questions, such as the composition, structure, and thermal state of the interior, can be adequately addressed only by long-term observation and electromagnetic sounding of the lunar surface (*Hood et al.,* 1985). A long-term remote-sensing mission, in conjunction with a manned lunar base, can expand the LGO's geophysical and geochemical database and serve as the "eyes and ears" of the manned base by searching for lunar transient events and by monitoring humans' impact on the lunar environment. The LGO provides a great start, but its limited lifetime and its inability to evolve to meet changing mission requirements prevent its meeting long-term needs.

THE LUNAR ORBITAL PROSPECTOR

The Lunar Orbital Prospector (LOP) is a lunar-based, orbiting platform whose primary mission is to prospect the Moon in support of early lunar colonization and exploitation. Using the LGO mission as a baseline, the LOP mission is designed to support the next generation of lunar exploration in conjunction with a manned base.

The primary purposes of the LOP are to map the Moon's chemical and mineralogical composition and to conduct lunar science studies from orbit. Data returned from onboard passive and active sensors will identify mineral and chemical species and allow examination of surface and subsurface geological structures. Through careful processing and examination of sensed information, lunar resource distribution on a global scale can be determined.

Perhaps the LOP's most important characteristic, and its primary improvement over the LGO, is its highly modular design. Remote-sensing instrument packages can be exchanged or upgraded to fulfill many different mission requirements. Subsystems that support the science instruments, e.g., power and communications, can also be upgraded as needs dictate. This modularity also permits the refurbishing and reuse of sensing packages.

MISSION DESCRIPTION

The LOP mission is divided into three primary phases: transport from Earth to low lunar orbit (LLO), operation in lunar orbit, and platform servicing in lunar orbit. Transport of the LOP from Earth

can be accomplished by a vehicle with a 1000-kg translunar payload capabilty. This is within the range of the Titan 34D with an upper stage or a space shuttle/upper stage combination. The upper stage provides the initial translunar insertion burn, and the platform's onboard propulsion system provides midcourse corrections and lunar orbit insertion burns.

After delivery to LLO, normal orbital operation commences. The initial orbit is baselined to be a 100-km, near-polar orbit to permit global viewing of the lunar surface. The initial remote-sensing package is projected to be an updated version of the LGO instrument suite. The basic objectives of the initial mission are to conduct a global survey of the Moon, calibrate instruments, and gain experience with the spacecraft.

Since the platform can mount various remote-sensing instruments, many types of follow-on missions can be contemplated. Conceivably, these will be more ambitious and complicated. When such missions are desired, a lunar-based servicing vehicle provides a means for changing instrument systems and spacecraft subsystems. This operation is illustrated in Fig. 1. The LOP propulsion module can also be replaced by a refueled version, thus renewing the platform's ability to change orbits. This ability to service the platform in lunar orbit allows a great deal of mission flexibility in support of new exploration and science needs, and is vital to the utility and usability of the LOP.

The general configuration of the LOP is driven by three major requirements: overall system modularity and expandability, onboard propulsion, and a preferential nadir-pointing instrument platform. The system configuration is shown in Fig. 2.

The overall configuration objective is to allow the system to grow and adapt to new and different science and exploration needs. The base structure provides 24 mounting ports on the sides of the spacecraft to support power, communications, attitude control, and minor instrument subsystems. The ports provide the necessary power and communications utilities to support these subsystems. The propulsion module is located on the antinadir end of the LOP; the primary remote-sensing instruments are mounted opposite the propulsion module in the preferential nadir-pointing direction. This latter position gives the instruments required nadir viewing while providing unrestricted expansion away from undesired spacecraft thermal and magnetic interference.

Communications are provided by four phased-array, medium-gain antennae mounted on the sides of the spacecraft. These antennae are electronically steered to track relay satellites located

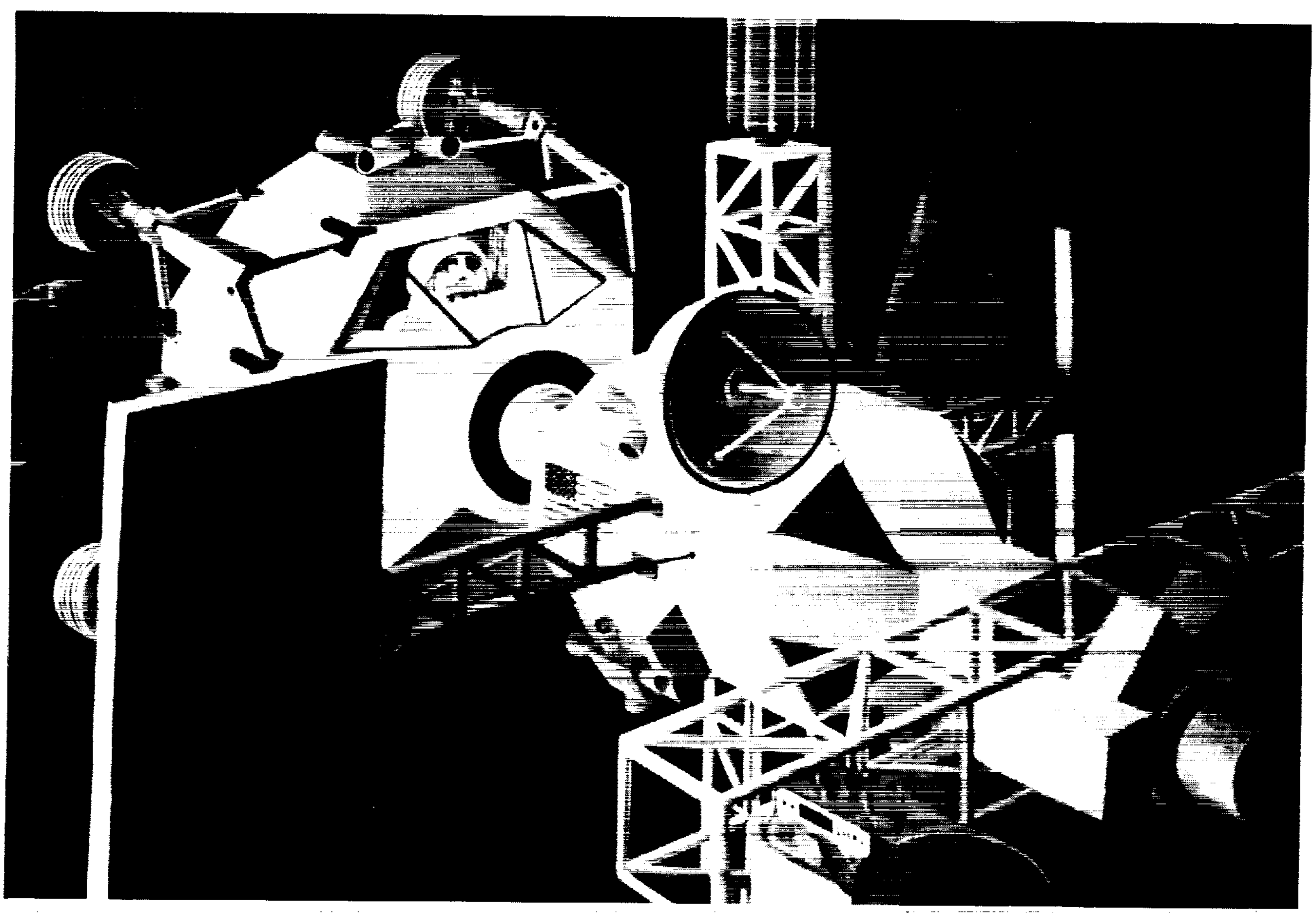

Fig. 1. To accomplish on-orbit servicing, the LOP relies on the use of a lunar based servicing vehicle. This illustration shows the servicing vehicle docking on the LOP's antinadir end. Any vehicle capable of on-orbit satellite servicing can perform this function; this eliminates the need for a specialized vehicle to perform this function.

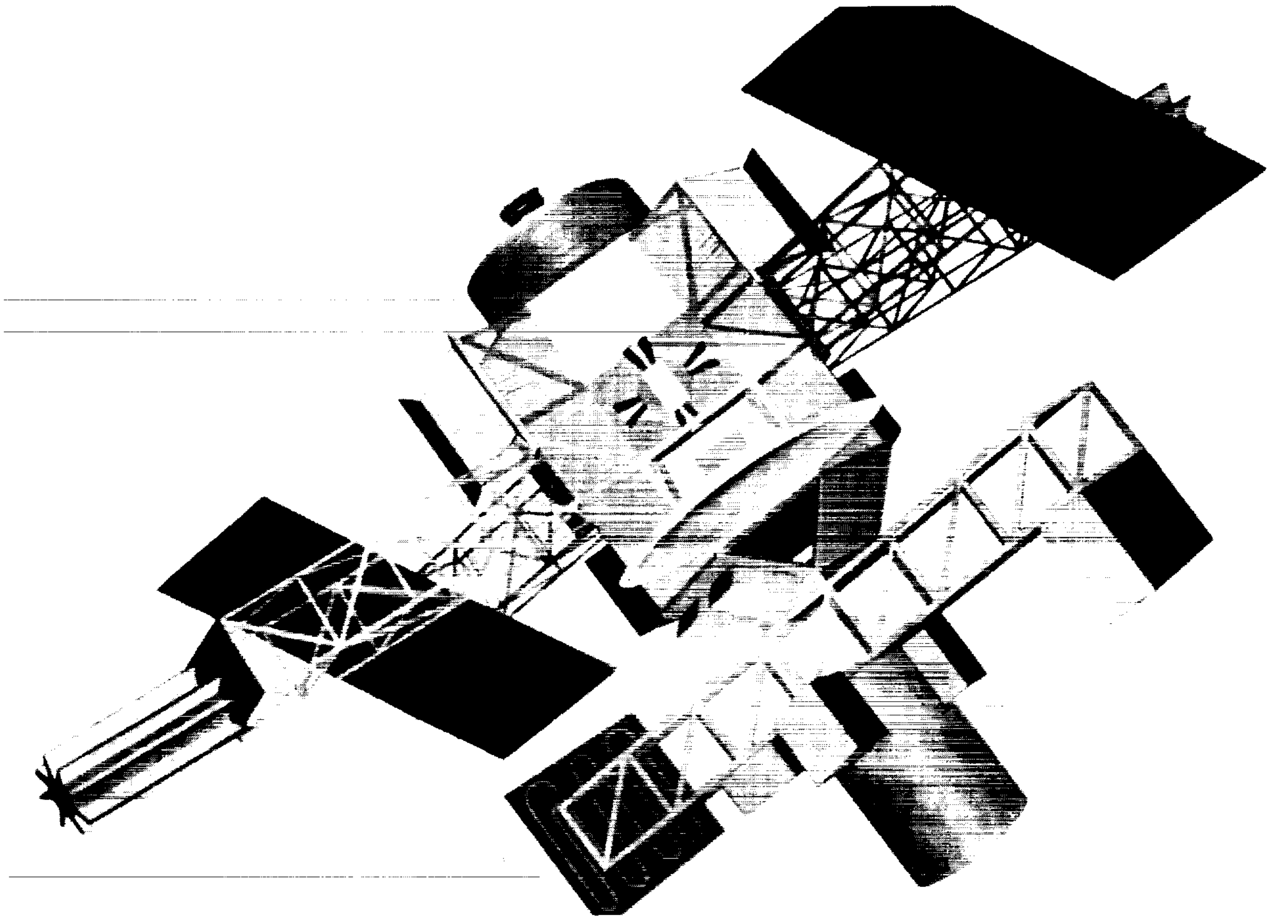

Fig. 2. The LOP consists of three major systems: the primary structure, the propulsion module, and the remote-sensing instrument package. The primary structure houses the propulsion module on its antinadir end. The remote-sensing instruments are mounted on the lower portion of the satellite in the nadir direction.

at LaGrange points L1 and L2. The relay satellites and their locations allow constant data transmission to either the lunar base or Earth. Present estimates indicate that a data rate of 10^9 bps will be attainable. If this rate is achieved, real-time transmission of a $1024 \times 1024 \times 8$-pixel image can be performed without data compression or storage.

The spacecraft is expected to pitch continuously to maintain nadir pointing, and yaw as required to position solar panels normal to incident sunlight. Attitude control is maintained by a bias momentum system consisting of four bias momentum wheels, hydrazine control thrusters, and pitch and roll attitude sensors. Analysis estimates that the average pitch pointing accuracy will be approximately ± 2 mrad. In the event of attitude control failure, the satellite is gravity-gradient stabilized.

The following sections will further amplify the unique characteristics of LOP remote-sensing packages, subystems, orbital strategies, and mission considerations. The remote-sensing discussion will focus upon instrument systems that are not included within the capabilities of the currently envisioned LGO. Readers interested in the LGO instrument suite are referred to "Contributions of a Lunar Geoscience Observer" (*LGO Science Workshop*, 1986). The use of a propulsion system designed to use lunar-derived propellants is also emphasized.

REMOTE-SENSING INSTRUMENTS

Two sensor systems are discussed: the Remote Raman Spectrometer (RRS) and the Radar Subsurface Mapper (RSM). Neither of these instruments can be flown on the LGO due to size and mass constraints and the priority of other sensor payloads. However, both the RRS and the RSM can greatly enhance our understanding of the Moon.

REMOTE RAMAN SPECTROSCOPY

The RRS is an active remote-sensing instrument; it uses a laser to stimulate raman emissions at the lunar surface, which are received by an onboard detector system. This instrument is more sensitive than the LGO's present instruments and it yields much higher spatial resolution. It is expected to detect more mineral and chemical species.

The primary advantages of using raman spectroscopy in lunar remote sensing over traditional methods such as reflectance and gamma-ray spectroscopy are (1) the raman spectra of most substances are less ambiguous than corresponding reflectance spectra; (2) the obtainable spatial resolution is better; (3) smaller

concentrations of mineral and chemical species can be detected; (4) an estimate of concentrations is obtainable; and (5) raman spectroscopy can be used to study additional features of the molecule beyond composition and volumetric concentration.

Raman scattering has been used extensively in laboratory applications to study molecular and crystal structures of substances. Raman spectroscopy has also been applied to the analysis of extraterrestrial materials such as the Apollo Moon rocks (*Karr,* 1975). The application of raman spectroscopy to remote sensing has been very limited, however. This is due mostly to the very weak nature of raman scattering; the raman spectra intensity is approximately six orders of magnitude smaller than the incident light intensity. The return of raman scattering is also further weakened by the presence of an atmosphere that scatters the returning signal. Despite this, some atmospheric pollution studies have been performed using remote raman spectroscopy and have met with limited success (*Freeman,* 1974).

Recent advances in laser, detector, and filter technology will allow the raman scattering principle to be more successfully applied to remote-sensing applications. This is particularly true on the Moon, where the lack of an atmosphere and its cold surface temperatures are favorable environmental conditions.

Principles of Raman Spectroscopy

Incident light can interact with a substance by either absorption or scattering. This interaction can be thought of as photons colliding with molecules. Absorption occurs when a molecule completely absorbs the photon's energy and no photons are reemitted. Scattering occurs when a module absorbs a photon's energy and re-emits photons at the same or different energy. When there is no change in photon energy, this is known as rayleigh scattering. When photons are re-emitted at a different energy, this is known as raman scattering. The change in the photon's energy gives rise to a change in its wavelength and frequency.

The distinguishing characteristic of the raman effect is the shift in frequency that occurs between the exciting light energy and the re-emitted light energy. This frequency difference, called the raman shift, is directly characteristic of the molecule and is independent of the incident light frequency. Figure 3 illustrates laboratory raman spectra of two lunar materials. The high intensity spikes represent raman lines characteristic of the lunar materials. These lines are shifted from the incident laser frequency, and they correspond to photons that are reflected back at energies different than the laser light.

Raman scattering can be further understood by examining the energy changes that take place in the molecule. When raman scattering occurs, some of the molecules in the substance return to a different energy level than they had before the collision with the photons. Since the molecules have either gained or lost net energy during the collison, the photons will be emitted with a corresponding loss or gain in energy. The molecular energy transition E_2-E_1 can be related to the photon's energy change by

$$E_2 - E_1 = -h\delta\upsilon \quad (1)$$

where h is Planck's constant, and $\delta\upsilon$ is the photon's shift in frequency. Molecules that pass to a lower energy state scatter photons at a higher frequency; this gives rise to what is known as antistokes lines. Molecules that change to a higher energy state give rise to a decreased photon frequency called a stokes line.

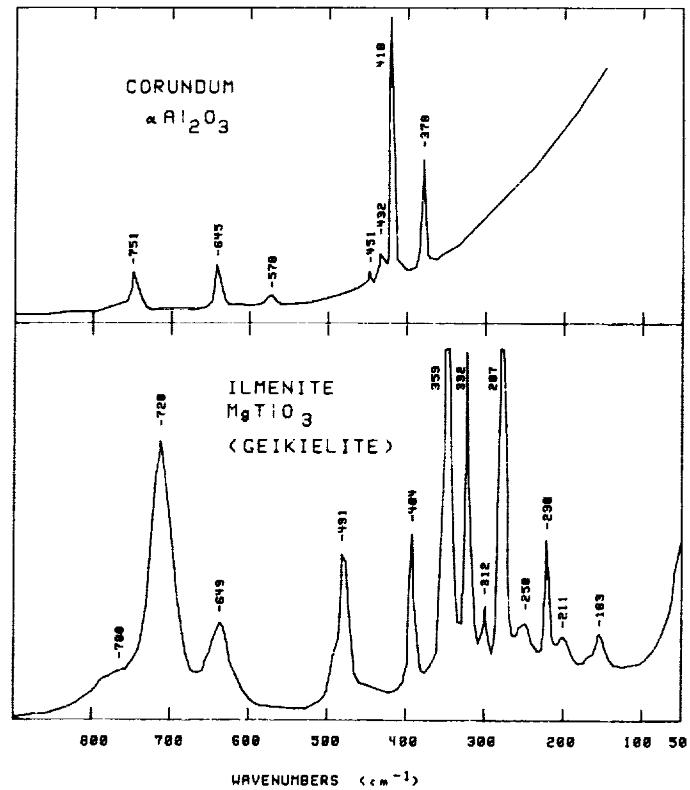

Fig. 3. The raman spectra of two lunar materials, corundum and ilmenite, are shown. The high-intensity spikes represent characteristic raman lines. The exact location of these lines is determined by the individual substances. The data are from *Karr* (1975).

The stokes and antistokes lines are, in most cases, mirror images of each other and contain the same information. For substances at cold temperatures, however, the stokes lines are more intense than the antistokes lines. It is for this reason that stokes lines are most often studied.

The net change in molecular energy is a quantum effect and is a function of the polarizability of the molecule. The molecular energy transitions that occur in raman scattering are discrete; the photon frequency shifts are accordingly discrete. Stokes and antistokes lines are thus directly characteristic of the molecule responsible for the scattering and can be used to identify the substance.

Since the raman effect is, in principle, an emission phenomenon, raman line intensities are proportional to elemental concentrations. This is a valuable attribute that will enable the determination of concentrations of mineral and chemical species.

Instrumentation

The RRS configuration is shown in Fig. 4. A laser is used to stimulate raman emissions at the lunar surface, and the returning energy is focused on a spectrally sensitive detector array. A filter is required to remove the laser frequency from the returning signal to avoid overwhelming the low intensity raman spectra.

The primary difficulty with using raman scattering as a remote-sensing method is the low intensity of the returned signal. Roughly 10^{-6} of the incident laser energy is returned as raman scattering; this is due to a very small fraction of the substance's molecular population that will alter energy states.

The ability of the detector system to read weak raman signals is an important consideration. Figure 5 plots the returned signal intensity at the spacecraft for differing altitudes and laser outputs. Raman emissions are considered to be diffuse at the lunar surface, and the returning signal is diminished by $1/\text{altitude}^2$. Present estimates indicate that a combination of a 10-W laser, an imaging spectrometer similar to that being developed for HIRIS, and a 157-mm telescope objective will permit the detection of raman spectra at altitudes up to 50 km.

The wavelength of the laser is an important quality. If one chooses the proper excitation frequency, many instrument problems can be avoided. The intensity of the raman effect is proportional to the fourth power of the exciting frequency, thus lasers of a higher frequency will improve the detectability of returning raman spectra. The frequency of the laser, however, is limited by problems with mineral fluorescence. Lasers of higher frequencies, such as ultraviolet, are likely to cause fluorescence at the sample and this will interfere with detecting the raman spectra. A krypton laser, whose frequency is in the visible wavelength region, is an appropriate compromise. This laser uses available technology and is of a high enough frequency to maximize the raman intensity but low enough to avoid problems with mineral fluorescence.

Using a visible wavelength laser requires the use of a visible wavelength imaging spectrometer. The use of visible wavelengths has two important benefits: (1) cryogenic instrument cooling is not required and (2) highly sensitive detectors are being developed for this wavelength region. The use of visible wavelengths, however, imposes constraints on the times when remote sensing can be performed. The RRS is limited to sensing on the dark side of the Moon where visible light pollution is not sufficient to interfere with the raman spectra.

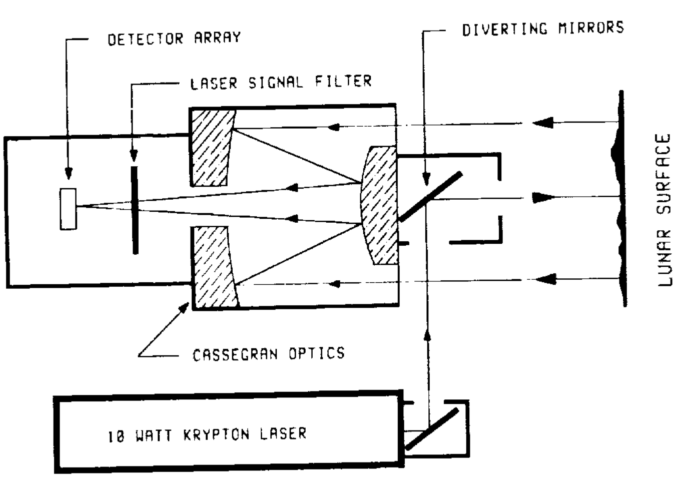

Fig. 4. The Remote Raman Spectrometer configuration. A laser is directed at the lunar surface and returning raman spectra are focused on an imaging array. The laser signal is removed by a mask filter.

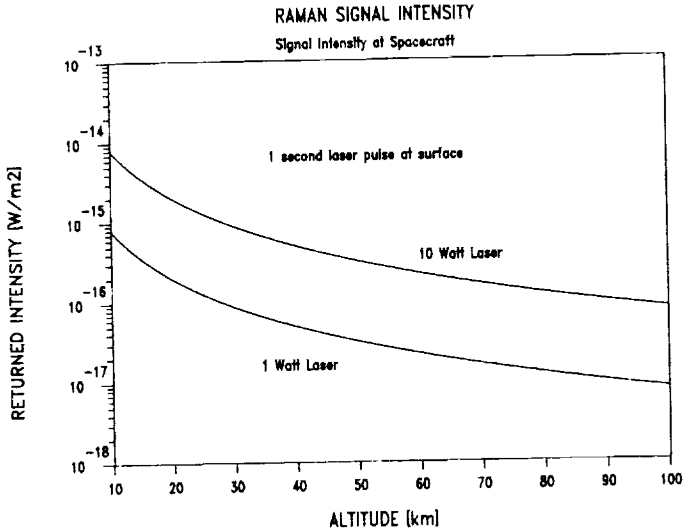

Fig. 5. Raman signal intensity at the spacecraft vs. altitude and laser power.

Spatial Resolution

The spatial resolution of the RRS is determined primarily by the size of the laser footprint on the lunar surface. The laser footprint at a 50-km orbital altitude is estimated to be 30 m in diameter; this corresponds to the surface resolution if the instrument is slewed to compensate for orbital velocity. If less resolution is desired, the slewing can be impeded. The laser pulse time and the orbital velocity will then combine to form an elongated footprint that will decrease the net surface resolution. The ability to alter spatial resolution enables either high-resolution site-specific coverage or lower-resolution coverage of larger regions.

RADAR SUBSURFACE MAPPER

In order to fully understand the subsurface attributes of the Moon, an orbital-based instrument capable of examining this region is needed. With such an instrument, subsurface layers can be traced and mapped; this will enable a better understanding of the local geology and will help locate lunar ores that might exist under the surface of the Moon.

The absence of moisture in the Moon's outer layers indicates that radio frequency electromagnetic energy will penetrate much deeper than is possible on the Earth. The use of radar signals pulsed at the lunar surface can explore the subsurface at depths of at least tens of meters (*Barrington,* 1986). Radar sounders carried by Apollo 17 penetrated as much as 1.5 km below the lunar surface and mapped subsurface layering in the Mare Serenitatis region (*Peeples et al.,* 1978).

This simple technique pulses radar energy at the lunar surface and detects the returned signals. A short pulse of electromagnetic energy is propagated into the lunar subsurface and is reflected by geologic interfaces. The ability of the radar signal to propagate is dependent on the electrical properties of the subsurface; in order for radar reflection to occur, an electrical property discontinuity must exist across the geologic interface.

Radar sounding can be used for both surface imaging and subsurface profiling. Profiling is accomplished by maintaining a time reference between the radar transmission and the returned signals; a longer time between the radar pulse and its return is associated with greater depths. A composite from different depths constitutes a full subsurface profile. Imaging is accomplished by recording the diffuse radar backscattering at the surface. The combination of imaging and depth profiling will be highly useful for the interpretation of surface geology.

Instrumentation

The RSM consists of three primary elements: a Coherent Synthetic Aperture Radar (CSAR) that contains the transmitting and receiving elements, and two separate antennae for imaging and profiling. For imaging and shallow profiling, an operating frequency of 150 Mhz is used; for deep profiling (10 m-1 km), an operating frequency of 15 MHz is used.

ORBITS

The Moon's low mass (about one eightieth that of the Earth) and lack of an atmosphere make it a very attractive body for remote-sensing orbiters. Low-energy orbits associated with the low lunar mass require substantially less propellant for orbit maintenance and alteration than comparable Earth orbits. The lack of atmospheric drag permits orbital altitudes that are limited only by topography and lunar gravitational perturbations.

Orbit/Instrument Synergy

The platform's ability to change orbital parameters is important to achieving desired mission versatility. Optimization of the relationship between surface coverage and instrument resolution requires a variety of orbits. Variation of orbital altitude, eccentricity, and inclination will permit tailoring to meet specific mission needs and instrument requirements. Several classes of orbital coverage are considered: global surface coverage at moderate to high resolution, and site-specific coverage at very high resolution. The parameters of the lunar orbit (inclination, eccentricity, and altitude) will dictate the type of surface coverage and resolution that can be realized.

Orbital inclination strongly influences the quality of regional surface coverage. Figure 6 illustrates the effect of various orbital inclinations on local coverage. In general, viewing is most complete at the lunar latitudes equal to and less than the orbit inclination angle. Viewing of the entire lunar surface is offered only by polar orbits, but this advantage is offset by sparse coverage at low-latitude regions where successive orbital tracks are far apart. Reducing orbital inclination from 90° improves the coverage at low latitudes and equatorial regions, but this completely eliminates polar coverage. Low-inclination orbits are consequently best suited to high-resolution, site-specific coverage of low-latitude regions; polar orbits are best suited to lower-resolution global coverage, and high-resolution coverage of the polar regions.

Orbital altitude strongly influences instrument resolution. In general, high altitudes decrease optical resolution and increase instrument dwell time, while low altitudes increase resolution and decrease dwell time. The higher orbital velocities associated with low-altitude orbits decrease the amount of time available for instruments to sense the surface; however, for reasonably small altitude differences, dwell time is not a significant factor since lunar orbits exhibit relatively small changes in orbital velocity with changes in orbital altitude. Thus, orbit altitude selection is primarily a function of required instrument resolution, viewing needs, and concerns for orbital stability.

Orbital Stability

Mission objectives require orbits stable enough to permit orbital maintenance with a reasonable amount of maneuvering but low enough for good instrument resolution. The Moon's gravity field directly affects the stability of lunar orbits, depending on their altitude and inclination. Experience from earlier lunar flights and known lunar gravitational harmonics indicates that polar orbital inclinations and low-altitude orbits are unstable. Precise polar orbits exhibit a rise in eccentricity leading to periapsis (closest approach to the surface) lowering and eventual collision with the lunar surface. This phenomenon is shown in Fig. 7 (*Chesley et al.,* 1988). The inability to precisely assess the effects of known large

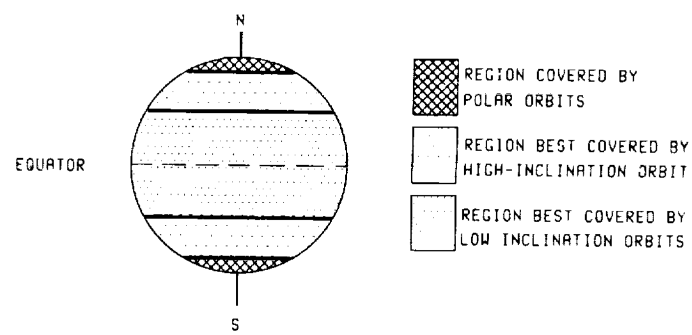

Fig. 6. Orbit inclination vs. optimum regional coverage.

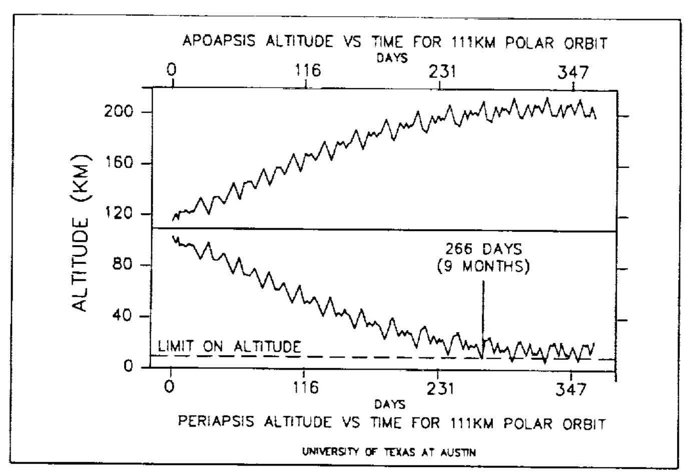

Fig. 7. Results from a numerical simulation of the effects of lunar gravitational pertubations on orbital stability. Data are from *Chesley et al.* (1988). The upper curve represents the altitude of orbit apoapsis and the lower curve represent the orbital periapsis altitude. The results indicate that a polar orbit will experience a rapid rise in eccentricity, and the result will be the spacecraft impacting the lunar surface after 266 days.

lunar gravitational anomalies on the stability of low-altitude orbits leaves a degree of uncertainty in prediction models. Numerical simulations of lunar orbits, based on low-order spherical harmonics, suggest a pattern of rising eccentricity on polar orbits (*Chesley et al.,* 1988) and on oscillating eccentricity on equatorial orbits (*Bond and Mulciby,* 1988). Present estimates of orbit lifetime (before surface collision) for a 100-km polar orbit are approximately 9 months.

The net effect of orbital instabilities is to increase propellant usage for orbital maintenance. General avoidance of very low altitudes and certain inclinations can partially negate stability problems. Estimates of lunar gravity harmonics indicate that stable near-polar orbits may be available. These slightly eccentric, so-called "frozen" orbits (*Burke,* 1976) have their periapses fixed near the Moon's south pole and may remain stable for up to two years (*Upboff,* 1976).

Orbital Strategies

Several surface-coverage strategies are made possible through variation of the orbital altitude (e.g., large regions at low resolution, small regions at high resolution, and global coverage at moderate to high resolution). Global coverage at moderate resolution can be achieved through the use of a 100-km near-polar orbit such as that planned with the LGO mission. Higher altitudes reduce the resolution obtained but increase the instruments' field of view, thus allowing the equatorial regions to be more adequately viewed. Complete surface coverage from a polar orbit can be obtained in 27 days.

A highly elliptical orbit with a low periapsis over the region of interest gives a high-resolution sensing opportunity without the stability problems associated with low circular orbits. Since a small fraction of the orbit is spent at low altitudes, the net effect is to provide a low-altitude sensing opportunity while maintaining a higher average orbital altitude. The drawback to this method is the small region of coverage that accumulates under the periapsis. The highly eccentric orbit is also capable of giving low-resolution, large regional coverage on the apoapsis (maximum distance from the surface) side with a smaller required velocity change than is associated with an equivalent transfer to a high circular orbit. The elliptical orbit offers a variety of surface-coverage opportunities through the proper choices of periapsis latitude and orbit eccentricity.

A possibility exists for high-resolution global coverage by using a highly eccentric orbit and periodically shifting the latitude of periapsis. Figure 8 illustrates this concept. Since the orbital precession is small, the orbit can be considered to be fixed in space while the Moon rotates under it. By rotating the orbit's line of apsides (the line connecting the orbit's periapsis and its apoapsis) at appropriate intervals, the entire lunar surface can be viewed from a low altitude without severe orbital stability problems.

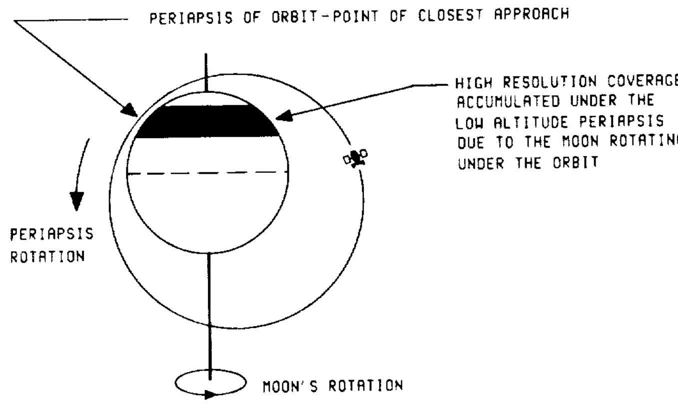

Fig. 8. High-resolution coverage of large regions can be accomplished by periodically rotating the latitude of periapsis. This will yield an orbit more stable than an equivalent low-altitude circular obit.

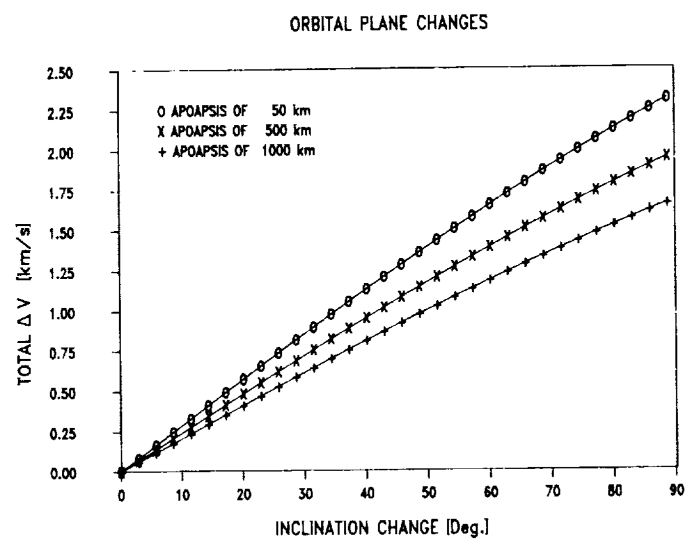

Fig. 9. Delta V requirements for orbit inclination changes.

Required Velocity Changes

Orbital plane changes are most efficiently accomplished at the apoapsis where the orbital velocity and, thus, the required velocity change are lowest. Figure 9 displays the velocity changes required to achieve inclination changes for orbits of varying apoapses. Figure 10, in turn, displays the propellant mass required to achieve a given velocity change. The designed platform propellant capacity will enable a maximum velocity increment of 1.5 km/sec. From Fig. 9 it is seen that this capacity allows orbital changes exceeding 50° for most orbits with changes of 90° possible for highly elliptical or high-altitude orbits.

Velocity changes required for coplanar orbit changes are shown in Fig. 11. The calculations assume an initial 25-km circular orbit with minimum energy transfer to the final circular or elliptical configuration. As shown, transfer from a 25-km circular orbit to a 1000-km circular orbit requires a relatively small 320-m/sec velocity increment, and transfer to an eccentric orbit with a 100-km apoapsis requires a velocity increment of only 180 m/sec. From Fig. 10 it is seen that these velocity changes translate into propellant burns of approximately 100 kg and 55 kg, 100 kg and 40 kg, respectively. With such small maneuvering requirements, many platform orbital changes and adjustments are possible in one mission.

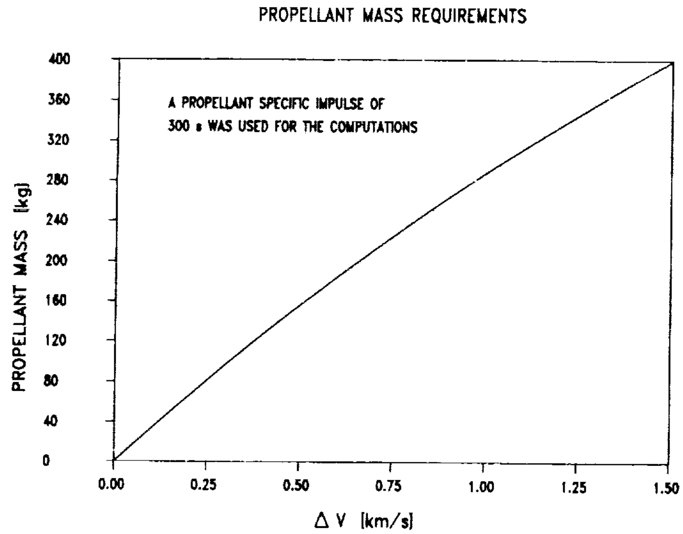

Fig. 10. Propellant mass requirements vs. ΔV.

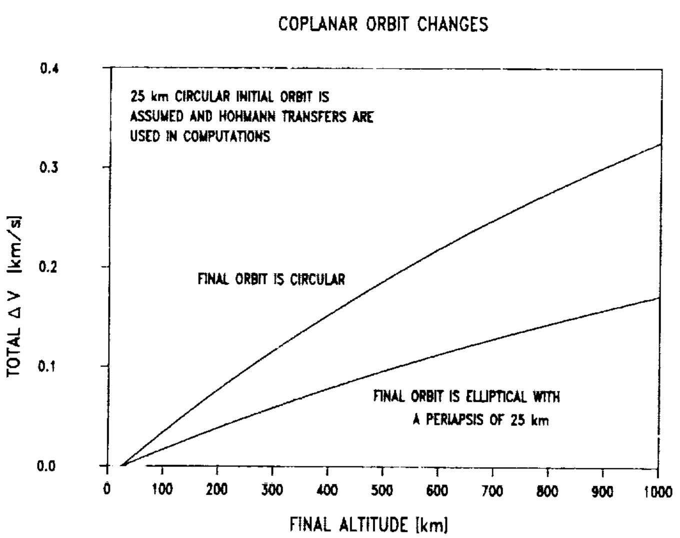

Fig. 11. Delta V requirements for coplanar orbit manuevers.

Rotation of the line of apsides can be accomplished in two ways. The first method involves a three-step maneuver that first circularizes the eccentric orbit at the apoapsis, then proceeds to the new line of apsides, and finally reduces the velocity to acquire the new periapsis. The second method causes periapses rotation by rotating the velocity vector at apoapsis, then correcting the eccentricity. The comparative efficiency of the two methods depends upon the magnitude of the desired rotation. In general, the first method would seem most efficient since it requires the same propellant expenditure regardless of the rotation magnitude.

PROPULSION

The versatility allowed by the LOP orbit-changing capability requires considerable propellant usage. This usage represents one of the principle expenses associated with platform operation. Propellant costs can be substantially reduced by using lunar-derived fuels that do not require costly transport out of the Earth's gravity well. Use of lunar-derived fuels also helps to minimize Earth support requirements.

Lunar Propellant Sources

Lunar minerals that show the most promise for lunar propellant production are olivine [$(Mg,Fe)_2SiO_4$], pyroxene (a family of silicates rich in Ca, Fe, and Mg), and ilmenite ($FeTiO_3$). Olivine and pyroxene occur in concentrations up to 60% and ilmenite in concentrations up to 20% (*Williams and Jadwick,* 1980). These minerals are particularly promising for oxygen production. Selected elemental concentrations from Apollo samples are shown in Table 1.

Inasmuch as oxygen is abundant on the Moon, this element is an obvious choice for a bipropellant oxidizer. Extraction from lunar minerals by chemical and thermal processes is already planned for lunar colonization activities. The critical constituent of a lunar-derived propellent, then, is the fuel. Hydrogen is an excellent fuel when used with oxygen, but its lunar concentrations are extremely small. Other lunar-derived fuel possibilities include, but are not limited to, silane (SiH_4), AlCa, Al, AlCaMg, Ca, and lunar soil.

Silane has been theorized to perform well but has never been used (*Rosenberg,* 1986). Its primary advantages are its lunar-derived silicon, its high specific impulse (360/sec), its thermal stability for regenerative cooling, and its high boiling temperature. Silane production from lunar materials, however, would be quite slow due to its reliance on MgO in the production process (*Rosenberg,* 1985).

A comparison of lunar-derived propellants normalized to the aluminum/liquid oxygen (LOX) combination is shown in Table 2. The high performance offered by liquid H_2/O_2 is really unavailable due to its extremely low availability on the Moon. Silane performs well, but its long production time plus its reliance upon terrestrially supplied hydrochloric acid in the production process

TABLE 1. Elemental concentrations of Apollo samples.

Element	Concentration (wt%) Mare	Highlands
Oxygen	41.7	44.6
Silicon	21.2	21.0
Aluminum	6.9	13.3
Calcium	7.8	10.7
Magnesium	5.8	4.5
Hydrogen	54 ppm	56 ppm

Data are from *Williams and Jadwick* (1980).

TABLE 2. A comparison of lunar-derived propellants with quantities normalized to 1 for Al/LOX.

	Al/LOX	LH_2/LOX	$LSiH_4$/LOX
Oxidizer Mass	1	0.70	0.77
Oxidizer Volume	1	0.70	0.78
Fuel Mass	1	0.78	1.12
Fuel Volume	1	29.84	4.47
Production Time	1	10.86	9.57

negate its performance advantage. Aluminum and oxygen thus show the most promise for lunar-derived propellants (*Utah State University*, 1988). Powdered aluminum, when used in combination with an Earth-based binder and lunar-derived oxygen, can provide up to 300 sec of specific impulse (*Streetman*, 1978).

Thruster Design

The thruster is a hybrid design using solid Al and LOX. A refillable oxygen tank is mounted in the center of the spacecraft where it is insulated and shaded to minimize boil-off. The solid fuel portion of the thruster mounts on the antinadir end of the LOP. The solid fuel, casing, injector, and nozzle are replaced with a refueled version on orbit. The nozzle and combustion chamber are regeneratively cooled for reusability.

CONCLUSIONS

Man's return to the Moon will be characterized by a commitment to a permanent presence designed to exploit a body rich in natural resources, to find answers to a host of intriguing scientific questions, and to establish a base for further exploration of the solar system. The early orbital exploration mission of the LGO will most certainly be followed by longer-term, more versatile orbiting systems that operate in a synergistic manner with human exploration efforts. The LOP provides a preliminary example of such a system. Designed-in modularity and orbital mobility allow flexible operation adaptable to many instrument/mission combinations. Provision for the replacement of instrument clusters and subsystem modules permits expansion to incorporate new technologies and evolution to extend mission capabilities. The LOP thus projects a broad range of long-term operational possibilities for continued lunar exploration well into the twenty-first century.

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REFERENCES

Barrington A. R. (1986) Radio frequency reflectivity experiments for AES lunar orbital flights. *Adv. Astronaut. Sci., 20,* 418-433.

Bond V. and Mulcihy D. (1988) Computation of seleocentric orbits using total energy (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 29. Lunar and Planetary Institute, Houston.

Burke J. D. (1976) Lunar polar orbiter: A global survey of the Moon. *Acta Astronaut., 4,* 907-920.

Chesley B., Korsmeyer D., and Monroe D. (1988) Mixed vehicle transportation fleet for missions to lunar polar orbit (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 50. Lunar and Planetary Institute, Houston.

Freeman S. K. (1974) *Applications of Laser Raman Spectroscopy.* Wiley, New York.

Hood L. L., Sonnet C. P., and Russell C. T. (1985) The next generation geophysical investigation of the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 253-263. Lunar and Planetary Institute, Houston.

Karr C. Jr. (1975) *Infrared and Raman Spectroscopy of Lunar and Terrestrial Minerals.* Academic, New York.

LGO Science Workshop (1986) *Contributions of a Lunar Geoscience Observer.* Dept. of Geological Sciences and the Anthro-Graphics Laboratory, Southern Methodist Univ., Dallas.

Peeples P. J., Sill W. R., May T. W., Ward S. H., Phillips R. J., Jordon R. L., Abbott E. A., and Kilpack T. J. (1978) Orbital evidence for lunar subsurface layering in Maria Serenitatis and Crisium. *J. Geophys. Res., 83,* 3459-3468.

Rosenberg S. D. (1985) A lunar based propulsion system. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 169-175. Lunar and Planetary Institute, Houston.

Streetman J. W. (1978) Investigation of the feasibility of chemical rockets using lunar derived fuels. In *Proc. 14th Joint Propulsion Conference.* American Institute of Aeronautics and Astronautics, Washington, DC.

Uphoff C. (1976) Stabilizing influence of Earth's perturbations on polar lunar orbiters. *AIAA/AAS Astrodynamics Conf.* American Institute of Aeronautics and Astronautics, Washington, DC.

Utah State University (1988) *Lunar Orbital Prospector.* Final report to NASA/USRA/ADP, Houston.

Vaniman D. T. and Heiken G. (1985) A closer look at lunar volcanism from a base on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 211-222. Lunar and Planetary Institute, Houston.

Watson K., Murray B. C., and Brown H. (1961) The behavior of volatiles on the lunar surface. *J. Geophys. Res., 66,* pp. 3033-3045.

Williams R. J. and Jadwick J. J. (1980) *Handbook of Lunar Materials.* NASA RP-1057. 120 pp.

LUNAR OBSERVER LASER ALTIMETER OBSERVATIONS FOR LUNAR BASE SITE SELECTION

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One of the critical datasets for optimal selection of future lunar landing sites is local- to regional-scale topography. Lunar base site selection will require such data for both engineering and scientific operations purposes. The Lunar Geoscience Orbiter or Lunar Observer is the ideal precursory science mission from which to obtain this required information. We suggest that a simple laser altimeter instrument could be employed to measure local-scale slopes, heights, and depths of lunar surface features important to lunar base planning and design. For this reason, we have designed and are currently constructing a breadboard of a Lunar Observer Laser Altimeter (LOLA) instrument capable of acquiring contiguous-footprint topographic profiles with both 30-m and 300-m along-track resolution. This instrument meets all the severe weight, power, size, and data rate limitations imposed by Observer-class spacecraft. In addition, LOLA would be capable of measuring the within-footprint vertical roughness of the lunar surface, and the 1.06-μm relative surface reflectivity at normal incidence. We have used airborne laser altimeter data for a few representative lunar analog landforms to simulate and analyze LOLA performance in a 100-km lunar orbit. We demonstrate that this system in its highest resolution mode (30-m diameter footprints) would quantify the topography of all but the very smallest lunar landforms. At its global mapping resolution (300-m diameter footprints), LOLA would establish the topographic context for lunar landing site selection by providing the basis for constructing a 1-2 km spatial resolution global, geodetic topographic grid that would contain a high density of observations (e.g., ~1000 observations per each 1° by 1° cell at the lunar equator). The high spatial and vertical resolution measurements made with a LOLA-class instrument on a precursory Lunar Observer would be highly synergistic with high-resolution imaging datasets, and will allow for direct quantification of critical slopes, heights, and depths of features visible in images of potential lunar base sites.

INTRODUCTION AND BACKGROUND

Many of the scientific and engineering issues associated with the selection of potential lunar base sites or any future lunar landing sites require a detailed knowledge of local and regional-scale topography (*Wilhelms,* 1985). Prior to the human Apollo missions to the Moon, extremely high resolution Lunar Orbiter photographs were acquired in stereo in order to assess the local topography and geology of candidate human landing sites. Such data were especially critical for those missions involving the Lunar Rover vehicle (i.e., Apollo 15-17), as the rover could only negotiate terrain with slopes less than several degrees. In an important study of local-scale lunar roughness and slopes, H. J. Moore et al. (unpublished data, 1969) outlined the kinds of local terrain data necessary for a realistic and accurate assessment of potential implications of lunar topography for human operations. *Moore et al.* (1980) used a combination of orbital stereo photography, bistatic radar, and Earth-based radar to assess lunar RMS roughnesses indirectly, and then compared the independent results with reasonable agreement. However, a direct means of globally measuring local lunar surface roughness and slopes was technologically not feasible in the early 1970s.

The Apollo Laser Altimeter (flown on Apollo 15-17) obtained surface elevation data for 3-m footprints at 1-2 m vertical accuracy; however, the limited lifetime and low pulse repetition frequency (PRF) of the ruby laser allowed only one range measurement for every 30 km along the suborbital track of the Command Service Module (*Kaula et al.,* 1974). Thus, the circumlunar topographic profiles that were acquired have a very low spatial resolution and can only be used to measure slopes on baselines of 60 km and longer.

Lunar Topographic Orthophotomaps (LTOs) have been constructed for many areas on the Moon, and these high-quality stereophotogrammetric topographic data have effective spatial resolutions as good as a few hundred meters with ~10-m relief contours (*Ravine and Grieve,* 1986). However, the LTO data do not form a global, high-integrity, topographic model for the entire lunar surface, or even for a large fraction thereof.

Orbital radar techniques during the Apollo era (i.e., the ALSE experiment flown on Apollo 17) obtained kilometer-resolution profiles of lunar topography (*Moore et al.,* 1980; *Sharpton and Head,* 1982), as have Earth-based radar techniques. A global lunar topographic model with 2-5-km grid cells (spatial resolution) could be obtained using existing narrow-beam radar altimeter techniques (*Phillips,* 1986) from a lunar polar orbiter. Such a dataset would be invaluable for establishing an accurate control net relevant to lunar landings and lunar base site selection, as well as for long-wavelength geophysical studies and for determining regional-scale topographic characteristics of all major terrain types on the Moon. Without resorting to very large antennae, high-frequency radar altimeter designs such as those currently under development for the Earth Observing System, local-scale high-resolution (<0.5 km) topography of the Moon can only be directly assessed from high-repetition-rate orbital laser altimetry. Technological breakthroughs in laser lifetime now permit pulsed laser altimeter instruments to operate continuously for a complete

Lunar Geoscience orbiter mission lifetime. On the basis of the inherent simplicity of the basic design of laser altimeter instruments (Fig. 1), their reliability in a 100-km lunar orbit poses no major technological challenges at this time, although long-lifetime lasers must still be space qualified. The recent tentative approval of an orbital laser altimeter experiment for the Mars Observer spacecraft and the experience being gained in constructing this instrument for long-lived operations under the more severe martian conditions (e.g., 360-km orbital altitude, dusty atmosphere with clouds, global dust storms, a 687-day operation lifetime requirement, and a dynamic range of topography of ~36 km) gives us further confidence that orbital laser altimetery is technically feasible for a lunar scenario (*Smith et al.,* 1989). The remainder of this report outlines the design and performance capabilities of a simple Lunar Observer Laser Altimeter (LOLA) instrument, and discusses performance simulations using airborne laser altimeter datasets for lunar analog surface features.

ORBITAL LASER ALTIMETRY

Altimeters are basic instruments that measure the time of flight of some type of electromagnetic signal from a platform (spacecraft, aircraft) to a target (lunar surface) in order to determine range, from which surface topography can be derived. Traditional orbital altimeters employ microwave signals (e.g., Seasat, Pioneer, Venus) in part because of their ability to effectively propagate through an atmosphere containing dust and clouds (e.g., Earth and Venus). The technology associated with radar altimetry is mature (*Phillips,* 1986), and such instruments are certainly suitable for global characterization of lunar topography at spatial scales of several kilometers. In order to achieve the kinds of height accuracies required for geophysical studies (i.e., better than 10-20 m), radar devices typically transmit a burst of 100 to 1000 pulses per second and then statistically determine the mean topography from each burst. The main reason for the statistical approach is to reduce individual pulse noise caused by the "coherent fading" due to the similarity between the scales of surface roughness elements, the radar wavelength (i.e., 8-30 cm), and the size of the radar transmitter antenna (i.e., 0.4-1.0 m in diameter). This statistical approach has proven to be very reliable (e.g., Seasat for Earth, Pioneer-Venus for Venus).

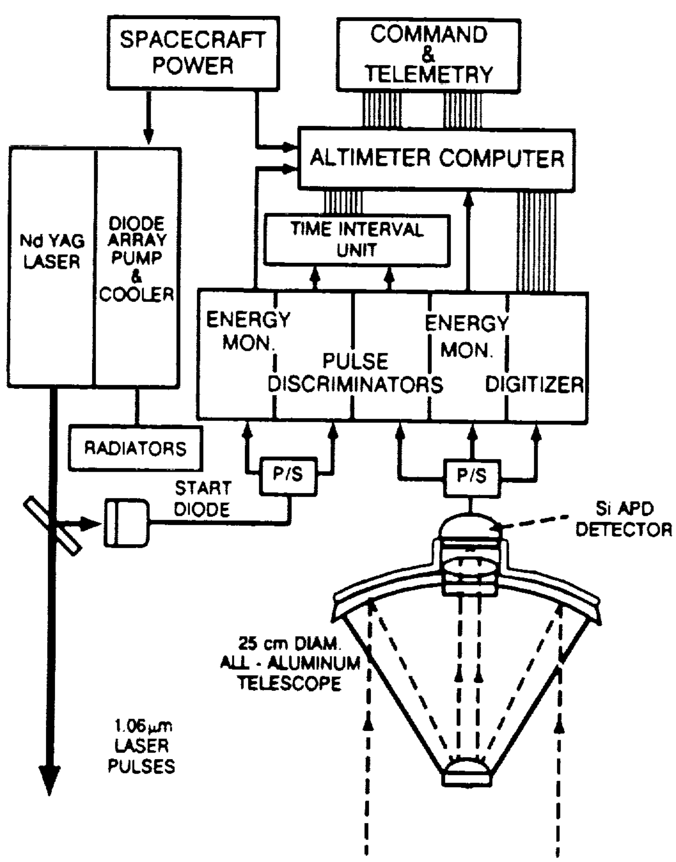


Fig. 1. Functional block diagram of the major components in the design of LOLA. The heart of the instrument is the diode-pumped Nd:YAG laser transmitter (at left) and the 25-cm-diameter telescope that serves as the receiver antenna. See text for details.

Unlike radar altimeters, laser ranging systems do not require extensive pulse averaging to achieve high vertical precision. This is because laser radiation at wavelengths near 1 μm does not produce coherent fading (i.e., speckle modulation) when received by means of a 25-50-cm-diameter telescope. In simple pulsed laser altimeter systems, a single laser pulse results in a unique range and hence topographic observation. By pulsing the laser as frequently as is needed to synthesize a contiguous-footprint profile on the surface (i.e., a profile in which the measurement footprints touch or overlap one another along track), a high-resolution cross section of surface topography along the nadir track of the spacecraft is generated. Therefore, laser altimeter systems are inherently simpler than radar devices on the basis of their nonstatistical approach to measuring topography.

Because of the small footprints required by laser altimeters in order for them to minimize signal loss due to extreme pulse spreading by a rough surface, spacecraft pointing knowledge is very important if meter-level-height accuracies are to be maintained. The heart of a simple orbital laser altimeter instrument is illustrated in Fig. 1. The laser transmitter emits pulses of laser radiation whose round-trip propagation time from the spacecraft to the surface and back is measured with a fast clock, which is known as a time interval counter, after having been received by a telescope and detected with an avalanche photodiode (APD); the shape of the backscattered laser pulse is measured with a waveform digitizer (in terms of power vs. time). Waveform digitization permits the complete interaction history of a single laser pulse with any surface to be quantified, and this information can be directly related to the vertical topographic variance within the footprint, and hence to roughness at 0.5-10-m spatial scales. Because of the intrinsic narrow beam width of optical laser radiation, laser altimeters naturally synthesize small footprints from orbital altitudues (e.g., 30-100 m from a 100-km orbit), and hence generate very narrow profiles of topography, in contrast with broader-beam radar systems that synthesize swaths several kilometers in width (*Bufton,* 1989).

In order to directly measure the topographic and slope characteristics of most lunar surface features of relevance in lunar base site selection and operations, an orbital laser altimeter must have a spatial resolution commensurate with high-resolution orbital imaging datasets (e.g., Lunar Orbiter, Apollo metric camera, etc.). For this reason, an along-track sampling interval of 30 m has been chosen for the LOLA system, although simple optics could be incorporated to decrease this value. For the 30-m-diameter footprints to provide contiguous sampling along track, the all-solid-state laser transmitter must operate with a pulse repetition frequency of 50 Hz, which is over 2 orders of magnitude more

rapid than the Apollo Laser Altimeter. Assuming a nominal 100-km orbital altitude, the 30-m-diameter footprint profiles will be separated across track by ~2 km at the lunar equator, but will rapidly converge at increasing latitudes. These high-resolution profiles would be circumlunar in their coverage. In order to adequately sample the regional topography of the Moon, a second spatial resolution mode is recommended for a LOLA instrument. Orbital simulations suggest that 300-m-diameter footprints could easily be achieved; this spatial resolution would diminish the effect of coverage gaps at the lunar equator by an order of magnitude and permit rapid acquisition of a dataset suitable for gridding at a 1-2-km level. The 300-m-diameter footprints could be contiguous or, if desirable, they could be overlapped by up to 50% to reduce sampling biases. If the LOLA instrument could continuously measure the lunar topography in this lower resolution mode for a period of ~1 year, nearly 60% of the lunar surface would be sampled, and a subkilometer topographic grid could be constructed.

Vertical resolution for either the 30-m or 300-m modes on LOLA is dictated by laser pulsewidth (i.e., the duration in time of the central part of the transmitted laser pulse), time-interval counter (TIC) resolution, and pointing control and knowledge. Table 1 summarizes the baseline design parameters that have been chosen for LOLA. The 3-nsec laser pulsewidth (full width at half maximum) coupled with a 1-nsec TIC resolution provides 15-cm vertical resolution under ideal conditions. This is because there are ~6.67 nsec per meter of relief (due to the speed of light), and the narrower in time a laser pulse can be made, the easier it is to track after it interacts with a random surface (and is naturally spread in time). The TIC is a very fast counter that is activated when each laser pulse is transmitted, and stopped once some critical threshold level on the backscattered laser pulse is detected. A 1-nsec TIC resolution ensures timing precision to 15 cm. However, final vertical accuracies will depend on local surface slopes, within footprint roughness, pointing knowledge, and orbit determination. Aircraft laser altimeter systems capable of this level of performance are now operating out of NASA's Goddard Space Flight Center.

Tables 1-4 summarize those design and performance parameters for LOLA that we feel should provide maximum information with respect to lunar geoscience objectives and lunar base site selection. Figure 1 illustrates the inherent simplicity of the LOLA design in a functional block diagram. It should be emphasized that LOLA requires no onboard signal analysis or complex pulse averaging to make a range measurement. In fact, every single transmitted laser pulse would result in a unique, independent range, and relative reflectivity measurement. The LOLA design under development as part of NASA's Planetary Instrument Definition and Development Program (PIDDP) provides three types of data relevant to lunar surface properties for each and every footprint: range (elevation), 1.06-μm relative reflectivity at normal incidence, and vertical RMS roughness (i.e., a measure of the total dynamic range of relief within each 30-m or 300-m footprint). The 1.06-μm reflectivity within 300-m-diameter footprints would be synergistic with Visual Infrared Mapping Spectrometer (VIMS) observations of lunar surface mineralogy (*Phillips,* 1986). The 30-m and 300-m LOLA footprint RMS vertical roughness would allow for after-the-fact retracking of LOLA topographic observations (in rougher terrains), and would be complementary to active or passive microwave observations of lunar roughness (e.g., from a multichannel microwave radiometer on LGO, or from Earth-based radar observations). Local

TABLE 1. LOLA laser altimeter instrument parameters.

Laser Transmitter	
Laser Type	Diode-pumped, Q-switched Nd:YAG
Wavelength	1.06 μm
Pulse Energy	2 mjoule
Pulsewidth	3 nsec FWHM
Repetition rate	10 Hz or 50 Hz
Divergence	0.3 or 3.0 mrad
Lifetime	10^9 shots minimum (3 yr @ 10 Hz)
Altimeter Receiver	
Telescope Type	f/1 diamond-turned aluminum parabola
Telescope Diameter	25 cm
Optical Filter	2 nm bandpass
Detector Type	Silicon Avalanche Protodiode
Quantum Efficiency	40%
Sensitivity	1 nwatt
Time-Interval Counter	1-nsec resolution
Waveform Digitizer	Pulse width, power, and energy

TABLE 2. LOLA payload parameters.

Size	25 cm × 25 cm × 35 cm (0.022 m^3)	
Weight	15 kg (33 lb)	
Operating Modes	Standard-rate (10Hz)	Burst (50Hz)
Power	10 W	15 W
Data Rate	1 kbps	3.5 kbps
Duty Cycle	Continuous	
Thermal	Passive control with insulation and radiators	
Mounting	Spacecraft bus	

TABLE 3. LOLA performance in 100-km lunar orbit.

Sensor Footprint	30 or 300 m
Along-Track Data Interval	Contiguous coverage
Vertical Resolution	15 cm
Vertical Accuracy	Submeter
Surface Roughness Resolution	Submeter
Surface Albedo Resolution	1% for 1.06 μm backscatter

TABLE 4. Typical standard deviations (SD) of topography within 30-m-diameter footprints (LOLA scale) for analog landforms relevant to planetary surfaces.

Target	Landform Type (and subtype)	SD Topography (m)
Meteor Crater	Impact crater	1-26
	Steepest inner wall	12-26
	Near-rim ejecta	2-6
	Rim	2-7.5
	Distal ejecta	0.2-2
	Floor	1-2
	Edge of ejecta	4-6
Grand Canyon	Erosional canyon (fluvial/tectonic)	1-72
	Walls of deepest canyon	60-72
	Typical canyon walls	35-60
	Canyon floors	3-8
Iceland lavas	Lava channel (pahoehoe)	0.2-2.0
	Flow margins	0.9-2.0
	Typical flow interior surface	0.2-0.3
	Roughest flow surface	0.3-0.5
Water*	Atlantic Ocean (Iceland bay)	0.15-0.3

* Control surface for reference with respect to 30-m baseline roughness.

topographic gradients on baselines as short as 60 m could be computed directly from LOLA profiles.

In order to obtain these complementary global datasets using a LOLA instrument in a 100-km lunar orbit, the fundamental engineering challenges are primarily related to the laser transmitter and receiver. There must be adequate "link margin" for the LOLA laser to obtain range observations during both lunar night and day, and for surfaces with a diverse range of infrared albedos and local height variations. "Link margin" is analogous to signal-to-noise ratio (SNR) and is simply a measure of the degree of confidence that an adequate number of photons above background level will be received and detected by the instrument in order for a useful range measurement to be achieved. The laser transmitter pulse energy and detector sensitivity must be adequately flexible to respond to a range of operational extremes. Our computations suggest that the instrument parameters listed in Table 1 are sufficient to meet the anticipated range of surface albedos and background 1.06-μm illumination conditions (e.g., solar). The experience with the Apollo Lunar Laser Altimeter provides adquate data to make this assessment (*Kaula et al.,* 1974; *Moore et al.,* 1980). Less is known about what to expect with respect to the spectrum of 30-300-m scale height variations and local slopes (Moore et al., unpublished data, 1969).

The primary engineering challenge associated with lunar orbital laser altimetry concerns development of a space-qualified laser using modern, all-solid-state technology. In addition, a compact, low-power laser backscatter waveform digitizer must be validated for operations in lunar orbit. Waveform digitizers record the shapes of input signals in terms of amplitude as a function of time and are electronically complex, but they have been space qualified for use in microwave instruments. Recent breakthroughs in all-solid-state laser oscillators (*Byer,* 1988) now permit high pulse-repetition-rate laser operations for at least 1 billion pulses, and perhaps up to 3 billion. This is because the traditional flashlamp method of pumping the Nd:YAG (a material that has replaced ruby) laser rod has been replaced with a highly efficient array of laser diodes. Flashlamps have traditionally been required to inject enough optical energy into the laser material for it to lase. Longer-lived arrays of laser diodes can now serve this purpose. The so-called diode-pumped Nd:YAG laser oscillator offers the required performance characteristics (e.g., 50-Hz repetition rate, short pulse, low power and mass) for spaceborne operation. Efficient, low-power diode-pumped Nd:YAG lasers have recently been space qualified by McDonnell Douglas for laser tracking purposes (*Long et al.,* 1989). The lunar environment with its total absence of an atmosphere is ideal for orbital laser altimeter operations. Figure 2 is a photograph of the LOLA breadboard that illustrates its compact form. Refer to Table 2 for specific payload parameters.

Airborne simulations of the LOLA breadboard are expected to occur during the 1990-1991 timeframe in order to facilitate development of a full-instrument prototype. Simulations of LOLA performance using degraded airborne laser altimeter profiles for lunar analog terrains are currently underway to explore potential requirements for lunar base site selection and activities. Examples from these datasets will be discussed in the next section.

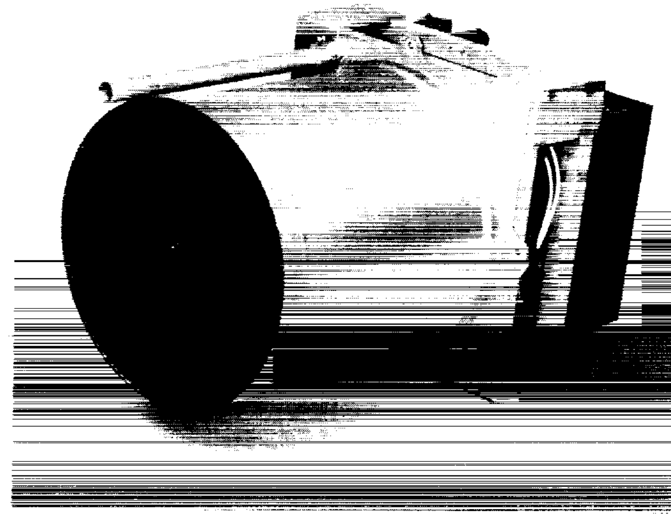


Fig. 2. Photograph of the LOLA breadboard with bar for scale. The instrument is similar in size to the Thermal Emission Spectrometer that is to be flown to Mars on the Mars Observer in 1992. The laser transmitter and receiver electronics are located in the rectangular box that is affixed atop the telescope. The laser beam is guided by the small tube parallel to the telescope.

LOLA SIMULATIONS

A high-altitude airborne laser altimeter instrument is currently in operation at NASA's Goddard Space Flight Center (*Bufton and Garvin,* 1987). This instrument is configured in a NASA Wallops Flight Facility T-39 Sabreliner jet aircraft, and is capable of obtaining laser profiles from altitudes as high as 11 km, or as low as 1-2 km above terrain. This airborne laser altimeter instrument has been used to acquire 3-10-m spatial resolution topographic profiles with submeter vertical precision for a variety of volcanic, erosional, and impact targets in the western and southwestern United States. Another currently operational instrument (at NASA's Wallops Flight Facility) is the Airborne Oceanographic Lidar (AOL), which has a low-altitude laser altimeter mode with 25-30-cm-diameter footprints and ~30-cm vertical resolution (*Hoge et al.,* 1984). We present and discuss the implications of profiles obtained from these airborne systems that have been degraded to LOLA spatial and vertical resolution using straightforward offset-averaging techniques. Specifically, for each n-point window (where "window" refers to the width in meters of a profile subsection), the mean and standard deviation of the topography is computed (i.e., n = 10 to 100). The window is then translated n points, and the process is repeated. The standard deviation is a good estimate of the LOLA vertical roughness parameter, while the n-point mean is a reasonable indicator of the 30-m footprint topographic measurement. Table 4 summarizes the range of 30-m-diameter footprint topographic standard deviations observed for a diverse set of lunar analog surfaces.

While most lunar landforms within the younger maria are relatively pristine, micrometeorite bombardment over the past 2-3 b.y. (AE) has resulted in the generation of a regolith layer that mantles the original topography, especially the lunar lava flows. The thickness of this regolith mantle varies from meters to hundreds of meters (*Wilhelms,* 1987). In spite of this lunar erosion effect, aspects of the morphology of such basic lunar landforms as craters, rilles, lava flow fronts, and hummocky ejecta have been preserved (*Wilhelms,* 1987); thus the LOLA simulations for the lunar analog landforms are relevant, even if they represent

endmember scenarios. Lunar base site selection is likely to be a very complex process involving many tradeoffs, and base location in proximity to youthful lunar volcanic or impact deposits is certainly worth consideration. What follows is a brief description of a set of LOLA simulation profiles for representative lunar analog landforms, with comments relevant to lunar base site selection and eventual operations.

Meteor Crater (Barringer), Northern Arizona

One of the most youthful impact craters on Earth is Meteor (Barringer) Crater, located in north-central Arizona (Fig. 3a). This well-studied impact crater is ~1.2 km in average diameter, and formed as a result of a ~20-megaton hypervelocity impact of an iron meteorite about 49,000 years ago (*Shoemaker,* 1987). Figure 4 is a LOLA resolution laser altimeter profile across the center of the crater from northwest to southeast, with the standard deviation of the 30-m-scale topography shown as individual dots. A full-resolution (3-m-diameter footprints) profile is illustrated in Fig. 3b for comparison. Figure 5 is a Lunar Orbiter III photograph of a simple impact crater 520 m in diameter in Oceanus Procellarum, which is located about 16 km from the Apollo 12 landing site (*Cintala et al.,* 1982) and is similar to Meteor Crater, in part due to its freshness. One can observe the 2-30-m-diameter impact-generated blocks within the ejecta blanket of this crater. As part of a study of excavation efficiency of the cratering process on the Moon, *Cintala et al.* (1982) measured the size distribution of all the blocks larger than 1.5 m in diameter around this crater as a function of distance from the crater center. Figure 6 displays the size distribution of ejecta blocks for the entire continuous ejecta blanket and indicates the significance of meter-scale roughness elements such as blocks at local scales, even on the relatively smooth lunar maria. The simulated LOLA topographic profile in Fig. 4 illustrates the 2-6-m vertical roughness (standard deviation of topography) that occurs in the Meteor Crater near-rim ejecta blanket. It is also possible to observe the large range of topographic variance associated with the inner crater walls, which are known to have local slopes as high as 41° (computed from Fig. 3b; see also Table 4). The inner wall slopes of the lunar crater shown in Fig. 5 are approximately as steep as those in Meteor Crater on the basis of shadow measurements. Such inner crater walls have over 25 m of vertical roughness in several instances (see Fig. 4). Although terrestrial erosion processes have filled in the floor of Meteor Crater beyond the extent typical of cratering-related slumping effects, most of the critical morphologic elements are preserved, including the discontinuous ejecta blanket. Thus it is possible to learn about the types of local topography and roughness that are likely to be common in association with the ubiquitous simple impact craters that frequently occur even on the smooth mare. If lunar base site selection is to follow the safety criteria imposed during the first few Apollo landings, then surfaces like those illustrated in Fig. 5 are likely to be commonplace, and the scales of roughness observed at Meteor Crater (Fig. 3a) will be relevant for base construction and local operations.

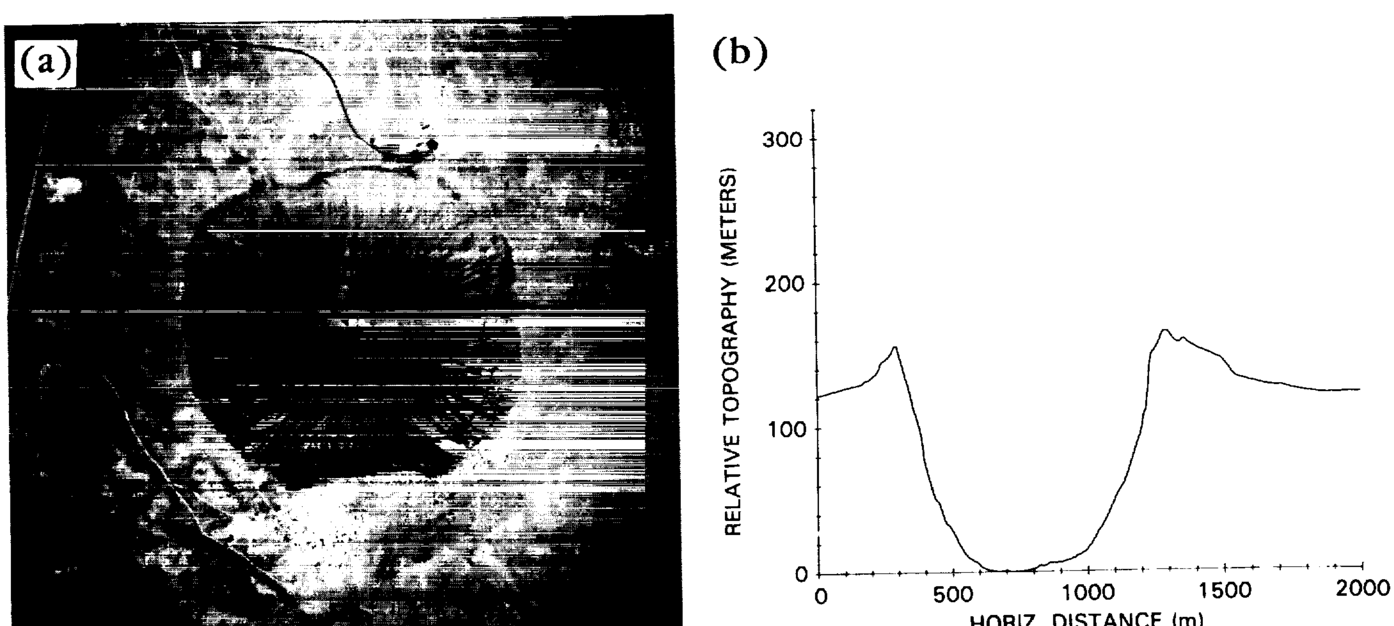


Fig. 3. **(a)** Vertical airphoto of Meteor Crater acquired in 1967 with a large format metric camera. Resolution is ~1 m. The frame shows the Crater Museum at the top (north), as well as the full extent of the preserved ejecta blanket (out to about 2 crater radii). The rim crest diameter of Meteor Crater averages 1.2 km, and the average depth is 165 m. **(b)** Airborne laser altimeter profile of Meteor Crater from northwest to southeast. Horizontal sampling internal is ~3 m, and vertical precision is submeter. This profile crosses the impact crater center and illustrates the asymmetry of the ejecta blanket (profile acquired in October 1986).

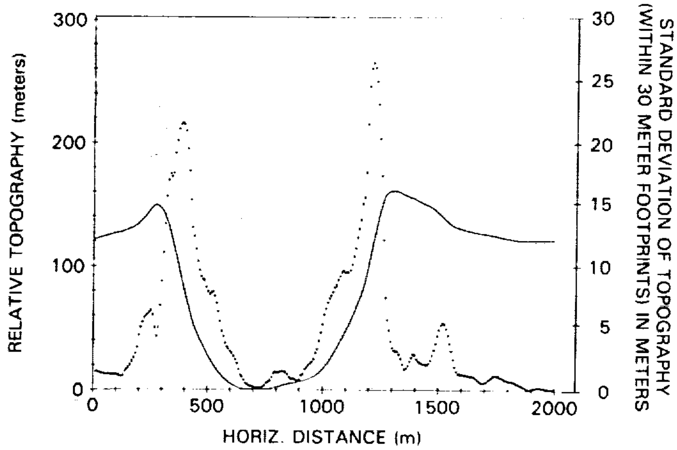


Fig. 4. Spatially degraded version of the profile shown in Fig. 3b using the sliding window offset technique described in the text. The horizontal sampling interval is 30 m. The superimposed dots are a measure of the standard deviation of topography within each 30-m footprint (see right axis). Note the edge effects displayed by this parameter, especially for the locally steep inner walls of the crater (see Table 4). The standard deviation parameter is a good estimate of the LOLA vertical roughness parameter to be derived from the shape of the backscattered laser waveform.

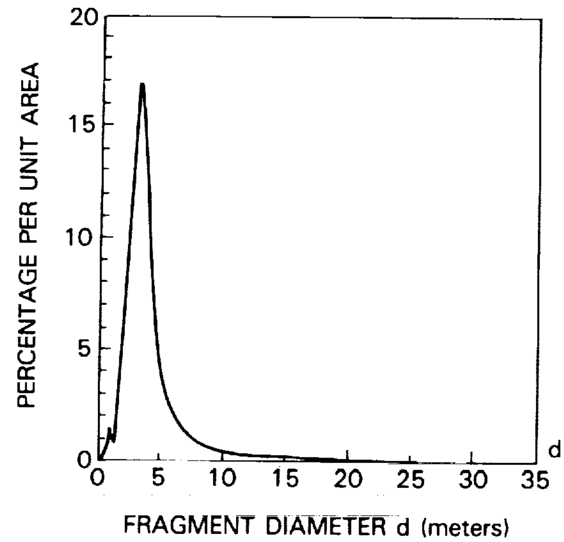


Fig. 6. Size frequency distribution for ejecta blocks within continuous ejecta deposit for the crater shown in Fig. 5 (*Cintala et al.,* 1982). In combination with orbital imagery, LOLA profile data would permit assessment of local hazards (such as 20-30-m-diameter blocks) associated with commonplace fresh impact craters. Note that only blocks 1.5 m and larger were included in the study due to orbital image resolution constraints (Lunar Orbiter III).

Fig. 5. Lunar Orbiter III image of a 520-m-diameter crater near the Apollo 12 site (Oceanus Procellarum) in a region of the maria with thin regolith (~7 m). Observe the 2-30-m-diameter ejecta blocks scattered around the rim of this simple, fresh impact crater.

Grand Canyon, Northern Arizona

As an example of the extremes of lunar topography, LOLA resolution topographic profiles of the interior of the Grand Canyon (including the Colorado River, Fig. 7a) have been examined. While erosional topography associated with fluvial processes acting on a variety of sedimentary, igneous, and metamorphic lithologies is not likely to exist on the Moon, the deepest lunar rilles and most youthful complex impact craters are known to have steep topography (*Wilhelms,* 1987). Thus we use the severe local slopes associated with the southern Grand Canyon as a means of demonstsrating a worst-case scenario for the Moon. Figure 7b is a 3-m spatial resolution profile from south to north across the southern rim of the Grand Canyon; the deepest point is the Colorado River (at right). Figure 8 is a LOLA resolution profile with superimposed vertical roughness shown by individual dots, as with Meteor Crater above. Up to 75 m of vertical relief is observed within the 30-m-diameter simulated LOLA footprints (Table 4). Deep lunar sinuous rilles, which may have formed by thermal erosion from turbulent, low-viscosity lavas, could have over 50 m of relief within each 30-m LOLA footprint. It is clear from the LOLA simulation that high spatial and vertical resolution orbital laser altimetry will permit adequate sampling of the local topography, slopes, and roughnesses of many types of lunar surfaces, however extreme or subtle. With respect to lunar base site selection, the location of a base in close proximity to deep rilles may offer advantages with respect to possible resource development, as the walls of such landforms often display exposed volcanic units (e.g., as at Apollo 15 near Hadley Rille), which could provide accessible materials for various purposes. Knowledge of the extremity of local topography and slopes is certainly a requirement to ensure that the potential base site is not within possible slump-failure zones, which might be associated with the areas adjacent to the walls of such rilles.

While meter-resolution stereophotogrammetric techniques have been successfully employed for measuring local slopes (e.g., H. J. Moore et al., unpublished data, 1969; lunar topographic photomaps), it is virtually impossible to establish geodetic control with such methods beyond the field of view of individual stereo pairs; orbital altimetry is usually necessary to provide the required

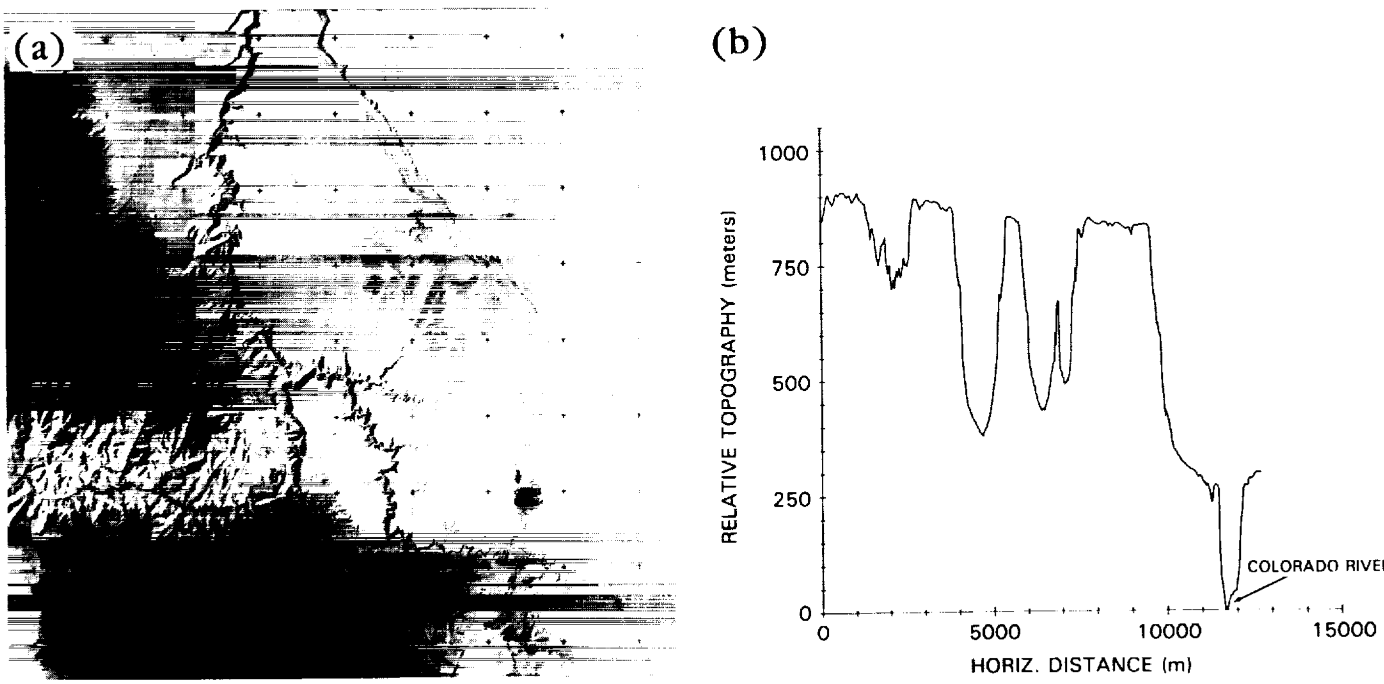


Fig. 7. (a) Landsat MSS orbital image of the Grand Canyon region of northern Arizona acquired in autumn of 1980 at a resolution of 80 m. The profile shown in (b) extends from south to north and traverses the section of the canyon illustrated in the middle left of this image (from the south rim to the Colorado River). This image shows the east-west and other branches of the Grand Canyon, as well as the heavily forested area south of the canyon (dark region in lower part of image). (b) Airborne laser altimeter profile of the southern part of the Grand Canyon (from south to north in Arizona). The deepest point (at right) is the Colorado River. As in Fig. 3b, the horizontal sampling interval is ~3 m (profile acquired in October 1986).

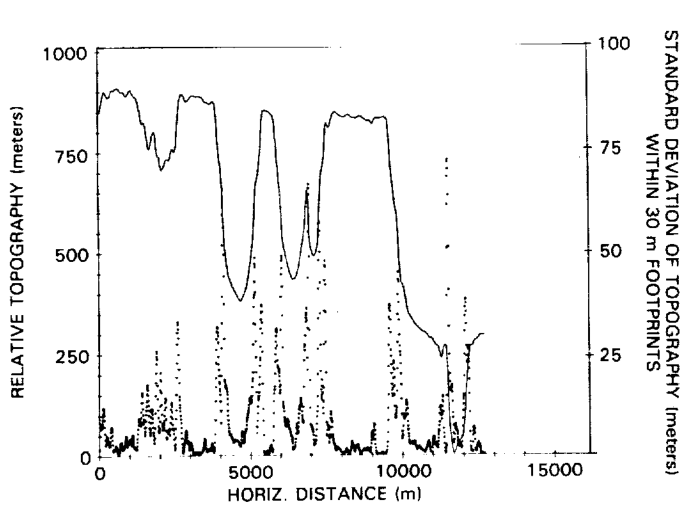
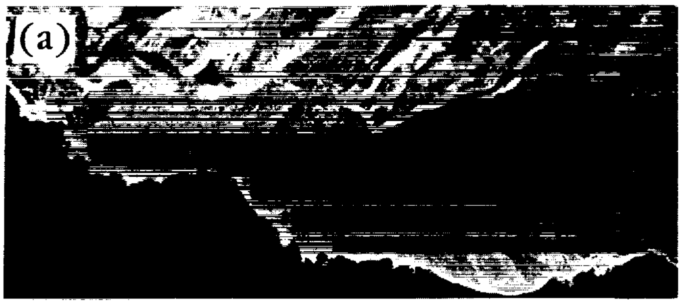


Fig. 8. As in Fig. 4, simulated LOLA profile for the Grand Canyon derived from the data shown in Fig. 7b, with superimposed 30-m scale topographic variance parameter (see righthand side vertical axis for scale) shown as dots.

longer-wavelength control. In addition, stereophotogrammetric methods are somewhat operator dependent (subjective), and time consuming. Our approach has been to design an orbital altimeter system capable of both tasks, and the LOLA simulations of the Grand Canyon and Meteor Crater discussed above provide some justification of this claim.

Lava Channel, Southwest Iceland

As a final example, an ~1-km-wide lava channel in southern Iceland (part of the Ögmundarhraun lava flow sequence on the Reykjanes Peninsula, Fig. 9a) is illustrated by means of a 30-cm horizontal resolution topographic profile in Fig. 9b (these data were collected by the AOL laser altimeter in a NASA P-3 aircraft). This channelized basaltic lava flow is extremely smooth at a variety of length scales (i.e., only 25-30 cm of topographic variance over baselines of hundreds of meters as indicated in Table 4) and represents a classic example of a low-viscosity Icelandic lava surface. The lava channel stands only about 6 m above an older flow surface. Figure 10 is a LOLA resolution profile of the perched lava channel that illustrates the 20-40-cm vertical roughness that



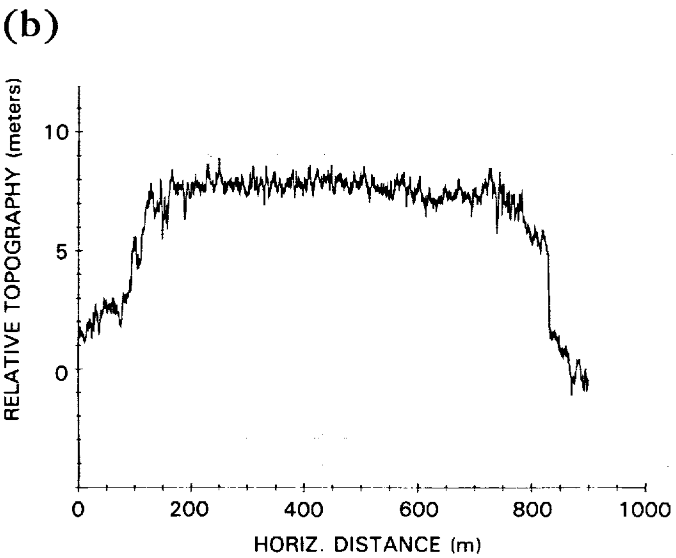


Fig. 9. **(a)** Airborne synthetic aperture radar (SAR) image of the Ögmundarhraun lava field in the Reykjanes region of southwestern Iceland. The radar look direction is from the south to the north at an average incidence angle of 15° from the horizon, and at X-band wavelength (data collected for NASA by INTERA Technologies using the STAR-2 system with 6-m resolution). This lava field consists of low-viscosity basaltic flows, many of which formed channels. The main lava channel in the center of the image displays well-defined levees, and an east-to-west cross-flow profile of this channel is shown below in **(b)**. The Ögmundarhraun lava field was apparently emplaced about 700 years ago in a high volume eruption rate event (*Garvin et al.,* 1989). **(b)** Airborne laser topographic profile of a lava channel in the southern Reykjanes region (Ögmundarhraun) of Iceland acquired with a horizontal sampling interval of ~30 cm by the AOL instrument in May 1987 (*Hoge et al.,* 1984). This smooth basaltic lava flow represents an analog to the smoothest lunar maria lavas and has an effective yield strength similar to that for Imbrium flows (*Garvin et al.,* 1989).

is characteristic of the flow interior, and the 2 m of topographic variance associated with the flow margins (levees). This profile extends from east to west across the long-axis of the flow direction (transverse profile). Low-viscosity lunar lava flows are inferred to behave much like this Icelandic basalt example on the basis of rheologic parameters estimated by *Hulme* (1975), *Moore et al.* (1978), and *Hulme and Fielder* (1977). Using the *Moore et al.* (1978) method as applied to profile in Fig. 10, we estimate the yield strength of the Icelandic flow to be on the order of 1300 Pascals (*Garvin et al.,* 1989). If lunar bases or science outposts were to be located in proximity to the most youthful

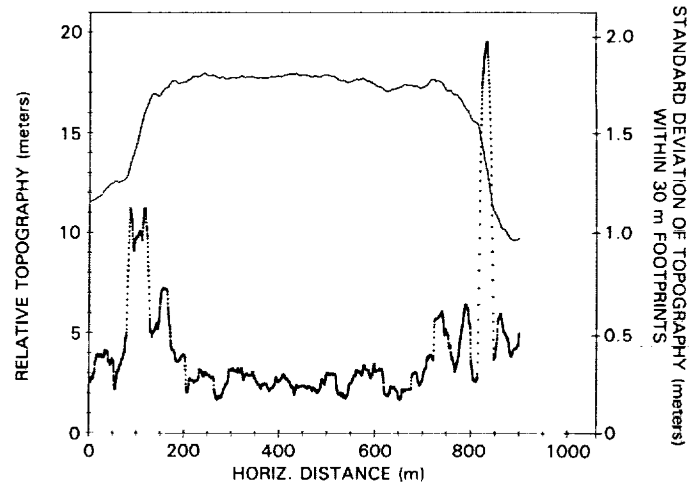


Fig. 10. The LOLA profile (as in Fig. 4) of the lava channel shown in Fig. 9b at 30-m horizontal sampling interval, with superimposed topographic variance parameter (dots). Observe the submeter scale roughness of this low-viscosity lava flow. Such surfaces could be common on the youngest lunar maria (H. J. Moore et al., unpublished data, 1969).

low-viscosity lunar flows, perhaps to investigate their accessory mineral abundances (i.e., to measure Cr, Ni, Ti contents etc.), then examination of high-resolution topographic profiles may be useful in terms of identifying those flows with the least regolith development and hence the most pristine and accessible surfaces. Extremely smooth lava surfaces like that illustrated from southern Iceland are inferred to exist on the Moon, possibly associated with the freshest mare surfaces, but direct observation of their occurrence has not been possible. Lunar Observer Laser Altimeter topographic data could assist in the identification of the smoothest lunar surfaces in association with known volcanic features. Perhaps channelized lunar lava flows with submeter surface textures could be identified within Copernican-age surface units of suggested volcanic origin and evaluated as a possible lunar base site on the basis of synergistic laser altimeter data and high-resolution images.

Table 4 summarizes the anticipated range of within-footprint (on a 30-m baseline) vertical roughness as computed from the LOLA simulation profiles (and displayed as scaled standard deviation of topography on the figures). The most severe case would result in ~50 m of dynamic range, with 20-30 m expected for the inner walls of fresh, simple impact craters. Smooth maria are likely to have anywhere from <1 m to a few meters of vertical roughness, and for such surfaces, vertical precision of the LOLA measurements could approach the 15-cm lower limit. The potential ability of LOLA to detect locally rugged (as well as extremely smooth) topography will provide an important constraint in choosing lunar base location, especially for target areas not within the zone that was intensively surveyed for the Apollo site selection process (largely from stereo orbital high-resolution photography).

SUMMARY

Analysis of laser altimeter profiles for representative lunar analog surfaces demonstrates that high spatial resolution topography of the lunar surface is a very desirable if not necessary

dataset for lunar base site selection and subsequent operations. A simple Observer-class instrument such as the Lunar Observer Laser Altimeter (LOLA) now under development as part of NASA's planetary program would be well suited for characterizing lunar topography and surface roughness at a variety of length scales appropriate for both addressing fundamental geoscience problems and local site selection and evaluation. A LOLA-class instrument would be ideally suited for operation in a low-altitude orbit around any body devoid of an atmosphere including the Moon, Mercury, or an asteroid.

Lunar base site selection will clearly involve complex decisions and tradeoffs on the basis of science potential, operational and safety factors, objectives, and types of precursory data that will be available. We suggest that laser topographic profiles and global topographic maps, together with orbital imaging (stereo, high-resolution, and multispectral) and Earth-based radar mapping will provide a necessary and sufficient framework from which to select optimal future lunar landing sites (e.g., for robotic rovers and astronauts) and eventually lunar base localities.

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REFERENCES

Bufton J. L. (1989) Laser altimetry measurements from aircraft and spacecraft. *Proc. IEEE, 77,* 463-476.

Bufton J. L. and Garvin J. B. (1987) Performance testing of the Shuttle Laser Altimeter. In *Proc. Optical Soc. Am. Topical Mtg. on Laser and Optical Remote Sensing,* pp. 13-16.

Byer R. L. (1988) Diode laser-pumped solid state lasers. *Science, 239,* 742-747.

Cintala M. J., Garvin J. B., and Wetzel S. J. (1982) The distribution of blocks around a fresh lunar mare crater (abstract). In *Lunar and Planetary Science XIII,* pp. 100-101. Lunar and Planetary Institute, Houston.

Garvin J. B., Williams R. S., and Jónsson J. (1989) Geologic remote sensing of southwestern Iceland: An imaging radar and laser altimeter approach to problems in planetary volcanism. In *Proc. 28th Intl. Geol. Congress,* pp. 1-531 to 1-532. Washington, DC.

Hoge F., Krabill W., and Swift R. (1984) The reflection of airborne UV laser pulses from the ocean. *Marine Geodesy, 8,* 313-344.

Hood L., Sonnett C., and Russell C. (1985) The next generation geophysical investigation of the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 253-264. Lunar and Planetary Institute, Houston.

Hulme G. (1975) The interpretation of lava flow morphology. *Geophys. J. R. Astron. Soc., 39,* 361-383.

Hulme G. and Fielder G. (1977) Effusion rates and rheology of lunar lavas. *Philos. Trans. R. Soc. London, A285,* 227-234.

Kaula W., Schubert G., Lingenfelter R., Sjogren W., and Wollenhaupt W. (1974) Apollo laser altimetry and inferences as to lunar structure. *Proc. Lunar Sci. Conf. 5th,* pp. 3049-3058.

Long L. E., Ryan-Benoist R., and Konopelski T. W. (1989) Space-qualified diode-pumped slab laser. Paper given at SPIE, Los Angeles, California, January 1989. 7 pp.

Moore H. J., Arthur D., and Schaber G. (1978) Yield strengths of flows on the Earth, Mars, and Moon. *Proc. Lunar Planet. Sci. Conf. 9th,* pp. 3351-3378.

Moore H. J., Boyce J., Schaber G., and Scott D. (1980) Lunar remote sensing and measurements. *U.S. Geol. Surv. Prof. Pap. 1046-B,* pp. B1-B78.

Phillips R. (1986) *Contributions of a Lunar Geoscience Observer to fundamental questions in lunar science.* LGO Science Working Group Members, Dept. Geol. Sci., SMU, Dallas. 86 pp.

Ravine M. A. and Grieve R. A. F. (1986) An analysis of morphologic variation in simple lunar craters. *Proc. Lunar Planet. Sci. Conf. 17th,* in *J. Geophys. Res., 91,* E75-E83.

Sharpton V. and Head J. W. (1982) Stratigraphy and structural evolution of southern Mare Serenitatis: A reinterpretation based on Apollo Lunar Sounder Experiment data. *J. Geophys. Res., 87,* 10,983-10,998.

Shoemaker E. M. (1987) Meteor Crater, Arizona. *Geol. Soc. America Centennial Field Guide—Rocky Mountain Section,* pp. 399-404.

Smith D. E., Zuber M., Garvin J., Bufton J., and Abshire J. (1989) The Mars Observer Laser Altimeter (MOLA) (abstract). In *Fourth Intl. Conf. on Mars,* Jan. 1989. Tucson, AZ, pp. 191-192.

Wilhelms D. E. (1985) Unmanned spaceflights needed as scientific preparation for a manned lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 245-252. Lunar and Planetary Institute, Houston.

Wilhelms D. E. (1987) *The Geologic History of the Moon.* U.S. Geol. Surv. Prof. Pap. 1348. 302 pp.

A SEARCH FOR INTACT LAVA TUBES ON THE MOON: POSSIBLE LUNAR BASE HABITATS

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We have surveyed lunar sinuous rilles and other volcanic features in an effort to locate intact lava tubes that could be used to house an advanced lunar base. Criteria were established for identifying intact tube segments. Sixty-seven tube candidates within 20 rilles were identified on the lunar nearside. The rilles, located in four mare regions, varied in size and sinuosity. We identified four rilles that exhibited particularly strong evidence for the existence of intact lava tube segments. These are located in the following areas: (1) south of Gruithuisen K, (2) in the Marius Hills region, (3) in the southeastern Mare Serenitatis, and (4) in eastern Mare Serenitatis. We rated each of the 67 probable tube segments for lunar base suitability based on its dimensions, stability, location, and access to lunar resources. Nine tube segments associated with three separate rilles are considered prime candidates for use as part of an advanced lunar base.

INTRODUCTION

Early observations indentified many meandrous channels, or sinuous rilles, on the lunar surface (*Schröter,* 1788). Since then, numerous studies have shown that these features formed as a result of the extrusion of hot, fluid, low-viscosity basaltic magma (e.g., *Hulme,* 1973; *Wilson and Head,* 1981; *Coombs et al.,* 1987), some of which may have evolved into lava tubes when segments of the channels roofed over (e.g., *Oberbeck et al.,* 1969; *Greeley,* 1971; *Cruikshank and Wood,* 1972; *Hulme,* 1973; *Coombs et al.,* 1987). The prospect of using the natural cavity formed by a drained intact lunar tube for housing a manned lunar base has long been the subject of speculation (e.g., *Brown and Finn,* 1962; *Henderson,* 1962).

In his discussion of lava tubes as potential shelters for lunar habitats, *Hörz* (1985) noted that the lunar lava tubes would be ideal for locating the lunar base because they (1) require little construction and enable a habitat to be placed inside with a minimal amount of building or burrowing; (2) provide natural environmental control; (3) provide protection from natural hazards (i.e., cosmic rays, meteorites and micrometeorite impacts, impact crater ejecta); and (4) provide an ideal natural storage facility for vehicles and machinery.

A lava tube may form when an active basaltic lava stream or leveed flow develops a continuous crust. More specifically, depending on the rate of flow and the rheology of the lava, a lava tube may form by one of several methods (e.g., *Cruikshank and Wood,* 1972; *Greeley,* 1987): (1) An open channel may form a crust that extends from the sides to meet in the middle and may eventually thicken and form a roof; (2) in more vigorous flows, the crustal slabs may break apart and raft down the channel; as pieces are transported down the channel they may refit themselves together to form a cohesive roof; (3) periods of spattering, sloshing, and overflow may form levees that may eventually build upward and inward and merge into a roof; or (4) lava tubes may also form by the advancement of pahoehoe lava toes (see *Wentworth and McDonald,* 1953). Under the conditions of lunar basaltic eruptions (lower gravity field, no atmosphere), such processes would have produced lunar lava channels and associated tubes at least an order of magnitude greater in size than those found on Earth (*Wilson and Head,* 1981). Such lunar lava tubes could be tens to hundreds of meters wide by hundreds of meters deep and tens of kilometers long. These dimensions make lunar lava tubes ideal sites in which to house a lunar base habitat.

Swann (personal communication, 1988) has raised the question of the existence of open, evacuated lunar lava tubes. He contends that the lunar lava tubes may have been filled in upon cessation of the eruption that formed them. Here, we present evidence to the contrary and show that it is entirely possible that open lava tubes were left on the Moon and that it is very likely that they still exist. Work done by *Hulme* (1973, 1982), *Wilson and Head* (1980, 1981), and others has shown that lunar sinuous rilles are products of low-viscosity, high-temperature basaltic lava flows, and that these features are very similar to terrestrial basaltic lava channels and tubes. With the lack of extensive field evidence from the Moon, terrestrial lava tubes have often been studied as examples of what the formation process would have been like for the lunar features (e.g., *Greeley,* 1971; *Cruikshank and Wood,* 1972). Although there is an order of magnitude size difference between lunar and terrestrial tubes, the formational processes appear very similar.

Terrestrial evidence indicates that a tube may or may not become plugged with lava depending on the viscosity, temperature, supply rate, and velocity of the lava flowing through it. Numerous terrestrial lava tubes have been observed during their formation (e.g., *Greeley,* 1971; *Cruikshank and Wood,* 1972; *Peterson and Swanson,* 1974). Of the tubes observed during the process of formation, most did not become congealed and solidify into hard tubular masses of rock at the time of their origin (*e.g., Greeley,* 1971; *Cruikshank and Wood,* 1972). It has been noted that as the supply rate of lava diminished during an eruption, the level of liquid in the tube dropped, leaving a void space between the top of the lava flow and the roof of the tube (*Peterson and Swanson,* 1974). At the end of an eruption most of the lava was

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observed to drain out of the main tube to leave an open tunnel of varying dimensions (e.g., *Macdonald et al.,* 1983). However, the tubes may later act as conduits for younger lava flows erupting from the common source vent. These later flows may partly or completely fill the older lava tube depending on their supply rate and amount of material flowing through. While we cannot discount that similar occurrences may have happened during the formation of the lunar lava tubes, it is unlikely that it happened to a high percentage of them. Field observations of several volcanic terrains in Hawaii indicate that the majority of lava tubes formed remain partially void, that is, less that 30% of the tube has been (partially) infilled by later flows. Fewer than 1% of the lava tubes observed in the field by *Coombs et al.* (1989) are completely filled in by later flows (e.g., Makapu'u Tube, Oahu, Hawaii). Of the several hundred lava tubes known to exist on and around Kilauea, more than 90% are open or void; less than 30% of the tube has been filled in by later flows. In those tubes that have been partially infilled, the lava toes extend inward from the tube walls. Very few lava tubes are completely filled and, in fact, only one, Makapu'u Tube, on Oahu, Hawaii, was observed by the authors to have been completely filled in by later flows (*Coombs et al.,* 1989).

Other evidence, derived from the study of lunar volcanic features, suggests that not all lunar tubes are plugged with congealed lava and that void lava tubes exist on the Moon. Many lava channels have been identified on the lunar surface (e.g., *Oberbeck et al.,* 1969; *Cruikshank and Wood,* 1972; *Masursky et al.,* 1978; *Schaber et al.,* 1976; *Greeley and King,* 1977). These channels do not appear to be filled with solidified lava along most of their lengths. The same is true for most lunar sinuous rilles. *Greeley and King* (1977) demonstrated that thin lunar flow units typically were emplaced by lava tubes. It appears that the low-viscosity lunar lavas drained from large segments of the identified channel-tube systems and sinuous rilles. Finally, it should be noted that there are numerous instances where several tube candidates are located along sinuous rilles and where these tube segments are bounded at each end by open, deep rille segments. Since these open, deep rille segments are not filled with congealed lava, we suggest that the roofed portions of the rille (i.e., the tubes) are not filled and are likely to be void or partly void tubes.

A great deal of discussion has centered around the strength and durability of existing lunar lava tubes (*Oberbeck et al.,* 1972). Whether or not the tube roofs are structurally stable enough to withstand prolonged meteoroid impact and sufficiently thick enough to provide protection from cosmic radiation have been pressing questions (*Hörz,* 1985). The results of calculations by *Oberbeck et al.* (1969), as well as terrestrial field evidence, support the concept that many of these features are evacuated and have remained intact during the billions of years of meteoritic bombardment and seismic shaking to which they have been subjected since their formation. *Oberst and Nakamura* (1988) have examined the seismic risk for a lunar base but have not dealt specifically with the effects of moonquakes on lunar lava tubes.

Many intact lava tubes exist along the east and southwest rift zones of Kilauea Volcano on the Big Island of Hawaii. Two of the largest of these tubes are Thurston Lava Tube (Keanakahina), which was formed 350-500 years b.p., and an unnamed tube on the floor of Kilauea Caldera that was formed during the 1919 eruption of Halemaumau. The respective average dimensions of these two lava tubes are 4.90 m wide × 2.20 m high and 8.60 m wide × 3.73 m high. Both of these lava tubes have maintained their structural integrity while constantly being shaken by local seismic tremors. On any given day the summit area of Kilauea may experience as many as 300 or more earthquakes of magnitude 4 or less (*Klein et al.,* 1987). The frequency and magnitude of these earthquakes intensifies immediately prior to the onset of an eruption. Much greater magnitude earthquakes are also common to the area as evidenced by two fairly major earthquakes that occurred in the recent past. The 1975 Kalapana and 1983 Kaoki earthquakes, with magnitudes of 7.5 and 6.6 respectively, appear to have had no effect on these two lava tubes. In summary, a long and varied seismic history has had minimal effect on the lava tubes in the Kilauea area and elsewhere on the island of Hawaii. It is possible then, that many of the lunar lava tubes have remained intact over the billions of years of the seismic shaking generated by meteorite impacts and tectonically originated moonquakes.

Oberbeck et al. (1969) determined that the ratio of roof thickness to interior tube width in terrestrial lava tubes ranges from 0.25-0.125. When applied to lunar conditions, they calculated that a 385-m-wide tube roof would remain stable providing the roof was 65 m thick. They further noted that the effect of roof arching, common in terrestrial lava tubes, would allow even thinner 385-m-wide roofs to remain intact and, if the lunar rock is assumed to be more vesicular than that on Earth, the maximum stable width could reach 500 or more meters. Thus, stable tube roofs could exist on the Moon provided that they span a width of no more than a few hundred meters and that the larger tubes have roofs that are at least 40-60 m thick.

As *Hörz* (1985) pointed out, these estimates are also in agreement with other important observations. He noted, for example, that impact craters a few tens of meters to 100 m across are supported by uncollapsed roofs of identified lava tubes. With depth/diameter ratios of small lunar craters ~1/4-1/5 (*Pike,* 1976), the excavation depths of these craters superposed on the lava tubes could reach 25 m. Hörz estimated further that the roof thickness must be at least twice any crater depth, otherwise a complete penetration of the tube roof would have occurred.

In this paper we propose a set of criteria for the identification of intact lunar lava tubes. A survey of all known sinuous rilles and channels, as well as other selected volcanic features, was conducted in an effort to locate lava tube segments on the lunar surface. In addition, we have attempted to assess the potential of the identified lava tubes as candidates for lunar base sites. This paper presents the results of our survey.

METHOD

We conducted a survey of all available Lunar Orbiter and Apollo photographs in order to locate possible intact lava tubes. The criteria used to identify tube candidates were (1) the presence of an uncollapsed, or roofed, segment or, preferably, a series of segments along a sinuous rille; (2) the presence of uncollapsed segments between two or more elongate depressions that lie along the trend of the rille; and (3) the presence of an uncollapsed section between an irregular-shaped depression, or source vent, and the rest of the channel.

The above criteria were applied to all previously identified lunar sinuous rilles (e.g., *Oberbeck et al.,* 1969; *Schultz,* 1976) and others identified from the survey of all available Lunar Orbiter and Apollo photographs. Other volcanic features such as endogenic depressions, crater chains, and other types of rilles were also examined as some may be associated with a lava tube. Exogenic crater chains were distinguished from the partially collapsed tube

segments by the lack of crater rays and herringbone patterns in the vicinity of the latter. Also, exogenic crater chains are commonly composed of closely spaced or overlapping round to oval craters, while the craters in endogenic chains are elongate or irregular in form. Further, secondary craters within a chain are often deeper at one end than the other and downrange craters are commonly superposed on uprange craters. Endogenic craters are more uniform in depth and generally do not exhibit systematic overlap relationships.

The locations of the identified lava tube segments and various measurements that were made for each tube are given in Table 1. Maximum tube widths were estimated by projecting the walls of adjacent rille segments along the roofed-over segments. Tube lengths were measured from the Lunar Orbiter and Apollo photographs. An estimate of the depth to each tube, or roof thickness, was made, when possible, following the crater-geometry argument presented by *Hörz* (1985) whereby the largest impact crater superposed on an uncollapsed roof may yield a minimum measure of roof thickness. To calculate this minimum roof thickness the following equation was used

$$d \cdot 0.25 \cdot 2 = t \qquad (1)$$

where d is the maximum crater diameter superposed on the tube segment and t is the estimated minimum thickness of the tube segment. This equation provides a conservative estimate of the

TABLE 1. Lava tube candidates.

ID No.	Latitude	Longitude	Orbiter Frame No.	Tube Length (km)	Tude Width (km)	Crater Width* (km)	Roof Thickness† (m)	Rank‡ (A = Prime, B = Good, C = Possible)
A1	22°00′ N	30°30′ W	IV-133-H_3	2.20	0.55	0.44	110	B
2				1.10	0.55	0.22	110	A
3				0.55	0.33	0.22	110	C
B1	22°00′ N	29°oo′ W	IV-133-H_3	0.88	044	022	110	B
2				1.10	0.55	0.55	276	C
3				4.40	0.03	0.22	110	B
4				0.55	0.33	0.44	220	C
5				1.10	0.33	0.11	56	C
6				5.50	0.44	0.44	220	B
C1	35°00′ N	43°00′ W	V-182-M	0.90	0.72	0.06	30	A
2				1.20	0.97	0.09	46	A
3				0.66	0.60	0.27	136	B
4				0.54	0.60	0.12	60	A
5				0.60	0.45	0.09	46	A
6				0.60	0.82	0.09	46	A
7				0.51	0.82	0.09	46	A
D1	36°00′ N	40°00′ W	V-182-M	0.90	0.15	0.06	30	C
E1	36°00′ N	40°00′ W	IV-158-H_2	4.92	1.20	0.80	60	B
2	47°30′ N	57°00′ W		1.20	0.36	—	—	B
3				2.40	0.48	0.24	120	B
4				1.32	0.60	0.24	120	B
F1	30°00′ N	48°30′ W	IV-158-H_1	2.40	1.20	0.60	300	C
G1	27°00′ N	41°45′ W	V-191-M	2.37	0.32	0.38	190	B
2	26°30′ N	42°00′ W		1.87	0.23	0.75	375	C
3	27°00′ N	42°00′ W		0.88	1.25	0.20	100	C
H1	15°30′ N	47°00′ W	IV-150-H_2	1.65	0.88	0.22	110	C
I1	10°30′ N	49°00′ W	IV-150-H_2	1.32	0.88	—	—	B
2				1.32	0.77	0.22	110	C
3				6.60	0.66	—	—	C
4				4.60	0.55	0.33	165	C
Ja1	15°00′ N	57°00′ W	V-213-M	0.25	0.33	0.08	40	B
2				0.25	0.34	0.25	125	B
3				0.25	0.25	0.03	15	B
4				0.13	0.33	0.05	25	C
b5				0.35	0.37	0.08	40	B
6				0.25	0.32	0.05	25	B
7				1.63	0.33	0.15	75	B
8				1.63	0.38	0.08	40	C
9				7.37	0.47	0.25	125	C
10				0.37	0.28	0.08	40	C
11				0.75	0.48	0.08	40	C

TABLE 1. (continued).

ID No.	Latitude	Longitude	Orbiter Frame No.	Tube Length (km)	Tude Width (km)	Crater Width* (km)	Roof Thickness† (m)	Rank‡ (A = Prime, B = Good, C = Possible)
K1	12°00′ N	53°00′ W	IV-157-H_2	6.60	0.55	0.33	165	C
2				10.45	1.03	1.10	525	C
3				8.80	0.88	0.66	330	C
4				2.75	0.66	0.11	55	B
5				1.10	0.33	—	—	B
6				5.50	0.55	0.22	110	C
L1	2°00′ N	44°00′ E	IV-66-H_1	3.30	1.32	—	—	C
2				6.60	1.32	—	—	C
M1	4°00′ N	28°00′ E	IV-78-H_1	7.70	0.88	—	—	C
2				4.40	0.55	—	—	C
N1	11°00′ N	20°00′ E	IV-85-H_2	17.60	0.77	0.99	495	C
2				2.53	1.10	0.33	165	C
C				4.62	0.99	—	—	C
O1	18°00′ N	26°00′ E	IV-78-H_2	0.77	0.51	0.07	35	B
2				0.32	0.51	0.05	25	B
3				0.32	0.48	0.05	25	B
4				0.88	0.48	0.03	15	B
5				0.77	0.46	0.08	40	B
P1	51°00′ N	8°00′ W	V-129-M	0.84	0.21	0.13	65	B
2				1.26	0.21	0.17	85	C
3				1.68	0.29	0.13	65	C
Q1	16°00′ S	37°00′ W	IV-137-H_2	1.65	0.44	—	—	C
R1	29°00′ N	29°00′ E	IV-85-M	1.32	1.10	—	—	A
2				7.15	0.55	0.66	330	A
?1	20°00′ N	30°00′ E	IV-78-H_3	—	—	—	—	C
?2	1°00′ N	28°00′ E		—	—	—	—	C

*Crater width = maximum crater diameter measured on top of tube segment.
†Roof thickness = minimum tube roof thickness from depth of largest superposed crater (after *Hörz*, 1985).
‡Refers to the suitability of a particular tube segment for locating the advanced manned lunar base.

roof thickness, whereby the maximum crater depth is approximately one-quarter the crater diameter (0.25d) and the roof thickness (t) is at least twice that depth (after *Hörz*, 1985).

As part of the assessment of the suitability of the identified tube segments for housing an advanced lunar base, several factors were considered. The usefulness (presence of scientific targets near the base site, geologic diversity) of the locality, whether or not the site would be readily available (minor excavation and construction) for habitation, and its location were considered. The presence of potential lunar resources (e.g., high-Ti mare basalt or pyroclastics) in the region and ease of access to these deposits were important considerations, as were the proximity and ease of access to all features of interest in a region. Localities were sought in which little or no degradation appears to have occurred along and adjacent to the tube segments. The preferred sites exhibited few impact craters and/or young ridges that may indicate the presence of a fault system. It has been demonstrated that a more central location (near the equator) on the lunar nearside would provide a better site for the mass driver to launch various payloads to L_2 (*Heppenheimer*, 1985). Finally, base sites were sought where the rille/tube was located in a very flat region for greater ease of mobility and where tube widths and roof thicknesses were thought to be within the constraints determined by *Oberbeck et al.* (1969).

IDENTIFIED LAVA TUBE SEGMENTS

More than 90 lava tube candidates were identified along 20 lunar rilles on the lunar nearside, 67 of which were measured. These occur in four mare regions: Oceanus Procellarum, Northern Imbrium, Mare Serenitatis, and Mare Tranquillitatis (Fig. 1). Each of these rilles appears to be discontinuous, alternating between open lava channel segments and roofed-over segments. Those segments appearing structurally sound, according to the criteria discussed above, were measured and identified as tube candidates. Each of these tubes has met our tube identification criteria, is relatively close to the size constraints established by *Oberbeck et al.* (1969), and is situated in a relatively flat location. Table 1 lists each of these tube segments with its respective dimensions, location, and photographic reference frame. Of the 20 rilles examined, 4 exhibit very strong evidence for having intact, open tube segments, 2 in Oceanus Procellarum (C and J in Fig. 1; Figs. 2 and 3; Table 1) and 2 in Mare Serenitatis (O and R in Fig. 1; Figs. 4 and 5; Table 1). Each tube segment was evaluated for use as part of the structure of an advanced lunar base according to the criteria discussed above. The results of this evaluation are given in Table 1. Nine tube segments associated with three separate rilles (A, C, and R) are considered prime candidates for an advanced lunar base.

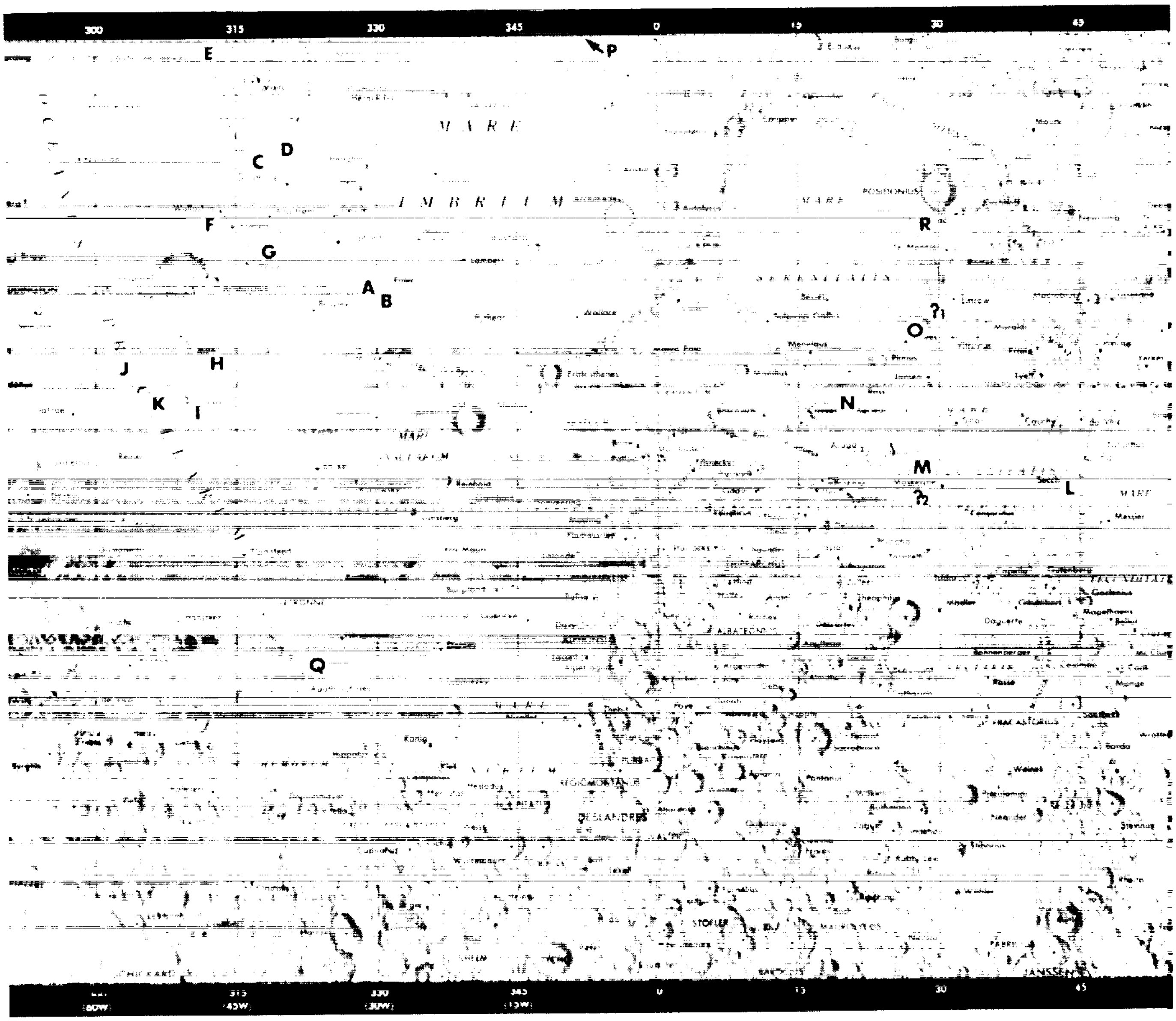

Fig. 1. Map showing the location of the 20 lunar rilles on the lunar nearside where potential lava tube candidates have been identified. Additional information concerning the location of these lava tubes is given in Table 1.

Northern Procellarum

Eleven rilles where probable intact lava tubes exist are found in the Oceanus Procellarum region; of these, two may be considered prime localities at which to find an intact lava tube or series of tubes (C and J in Fig. 1; Figs. 2 and 3; Table 1). Rille C is the classic example of a lunar lava tube and was first identified as such by *Oberbeck et al.* (1969). This rille is 60 k long and is broken into more than 15 segments, each of which may be a potential intact tube. Only seven of these were measured, however. These seven segments are longer than the others and have smaller craters superposed on their roofs. Lengths of these segments range from 510 to 1120 m, with widths ranging from 450 to 970 m. The 970-m-wide segment (as well as the other segments listed in Table 1 as > 500 m wide) was included because it appeared structurally stable in the photographs and was considered worthy of closer inspection. A variety of factors may allow tubes with apparent widths greater than 500 m to exist on the lunar surface. The largest crater on any of the segments is 270 m in diameter. The roof of the segment appears to be intact, with no sign of faults or slumping. A minimum roof thickness for this tube is 135 m.

Rille C, unnamed by previous mappers, begins at Gruithuisen K, a kidney-shaped depression at the top of the photograph (Fig. 2). This rille trends north-northwest, parallel to the major ridges in Procellarum, and is slightly sinuous. It is interpreted to be Eratosthenian in age (*Scott and Eggleton,* 1973). The highlands terrain just north of Gruisthuisen K is composed of primary and secondary impact material of the Iridum Crater and possible volcanic domes (*Scott and Eggleton,* 1973). Any or all of these

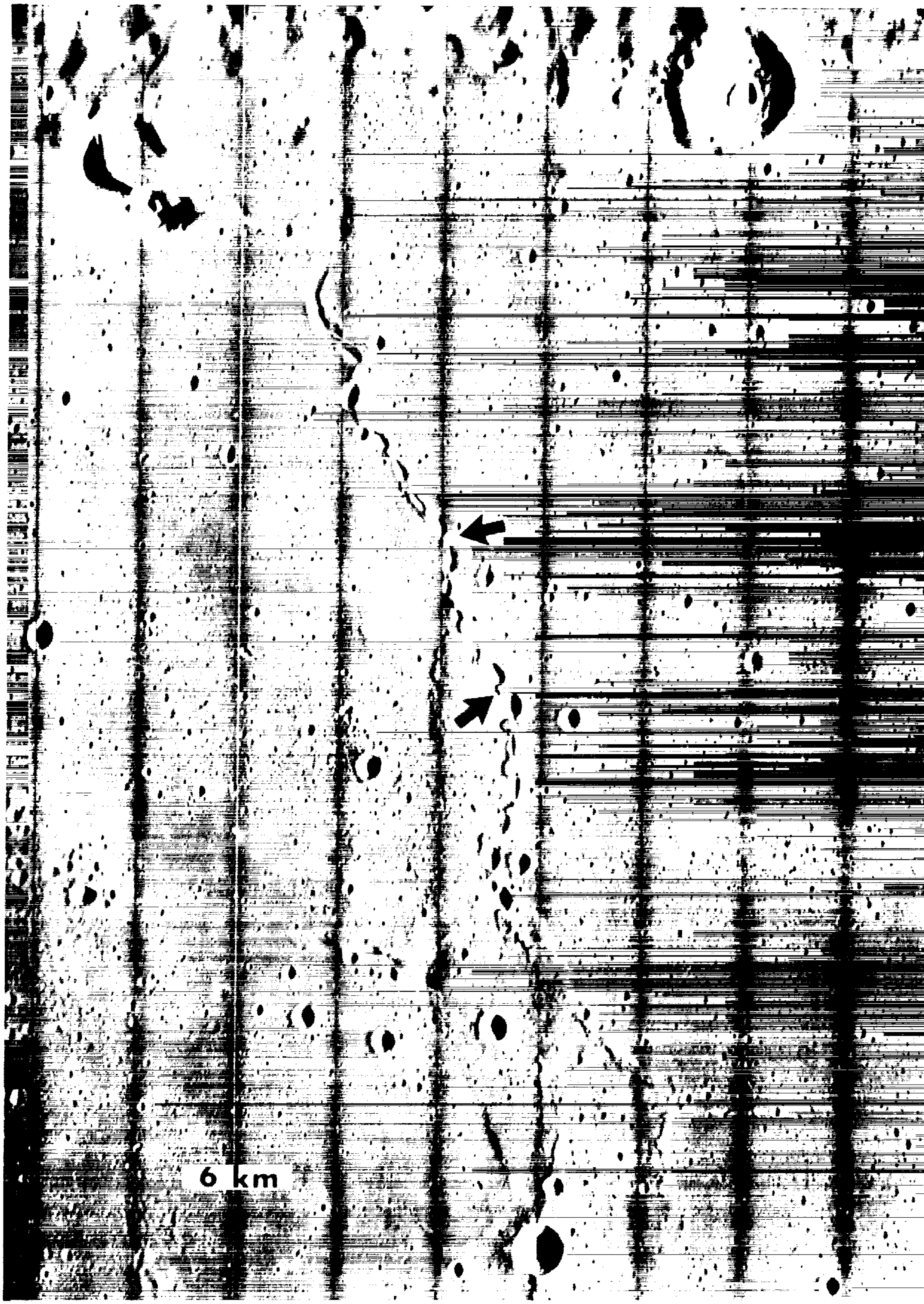

Fig. 2. This partially collapsed tube is located just west of the crater Gruithuisen. This is rille C as identified in Table 1 and is considered a prime candidate for finding an intact, vacant lunar lava tube. (LO-V-182-M)

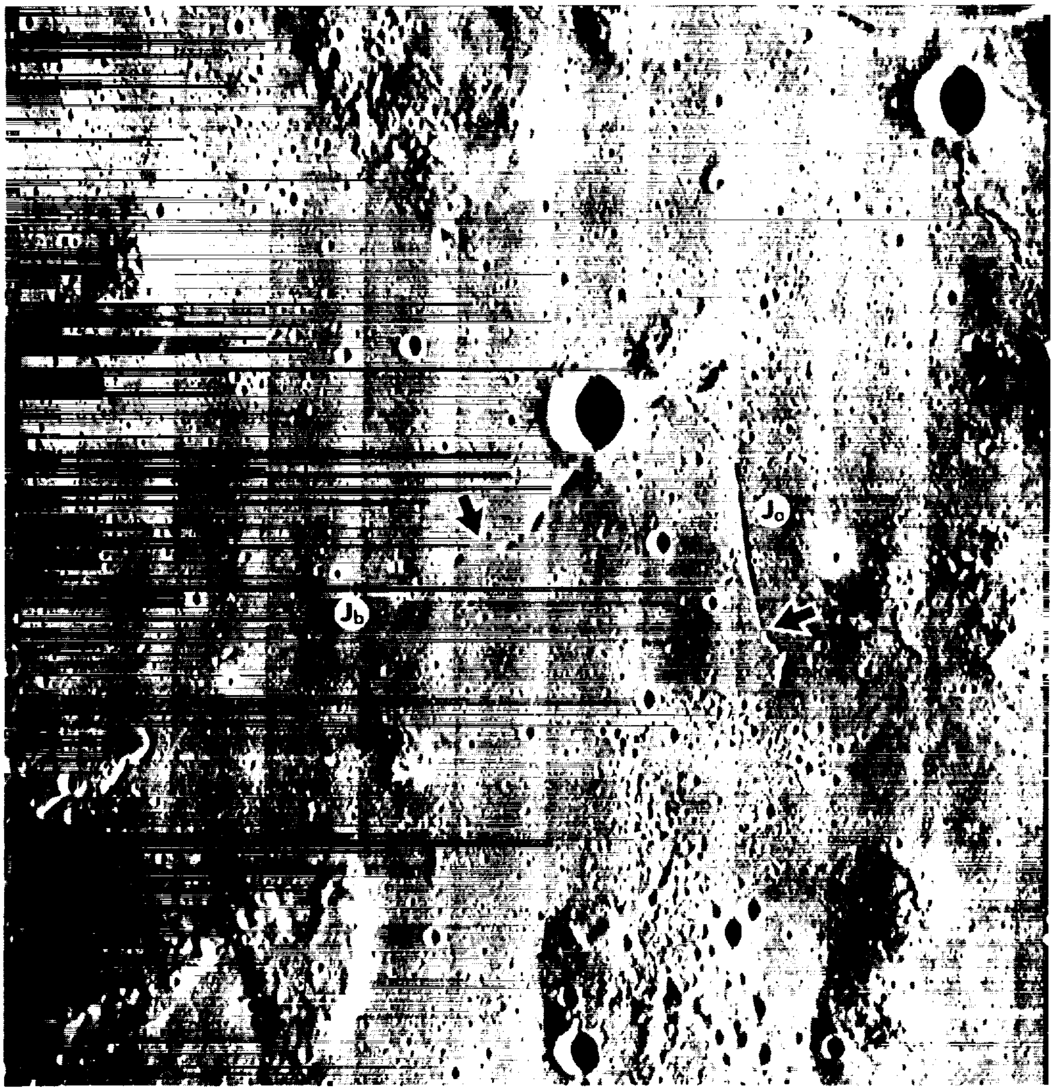

Fig. 3. Lava tube candidates in the Marius Hills region (J of Table 1 and Fig. 1). Two segments (J_a and J_b) meet at right angles at what may have been the source vent for them both. (LO-V-213-M)

tube segments would provide easy access to mare materials and the volcanic dome material just north of the rille.

Rille J, located in the Marius Hills region of Oceanus Procellarum (Figs. 1 and 3; Table 1), also exhibits extremely strong evidence for the existence of an intact lava tube. Rille J is actually a combination of two rilles that meet at right angles at an irregularly shaped depression that may have served as a common source vent (Fig. 3). Rille J_a trends north to south, is 15.5 km long, and is divided into seven segments, four of which were measured. The tube lengths range between 130 and 250 m, and the widths between 250 and 340 m. The maximum crater depth along rille J_a is 20 m, suggesting a minimum roof thickness in this area of 40 m.

Rille J_b is 42 km long, trends southwest, and is broken into more than 15 segments. Eleven of the best defined segments were measured. Here the lengths of the tube segments range between 250 and 1630 m and the widths vary between 270 and 480 m. One other tube segment along this rille varies radically from the others (J9, Table 1). This segment is 7370 m long, 270 m wide, and has a maximum superposed crater diameter of 250 m, or a minimum roof thickness of 125 m. The existence of a tube segment here is considered a strong possibility as the tube does appear to continue to the southwest. Whether or not an open tube is continuous along the entire length is difficult to determine. Closer inspection of this segment will be necessary in order to accurately determine its potential for human habitation.

The Marius Hills region is the product of a complex volcanic history. Northeast- and northwest-trending wrinkle ridges represent the intersection of two major structural trends (*Schultz,* 1976), while thick layers of lava flows and extrusive domes mark the sites of previous major volcanic activity. The Hills themselves are interpreted to be composed of pyroclastic and volcanic flow material that erupted and flowed over older mare material (*McCauley,* 1967). Both the pyroclastic debris and mare basalts in this region could be used as sources of lunar resources.

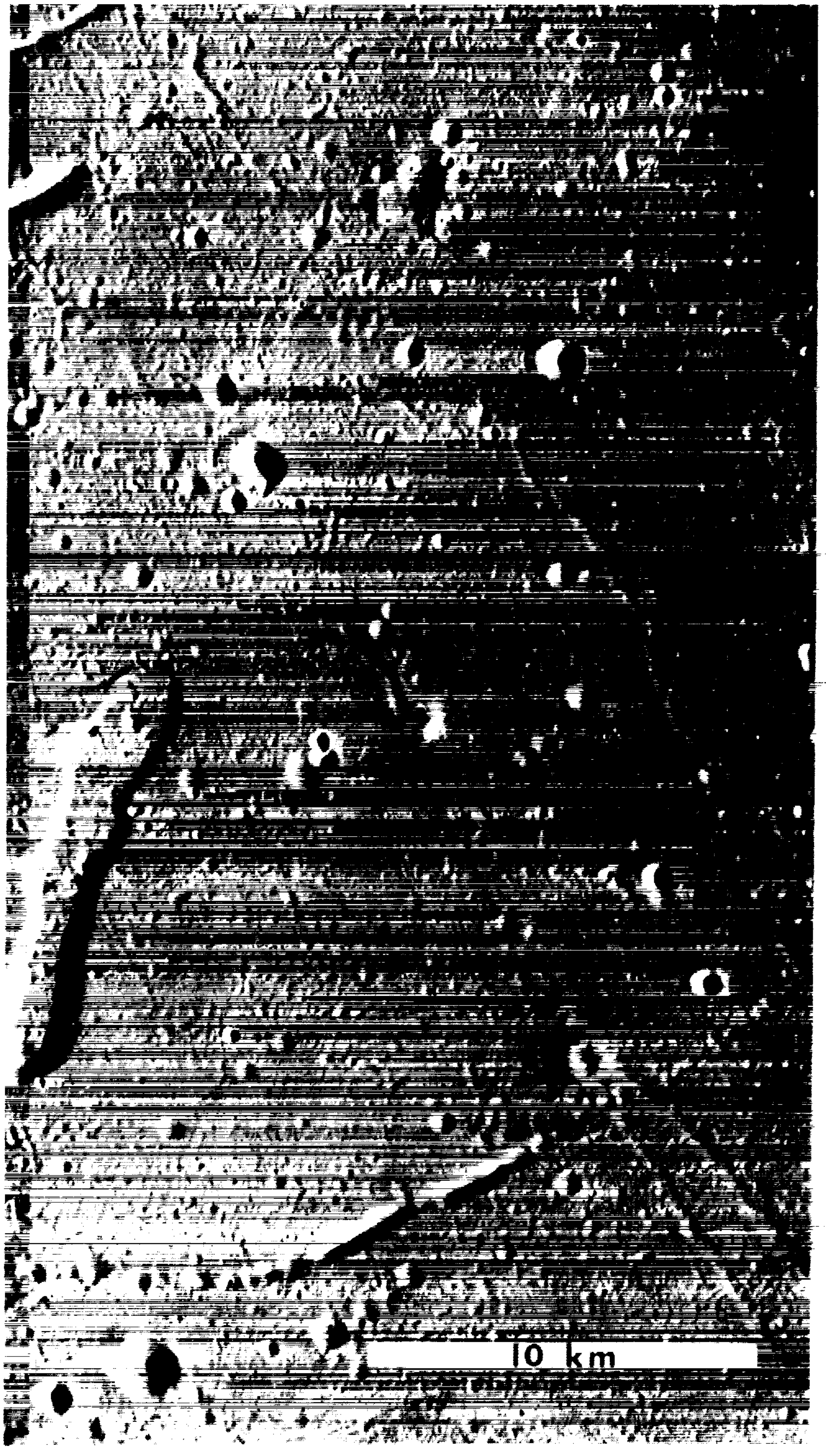

Fig. 4. Rille O, a prime candidate for having an intact lunar lava tube is located southwest of the Apollo 17 landing site in southern Mare Serenitatis. (AS17-2317)

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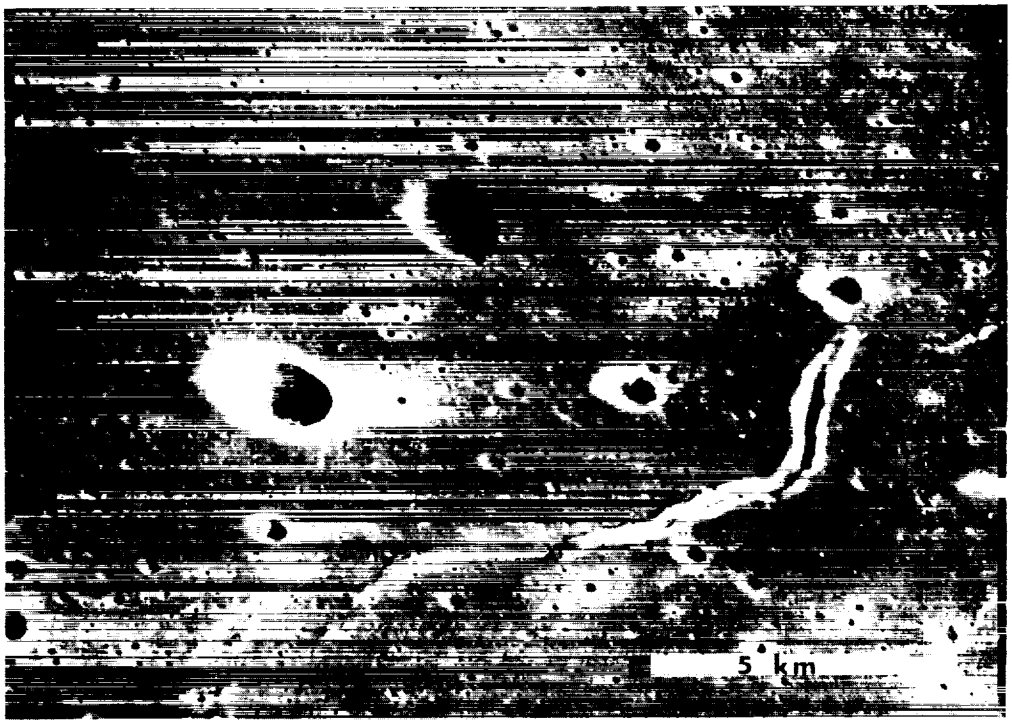

Fig. 5. Rille R, located in eastern Mare Serenitatis. This sinuous rille has retained its original levee walls and is roofed over at both its proximal and distal ends. The source crater is located at the right end of the rille, with the flow direction from right to left. (AS15-9309)

Southern Procellarum

One strong tube candidate was found in a complex of rilles north of Mare Humorum (Q in Fig. 1; Table 1). A number of very sinuous rilles that trend north-northeast are located near the boundary between Mare Humorum and Oceanus Procellarum. A tube segment was identified along one of these rilles. The tube segment is 1650 m long and 440 m wide. No measurable superposed craters were identified on the photographs available for this tube segment. The isolated, noncentral location, and presence of just one tube segment at this locality all argue against recommending this site to house the lunar base.

Northern Imbrium

Three tube segments were found along one sinuous rille in the northern portion of Mare Imbrium. This rille trends northeast and is located near the mouth of Alpine Valley (P in Fig. 1; Table 1) and, like the others, is surrounded by mare material. The three tube segments range in length from 840 to 1680 m and in width from 210 to 290 m. The maximum crater diameter supported by one of the segments is 170 m, with a potential roof thickness of 85 m. While this area may offer a variety of resources, and parts of it are flat, we do not feel that it would make an ideal locality to house the lunar base because of its noncentral location.

Serenitatis/Tranquillitatis

Five rilles with probable intact lava tubes were identified in the Serenitatis/Tranquillitatis region. Two other rilles in this region were also noted as potential candidates; however, the poor quality of the available photographs prohibited us from accurately measuring them and assessing their true potential. These two rilles should be inspected, however, when better data are available.

Of the five rilles identified, two exhibit extremely strong evidence for having intact tube segments (O in Figs. 1 and 4; R in Figs. 1 and 5; Table 1). Rille O is >30 km long, is located southwest of the Apollo 17 landing site, and is divided into more than 13 segments, 5 of which were measured. This rille trends east-northeast, parallel to the structural trend in this portion of southern Mare Serenitatis. The segments measured vary between 320 and 880 m long, are 460 to 510 m wide, and support a maximum crater 80 m in diameter. The minimum roof thickness for the longer tube segment is 40 m.

This locality offers major advantages for siting the lunar base. Not only is the region relatively flat, offering easy access to many areas, it is reasonably close to the Apollo 17 landing site (high-Ti mare basalt regolith), as well as a major regional dark mantle deposit of pyroclastic origin. This site may provide a good source for building materials as well as lunar resources such as Fe, Ti, Al, and K.

Rille R, located on the eastern edge of Mare Serenitatis, is 15 km long and 1.0 km wide. The rille originates at an acircular crater approximately 1.3 km in diameter and follows a general south-southwest trend. Channel levees mark the edge of the channel along a 6-km stretch between the two segments. The shorter of the two tube segments is 1.32 km long, and the other is 6.5 km long. The largest crater superposed on this rille was 0.66 km along the longest segment. The minimum roof thickness for this tube segment is 330 m.

Many other lunar rilles were examined but were considered less viable candidates for several reasons. The rejected rilles exhibited no uncollapsed sections and/or the adjacent and superposed craters were too large. The possibility that structurally sound lava tubes exist at these and other locations on the nearside of the Moon cannot be ruled out completely, however, until much more detailed analyses are completed.

ROLE OF LAVA TUBES FOR AN ADVANCED MANNED LUNAR BASE

There are many advantages to using intact lunar lava tubes as the site for a manned lunar base. The natural tube roof provides protection from cosmic radiation. The protected area offered by the intact tubes would provide storage facilities, living quarters, and space for industrial production. The constant temperature of around 20°C (*Hörz,* 1985; W. Mendell, personal communication, 1987) is conducive to many projects and experiments, and could be altered to maintain a controlled environment for a variety of experiments as well as comfortable living conditions.

Unused or uninhabitable portions of lava tubes would also provide an additional disposal facility for solid waste products generated from the manned lunar base. Biological and industrial (i.e., mining, construction) waste may be safely discarded within these structures without diminishing the vista of the lunar surface. This method of waste disposal may provide an alternative to the crater filling, burial, or hiding-in-the-shade methods proposed by *Ciesla* (1988).

Many potential resources may be located in the vicinity of lava tubes or complexes. For example, lunar pyroclastic deposits are known to be associated with some source vents for the lunar sinuous rilles and lava tubes (*Coombs et al.,* 1987). The black spheres that dominate some regional pyroclastic deposits are known to be rich in ilmenite (*Heiken et al.,* 1974; *Pieters et al.,* 1973, 1974; *Adams et al.,* 1974). These ilmenite-rich pyroclastics may in turn be a source of Ti, Fe, and O. Also, pyroclastics and regolith found in the vicinity of some of our tube candidates may be a good source for S as well as other volatile elements. Sulfur could be used as a propellant, as a fertilizer, and in industrial chemistry as suggested by *Vaniman et al.* (1988). The volcanic material may also be used as construction materials. Big pieces of rock may be utilized as bricks while small pyroclastic debris may be incorporated in cement compounds or broken down into individual elements.

There is one major problem to consider when planning the use of a lava tube to house the first manned lunar base. That is the difficulty in confirming, absolutely, that a tube does in fact exist in a particular spot and determining what its exact proportions are. Efforts are currently underway to determine a method for identifying evacuated, intact, lava tubes on the Moon. Such methods might include initial gravity and seismic surveys with later drilling and/or "lunarnaut" and rover reconnaissance, or a portable radar system. Several such methods are being tested now in Hawaii to identify lava tubes on Oahu and the Big Island. The construction of highly detailed geologic and topographic maps for the lunar areas also would greatly enhance the efforts to accurately determine the locations, dimensions, and existence of the lunar lava tubes. Thus, until a tube or tube system is positively identified on the lunar surface, mission planners should not rely on the presence of a lava tube when designing the first manned lunar habitat. This paper, rather, points out the strong possibility of there being an intact open tube system on the lunar surface that could be incorporated into future plans for an advanced, manned lunar base.

CONCLUSION

We conclude that lava tubes were formed on the Moon and that the probability of finding an intact, open tube segment that would be suitable for housing a permanent lunar base is quite high. Criteria were established for identifying intact lava tube segments. A survey of all known sinuous rilles and channels and other selected volcanic features was conducted in an effort to locate lava tube segments on the lunar surface. All available Lunar Orbiter and Apollo orbital photography of these features was utilized. We measured 67 tube segments associated with 20 rilles in 4 mare regions on the lunar nearside. It was determined that these tube segments are likely to be intact and open. Each tube segment was evaluated for suitability for use as part of an advanced lunar base. The results of this evaluation are given in Table 1. Nine tube segments associated with three separate rilles were given the highest ranking. We consider these nine segments to be prime candidates for use as part of an advanced lunar base. Finally, it should be pointed out that the emphasis in this paper was placed on relatively large lava tubes because evidence could be obtained from existing lunar photography. Numerous much smaller tubes may be present on the lunar surface, however, and some of these may also be useful in the lunar base initiative.

More analysis of the tubes discussed here is needed before an adequate selection can be made of a specific lunar lava tube to house a manned lunar base. One thing that may be done to help identify an intact lava tube or series of tubes would be construction of detailed geologic maps, topographic maps, and orthotopophotographic maps for areas showing potential for intact, vacant lava tubes. Also, further data are needed to adequately confirm the presence of open channels and tubes. These data might include radar, gravity, active and passive seismic experiments, rover and "lunarnaut" reconnaissance, and drilling. Once the proper tube is located, the possibilities for its use are numerous.

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REFERENCES

Adams J. B., Pieters C. M., and McCord T. B. (1974) Orange glass: Evidence for regional deposits of pyroclastic origin on the Moon. *Proc. Lunar Sci. Conf. 5th,* pp. 171-186.

Brown O. D. R. and Finn J. C. (1962) The permanent lunar base—Determination of biological problem areas. In *Aerospace Medical Div., Aerospace Medical Research Labs./6570TH/.* Wright-Patterson AFB, OH, Biologists for Space Systems Symposium, Final Report, pp. 233-241.

Carr M. H. (1974) The role of lava erosion in the formation of lunar rilles and martian channels. *Icarus, 22,* 1-23.

Ciesla T. M. (1988) The problem of trash on the Moon (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 54. Lunar and Planetary Institute, Houston.

Coombs C. R., Hawke B. R., and Wilson L. (1987) Geologic and remote sensing studies of Rima Mozart. *Proc. Lunar Planet. Sci. Conf. 18th*, pp. 339-354.

Coombs C. R., Hawke B. R., and Wilson L. (1989) Kauhako crater and channel, Kalaupapa, Molokai: A terrestrial analog to lunar sinuous rilles. *Proc. Lunar Planet. Sci. Conf. 20th*, pp. 195-206.

Cruikshank D. P. and Wood C. A. (1972) Lunar rilles and Hawaiian volcanic features: Possible analogues. *Moon, 3,* 412-447.

Greeley R. (1971) Lunar Hadley rille: Considerations of its origin. *Science, 172,* 722-725.

Greeley R. (1987) The role of lava tubes in Hawaiian volcanoes. In *Volcanism in Hawaii*, pp. 1589-1602. U.S. Geol. Surv. Prof. Pap. 1350.

Greeley R. and King J. S. (1977) *Volcanism of the Eastern Snake River Plain, Idaho: A Comparative Planetary Geology Guidebook.* NASA CR-154621. 308 pp.

Heiken G. H., McKay D. S., and Brown P. W. (1974) Lunar deposits of possible pyroclastic origin. *Geochim. Cosmochim. Acta, 38,* 1703-1718.

Henderson C. W. (1962) Man and lunar housing emplacement. *American Rocket Society Annual Meeting 17th, and Space Flight Exposition,* Los Angeles, CA, Nov. 13-18, Paper 2688-62, ARS Paper 62-2688, 7 pp.

Heppenheimer T. A. (1985) Achromatic trajectories and the industrial scale transport of lunar resources. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 155-168. Lunar and Planetary Institute, Houston.

Hörz F. (1985) Lava tubes: Potential shelters for habitats. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 405-412. Lunar and Planetary Institute, Houston.

Hulme G. (1973) Turbulent lava flow and the formation of lunar sinuous rilles. *Mod. Geol., 4,* 107-117.

Hulme G. (1982) A review of lava flow processes related to the formation of lunar sinuous rilles. *Geophys. Surveys, 5,* 245-279.

Klein F. W., Koyanaki R. Y., Nakata J. S., and Tanigawa W. R. (1987) The seismicity of Kilauea's magma system. In *Volcanism in Hawaii,* pp. 1019-1185. U.S. Geol. Surv. Prof. Pap. 1350.

Macdonald G. A., Abott A. T., and Peterson F. L. (1983) *Volcanoes in the Sea.* Univ. of Hawaii, Honolulu. 517 pp.

Masursky H., Colton, G. W., El-Baz F. (1978) *Apollo Over the Moon: A View from Orbit.* NASA SP-362. 255 pp.

McCauley J. F. (1967) Geologic map of the Hevelius Quadrangle of the Moon. *U.S. Geol. Surv. Map I-491.*

Oberbeck V. R., Quaide W. L., and Greeley R. (1969) On the origin of lunar sinuous rilles. *Mod. Geol., 1,* 75-80.

Oberbeck V. R., Aoyagi M., Greeley R., and Lovas M. (1972) Planimetric shapes of lunar rilles. In *Apollo 16 Preliminary Science Report,* Part Q, pp. 29-80 to 29-88. NASA SP-315.

Oberst J. and Nakamura Y. (1988) A seismic risk for the lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 183. Lunar and Planetary Institute, Houston.

Peterson D. W. and Swanson D. A. (1974) Observed formation of lava tubes during 1970-1971 at Kilauea Volcano, Hawaii. *Studies in Speleology, 2,* Part 6, 209-222.

Pieters C. M., McCord T. B., Zisk S. H., and Adams J. B. (1973) Lunar black spots and the nature of the Apollo 17 landing area. *J. Geophys. Res., 78,* 5867-5875.

Pieters C. M., McCord T. B., Charette M. P., and Adams J. B. (1974) Lunar surface: Identification of the dark mantling material in the Apollo 17 soil samples. *Science, 183,* 1191-1194.

Pike R. J. (1976) Crater dimensions from Apollo data and supplemental sources. *Moon, 12,* 463-477.

Schaber G. G., Boyce J. M., and Moore H. J. (1976) The scarcity of mappable flow lobes on the lunar maria: Unique morphology of the Imbrium flows. *Proc. Lunar Sci. Conf. 7th,* pp. 2783-2800.

Schultz P. H. (1976) *Moon Morphology, Interpretations Based on Lunar Orbiter Photography.* Univ. of Texas, Austin. 626 pp.

Schröter J. H. (1788) In *Astronomisches Jahrbuch für das Jahr 1791,* p. 201. Astronomisches Rechneninstitut.

Scott D. H. and Eggleton R. E. (1973) Geologic Map of the Rümker Quadrangle of the Moon. *U.S. Geol. Surv. Map I-805.*

Vaniman D., Pettit D., and Heiken G. (1988) Uses of lunar sulfur (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 244. Lunar and Planetary Institute, Houston.

Wentworth C. K. and MacDonald G. A. (1953) Structures and forms of basaltic rocks in Hawaii. *Geol. Surv. Bull., 994,* 98.

Wilson L. and Head J. W. (1980) The formation of eroded depressions around the sources of lunar sinuous rilles: Theory (abstract). In *Lunar and Planetary Science IX,* pp. 1260-1262. Lunar and Planetary Institute, Houston.

Wilson L. and Head J. W. (1981) Ascent and eruption of basaltic magma on the Earth and Moon. *J. Geophys. Res., 86,* 2971-3001.

A SEISMIC RISK FOR THE LUNAR BASE

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Shallow moonquakes, which were discovered during observations following the Apollo lunar landing missions, may pose a threat to lunar surface operations. The nature of these moonquakes is similar to that of intraplate earthquakes, which include infrequent but destructive events. Therefore, there is a need for detailed study to assess the possible seismic risk before establishing a lunar base.

INTRODUCTION

Contrary to the common belief of the pre-Apollo era, a belief that unfortunately is still found in today's popular literature, the Moon is by no means a seismically "dead" planet. It is an observed fact that quakes occur on the Moon. Such moonquakes, if sufficiently large, may damage man-made structures on the lunar surface, may disrupt the activities of the lunar inhabitants, and may cause injury or even loss of life.

From examination of the currently available observational data on lunar seismic activity, we believe that there is sufficient reason to be concerned about a possible seismic risk on the Moon. In this paper, therefore, we will present some basic lunar seismic data relevant to the question of a seismic risk and point out deficiencies in our knowledge on lunar seismicity that need to be resolved before we establish a base on the Moon.

LUNAR SEISMICITY

The data relevant to the seismic environment of the Moon were acquired from 1969 to 1977 during the operational period of the seismic station network established during the Apollo lunar landing missions. Of the two types of seismographs used at each station for covering two different spectral ranges of seismic waves, the long-period seismographs recorded most of the seismic signals relevant to the global seismicity. More than 12,000 seismic events were detected by these instruments (*Latham et al.,* 1970; *Nakamura et al.,* 1982). The majority of these events were tidally induced moonquakes of body-wave magnitude m_b less than 3 (energy release $E < 10^6$ J) deep in the lunar interior. These deep moonquakes are not likely to pose any threat to lunar surface operations. Next in abundance were the seismic events originating from impacts of meteoroids on the lunar surface. The hazard of these impacts and their effects on the lunar base activities require separate consideration and will not be evaluated in this paper.

The most important in terms of seismic hazard is a class of moonquakes that occurs at shallow depths. They are believed to be of tectonic origin and are likely to represent recurrent release of thermoelastic stresses that have accumulated in the outer zone by global cooling and contraction of the Moon (*Nakamura et al.,* 1979; *Binder and Lange,* 1980). Although less frequent than other types of seismic sources, they constitute the most intense seismic events on the Moon. The largest of them detected during the eight years of observation had an estimated body-wave magnitude m_b greater than 5 ($E > 6 \times 10^{10}$ J) (*Nakamura et al.,* 1974, 1979; *Nakamura,* 1977; *Oberst,* 1987).

We assign body-wave magnitudes to moonquakes by equating the estimated total amount of energy released at the source in the same way as for earthquakes: $m_b = (\log E + 1.2)/2.4$ (*Richter,* 1958, p. 365). The effect of a moonquake on the lunar base, however, may be quite different from what would be expected for an earthquake of equal magnitude, as will be discussed later.

OCCURRENCE RATE OF SHALLOW MOONQUAKES

Twenty-eight shallow moonquake events were identified during the eight years of seismic observations. Based on the size distribution of these 28 events and assuming that the shallow moonquakes occur randomly distributed both in time and space over the entire lunar surface, we estimate their occurrence rate to be

$$\log N = -1.8\, m_b + 7.8 \qquad (1)$$

where N is the cumulative number of shallow moonquakes having body-wave magnitudes greater than m_b, occurring within an area of 10^6 km^2 per year (Fig. 1). From this relationship we estimate that the chances for a lunar base at a randomly chosen site to experience a shallow moonquake of, for example, m_b greater than 4.5 ($E > 4 \times 10^9$ J) within a range of 100 km are approximately one in 400 years.

The average seismic energy release for the whole Moon from the shallow events is estimated to be about 10^{14} J per year (*Nakamura,* 1980). This is several orders of magnitude lower than the rate of seismic energy released for the entire Earth, estimated to be about 10^{18} J per year (*Kanamori,* 1977). However, this is not surprising, because the overwhelming majority of earthquakes are concentrated along active plate margins, which do not exist on the Moon. In fact, lunar seismicity is very similar to the terrestrial seismicity if the seismically active regions along the plate margins are excluded (*Nakamura,* 1980). The estimated occurrence rate for shallow moonquakes [equation (1)] is indeed very similar to the approximate terrestrial intraplate seismicity for the central U.S. (Fig. 1; *Nuttli,* 1979).

Much about a possible seismic risk on the Moon can be learned from studies of these intraplate earthquakes. Intraplate earthquakes occur preferentially in regions of preexisting weakness in the lithosphere (*Sykes,* 1978). However, because of their long recurrence intervals, they often occur in places where such weakness is unrecognized, and thus large earthquakes are

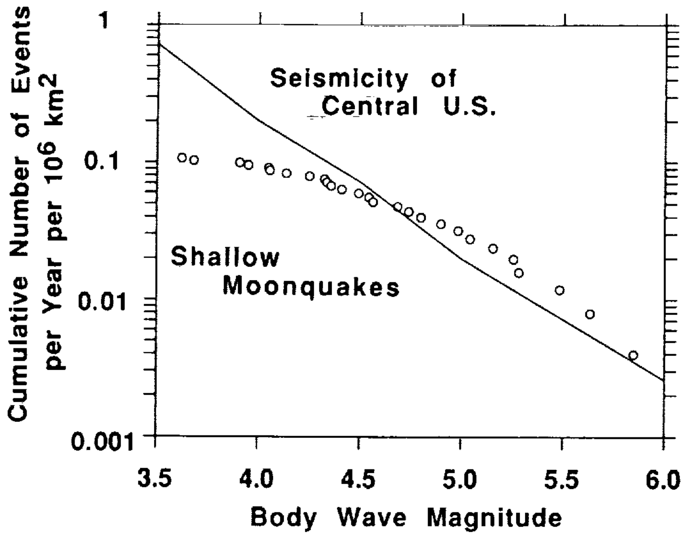

Fig. 1. Magnitude-frequency relationships of shallow moonquakes (open circles) and intraplate earthquakes (line). The moonquake body-wave magnitudes are computed from the estimated energy release of the 28 events observed during the 8 years of operation of the lunar seismic network. The number of observed events has been converted to the rate of occurrence within a unit area and time. Sampling is incomplete at small magnitudes, because amplitudes of small-magnitude events occurring at far distances fall below the detection threshold of the seismometers. The earthquake data are from *Nuttli* (1979).

unexpected. Three of the strongest earthquakes ever reported within the contiguous U.S. did not occur near the margin of the Pacific plate but occurred instead in the Mississippi valley near New Madrid, Missouri, in 1811 and 1812. These unexpected and most devastating intraplate events had estimated body-wave magnitudes of 7.1, 7.2, and 7.4 (*Nuttli,* 1973a). Another large intraplate event occurred in Charleston, South Carolina, in 1886, and had a body-wave magnitude of 6.7 (*Bollinger,* 1983). We cannot rule out the possibility that similar very large seismic events also occur on the Moon but were not detected during the short duration of our observations.

OTHER FACTORS THAT AFFECT SEISMIC RISK

There are several properties of shallow moonquakes other than the rate of occurrence that have direct bearing on the seismic risk. For some of these, direct observational data are still lacking. Moreover, the unique lunar environment that affects the propagation of seismic waves through the Moon also requires special consideration.

Spatial Distribution

We presume that certain regions of the lunar surface are more prone to shallow moonquake activities than others. Such regions may be recognized by statistically significant spatial concentration of observed shallow moonquakes. It is important to identify such regions before we set up a lunar base because the intensity of seismic risk is directly related to the proximity of the base to such concentrations of events.

Nakamura et al. (1979) note that moonquakes appear to occur preferentially near the margins of large mare basins, which may represent zones of weakness in the lunar lithosphere. In contrast, *Binder and Gunga* (1985) and *Binder and Oberst* (1985) suggest that shallow moonquakes are associated with young thrust fault scarps in the lunar highlands. Unfortunately, the number of detected shallow moonquakes to date is not large enough to make a statistically meaningful correlation with any topographic or selenologic feature.

Focal Depths

The severity of ground motion and, consequently, the potential danger to a lunar base of a shallow moonquake depend directly on focal depth. Unfortunately, the observational data do not provide accurate focal depths of any of the detected shallow moonquakes. This is because all shallow moonquakes detected to date occurred far outside the seismic network, while accurate determination of focal depth requires observations at near ranges. Once we identify any region of high seismic activity on the Moon, we need to set up local seismic stations in order to determine precisely the focal depths of individual shallow moonquakes there.

Character of Ground Motion

Even on Earth it is difficult to predict the level of ground motion for an earthquake of given magnitude occurring at a given range (see *Heaton and Hartzell,* 1988, for a review). The difficulty is further compounded for the Moon because of the vastly different environment. The observed lunar seismic data clearly show that the character of ground motion caused by a shallow moonquake is significantly different from that of an earthquake of equal magnitude at equal range for several reasons.

The source spectra of shallow moonquakes generally contain much more energy at high frequencies than those of earthquakes of comparable total energy release (*Oberst,* 1987). Since the response of some man-made structures to the ground motion near the epicenter is highly dependent on frequency, a significant difference in potential damage to the structures is expected between earthquakes and moonquakes.

Seismic waves are much less attenuated in the Moon than in the Earth (*Nakamura and Koyama,* 1982). As a result, seismic energy from a moonquake is likely to spread much farther from the epicenter, thus affecting a wider area of the Moon than would be expected for an earthquake of comparable magnitude. Such a difference is observed between earthquakes occurring in the eastern and western parts of the U.S. At a given magnitude, earthquakes near the east coast are often felt over an area up to 100 times larger than those in the western U.S., since in the east the more stable tectonic setting allows more efficient transmission of seismic waves (*Nuttli,* 1973b).

Another distinct difference between the lunar and the terrestrial seismic environment is the more intense scattering of seismic waves in the highly fractured near-surface zone of the Moon than on Earth. Seismic signals on Earth are relatively short and impulsive. In contrast, seismic signals from moonquakes show a very gradual increase of intensity within the first several minutes followed by an extremely slow decay, often lasting for several hours. The dispersion of seismic energy over a very long time span may reduce potential structural damage to the lunar base.

The intense seismic scattering in the near-surface zone and the consequent destruction of coherent surface reflections prevent efficient long-range propagation of surface waves on the Moon.

Since surface waves, which decay more slowly with distance than body waves, are often primarily responsible for earthquake damage, the absence of surface waves may be of benefit to the lunar base.

CONCLUSIONS

In spite of considerable scientific literature on lunar seismicity, the general public is not fully aware that the Moon is tectonically quite active. The seismic data now available suggest that shallow moonquakes may pose a significant risk to the operation of the lunar base. Studies of the seismic environment are common practice before starting important construction projects such as bridges, dams, and power plants on Earth. Such a study may also be indispensable for the Moon before establishing a base there.

The currently available data are not sufficient to firmly establish the seismic risk for the lunar base. In particular, further studies are needed in two general areas: (1) a better characterization of the shallow moonquake sources, including their focal depths and spatial distribution; and (2) the effect of the moonquake ground motion, which is significantly different from earthquake ground motion, to structures and inhabitants. For the former, establishment of a global network of seismic stations deployed by, for example, penetrators or soft landers may be required. Seismic observations during the initial phase of the lunar base operations will be beneficial for furthering our knowledge on the seismicity and seismic risk.

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REFERENCES

Binder A. B. and Gunga H.-C. (1985) Evidence for an initially totally molten moon: Young thrust fault scarps in the highlands. *Icarus, 63,* 421-441.

Binder A. B. and Lange M. A. (1980) On the thermal history, thermal state, and related tectonism of a moon of fission origin. *J. Geophys. Res., 85,* 3194-3208.

Binder A. B. and Oberst J. (1985) High stress shallow moonquakes: Evidence for an initially totally molten moon. *Earth Planet. Sci. Lett., 74,* 149-154.

Bollinger G. A. (1983) Speculations on the nature of seismicity at Charleston, South Carolina. In *Studies Related to the Charleston, South Carolina Earthquake of 1886—Tectonics and Seismicity* (G. S. Gohn, ed.), pp. T1-T11. U.S. Geol. Surv. Prof. Pap. 1313.

Heaton T. H. and Hartzell S. H. (1988) Earthquake ground motions. *Annu. Rev. Earth Planet. Sci., 16,* 121-145.

Latham G. V., Ewing M., Press F., Sutton G., Dorman H. J., Nakamura Y., Toksöz N., Wiggins R., Derr J., and Duennebier F. (1970) Passive seismic experiment. *Science, 167,* 455-467.

Nakamura Y. (1977) HFT events: Shallow moonquakes? *Phys. Earth Planet. Inter., 14,* 217-223.

Nakamura Y. (1980) Shallow moonquakes: How they compare with earthquakes. *Proc. Lunar Planet. Sci. Conf. 11th,* pp. 1847-1853.

Nakamura Y. and Koyama J. (1982) Seismic Q of the lunar upper mantle. *J. Geophys. Res., 87,* 4855-4861.

Nakamura Y., Dorman H. J., Duennebier F. K., Ewing M., Lammlein D., and Latham G. (1974) High-frequency teleseismic events. *Proc. Lunar Sci. Conf. 5th,* pp. 2883-2890.

Nakamura Y., Latham G. V., Dorman H. J., Ibrahim A. K., Koyama J., and Horvath P. (1979) Shallow moonquakes: Depth, distribution and implications as to the present state of the lunar interior. *Proc. Lunar Planet. Sci. Conf. 10th,* pp. 2299-2309.

Nakamura Y., Latham G. V., and Dorman H. J. (1982) Apollo lunar seismic experiment—Final summary. *Proc. Lunar Planet Sci. Conf. 13th,* in *J. Geophys. Res., 87,* A117-A123.

Nuttli O. W. (1973a) The Mississippi Valley earthquakes of 1811 and 1812: Intensities, ground motion and magnitudes. *Bull. Seism. Soc. Am., 63,* 227-248.

Nuttli O. W. (1973b) Seismic wave attenuation and magnitude relations for eastern North America. *J. Geophys. Res., 78,* 876-885.

Nuttli O. W. (1979) Seismicity of the central United States. In *Geological Society of America Review of Engineering Geology Vol. IV* (A. W. Hatheway and C. R. McClure Jr., eds.), pp. 67-93. Geological Society of America, Boulder.

Oberst J. (1987) Unusually high stress drops associated with shallow moonquakes. *J. Geophys. Res., 92,* 1397-1405.

Richter C. F. (1958) *Elementary Seismology.* Freeman, San Francisco. 768 pp.

Sykes L. R. (1978) Intraplate seismicity, reactivation of preexisting zones of weakness, alkaline magmatism, and other tectonism post-dating continental fragmentation. *Rev. Geophys. Space Phys., 16,* 621-688.

3 / Lunar Surface Archbiticture and Construction

LUNAR ARCHITECTURE AND URBANISM

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Human civilization and architecture have defined each other for over 5000 years on Earth. Even in the novel environment of space, persistent issues of human urbanism will eclipse, within a historically short time, the technical challenges of space settlement that dominate our current view. By adding modern topics in space engineering, planetology, life support, human factors, material invention, and conservation to their already renaissance array of expertise, urban designers can responsibly apply ancient, proven standards to the exciting new opportunities afforded by space. Inescapable facts about the Moon set real boundaries within which tenable lunar urbanism and its component architecture must eventually develop.

THE LONG VIEW

Many decades still insulate us temporally from true lunar urbanism. Indeed, many years will pass before even inchoate lunar architecture is realized. Why, then, examine a field so embryonic that its real features cannot yet be known? Three reasons motivate this essay.

First, given the proof of Project Apollo, no one could defensibly pretend that human expansion to other planets is impossible. Even many nonspecialists are already thinking and anxious about prospects for lunar civilization; their inevitable projections will be most productive if grounded realistically in a few inescapable facts that will constrain life on the Moon.

Second, those hoping professionally to design the built lunar environment tend to be either space engineers who know little about urban history, or architects who know little about space. Responsible lunar planners, however, must be versed in both worlds. Preparing rigorously for that joint future will take much time, and appreciating the depth and range of both fields is a first step.

Third, and most central, refining the direction of the path that will bridge present thinking to future history depends on setting goals from the beginning. Without some tangible idea of what the far future must, should, and might be, we have no sound basis for making the many immediate decisions along our way toward it. Now is the time to begin earnest discussion of how people will use Earth's moon. Acknowledging eventual facts of offworld urbanism can save resources and, finally, remorse.

TRAVELING, STAYING, AND LIVING

To begin, we draw distinctions among three human activities, each of which has a special role to play in the growth of space civilization: traveling, staying, and living. Space architecture so far has been entirely vehicular, based on components launched from Earth. Atmospheric flight governs their form from the outside in. Like trucks and vans, they only grudgingly permit concerted activity, being cramped, noisy, smelly, and too inertially jittery to permit precision work. The interior human environment of such capsules, shuttles, and modules is purloined from the available methods and familiar hardware of earlier atmospheric flight vehicles.

Because travel vehicles are inappropriate for lengthy stays, servicing longer missions with vehicular architecture requires either excrescent or modular approaches. The space shuttle uses the former, accommodating up to seven workers for roughly a week with ab-ware (Spacelab, Spacehab) installed in its capacious cargo bay. This allows but also enforces extensive ground support for every mission and is ultimately volume-limited. Mir, on the other hand, occupies the present stage in a modular space station lineage that began with Skylab. Distilling, as this approach does, the activities of traveling and staying allows much more growth, but is finally activity-limited by the dimensions of its units and connections.

A space architecture of linked, pressurized cylinders, even one that sprouts appendages and enormous exterior structures, is still vehicular in spirit. Such manned components on orbit are really like trains parked on sidings. Romanenko's recent 326-day record proves that, when specialized, such architecture can support individuals working and staying in space. While it is natural and common to envision even future space architecture based on this familiar vehicular vocabulary, however, only the very first stages of permanent construction in orbit or on planetary surfaces could in fact be sensibly vehicular.

Submarine and antarctic environments are frequently proffered as paradigms for space. Remote and hostile, all three are, after all, intrinsically deadly to people and thus require artifice to sustain life, promote efficiency, encourage conciliation, avoid conflict, and prevent disaster. From these urgent needs emerged human "factors" engineering, an attempt to quantify as completely as possible human behavior with the goal of designing more suitable environments. Such work holds great promise for enhancing our ability to stay in hostile places and will prove critical for long interplanetary manned missions and planetary outposts, which blur the boundary between traveling and staying. But Earth's oceans and poles, from which people eventually return, can only model space to a certain point.

Space cannot become the autonomous human economic arena widely regarded as inevitable until people establish their lives there. Travel time, expense, and risk will conspire to ensure that

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they eventually transform staying in space to living in space. Human living is an exceedingly complex activity, requiring much more than passably engineered accommodation because it includes all we do: working, resting, playing, and growing. Designing for living is a vastly messy problem, one not deeply solvable by Crays. People and their behavior cannot be reduced to factors in a numerical model of living. Instead, the sum of physical and abstract richness developed over all human history occupies the core and determines the aspects of human living.

The requirements and effects of environments that support human living are subtle and continue to be honed over millennia as society evolves. Manipulating those environments with skill and grace demands a fine multivariate balance that, as far into the future as we can defensibly see, only human experience and wisdom can feasibly provide, in space as on Earth. They demand in fact the practice of architecture.

FIRMNESS, COMMODITY, AND DELIGHT

We may define architecture succinctly as the professional activity of coordinating a set of specialty industries and services to make facilities that foster and enhance human living. To dissect the profession, we first need ways to evaluate its product. Then we can more critically review the specialties it coordinates.

Two millennia ago, the Roman Vitruvius proffered a clear, concise, and complete statement of the qualities defining good architecture: firmness, commodity, and delight. This tripolar standard covers anything that architecture can do or be. ***Firmness*** refers to structural integrity, appropriate material qualities, proper fabrication, and safety. Firmness addresses the question: Is it usable? ***Commodity*** subsumes all the ways a work of architecture serves the programmatic purpose for which it is built, accommodating the physical and abstract needs of its occupants and environment. Commodity addresses the question: Is it useful? ***Delight*** is often the diacritical signature of great architecture, frequently omitted in modern Western culture as a separable luxury. Delight addresses the subtle but penetrating question: Would people rather use this than other solutions? These three ancient principles apply to all ages and modes and styles of architecture, encapsulating distinct and complementary properties without any one of which architecture cannot be simultaneously structure, solution, and art.

At its best, architecture projects human values and aspirations; at the very least, it embodies human needs and behaviors. Because it depends on manipulating materials for human use, architecture has been called cynically the "second oldest" profession. The purview of architecture, even neglecting (as here) traveling and staying, is extensive and inclusive. We take all designed interfaces between human beings and their environment, from spoons to highways, gardens to sewers, and buildings, too, as architecture. Civil architecture, servicing and embodying human community, is convolved inextricably with civilization.

Archeologists generally define civilization as having begun about 5000 years ago in Mesopotamia, following two key inventions: writing and urbanism. The earliest applications of writing and urbanism must be considered artifacts of commerce, having used abstract yet practical formal design to facilitate efficient and dense intercourse. Subsequently, these permanently expressive media, both written and built, became intrinsically useful for encapsulating and stimulating human sensibilities. By thus transcending mere functionality, the recording arts of literature and architecture were born. Over the ages since then, civilization and its cultural expressions have continued to define each other iteratively. We cannot imagine "civilization" (from the Latin root for citizen) divorced from its creative artifacts.

The city is architecture's grandest product, a built armature within which throngs of people can arrange discrete but linked lives. As a tool permitting societal evolution, the city must first provide enduring organization and sustain the individual and collective needs of the people living in it. By accommodating simultaneously most of the conflicting, singular services its citizens desire, the city can enable a population density possible no other way. The synergy of that populace animates in turn a social organism much larger, more resourceful, and more consequential than any individual could be. It is this strength, this capacity, this influence available to a civic culture, that drives humans together to make cities wherever they live.

A civilization sustained enduringly and efficiently by its culture can upon that foundation achieve great things, advancing the reach of the human spirit. As we know, however, the extreme density encouraged by cities cannot alone guarantee greatness; urbanism often falls far short of both commodity and delight. Disease, violence, exploitation, environmental devastation, and spiritual impoverishment have historically accompanied unbridled concentrations of people. As contemporary physical limitations are approached, atavistic biological controls resurface in human populations. Certainly there is a vast gap between what is biologically tolerable for the human species and what is spiritually desirable for human civilization. Urban design tries to mitigate the negative aspects of dense populations while still fostering their priceless benefits.

Architecture necessarily occupies a central role in building civilization, by linking and reconciling otherwise isolated fields that can only make a firm, commodious, and delightful environment if combined coherently. Traditional specialties contributing to modern terrestrial architecture include subjects as sundry as human programming, historical study, abstract and representational modeling, psychology, structural engineering, law, materials testing and development, environmental control engineering, negotiation, construction management, engineering geology, economics, environmental study, and of course art.

Designers of cities must in addition address mass transportation, civic logistics, waste management and pollution avoidance, industrial production, crime, commerce, power sources and distribution, spectator events, communication networks and media, public recreation, resource conservation, death, park management, health maintenance, and defense.

Architects and urban planners try to satisfy simultaneously the needs of all these diverse subjects by manipulating the proportions, character, symbolism, and scale of material assemblages. In so doing they add incidentally to the long history of built human environments. Their central, coordinating effort remains invariant despite material and social features unique to time and place.

ANOTHER CHANCE

The time is the next century, and the place is cislunar space, particularly on and under the surface of Earth's moon. Until then and there, the vehicular nature of all space vessels ensures that their design can be influenced by only a skeleton (the human "factors") of the tremendous array of architectural issues. Poised still at the threshold of inhabiting the first truly new environment since the dawn of man, and having only essayed tentatively into it, we are now understandably preoccupied with technical

challenges. Keeping people alive and physically healthy still dominates all other problems of manned space activity.

Orbiting a few people has consumed the best engineering effort the twentieth century could muster. Enabling several people to stay in orbit simultaneously for many months, served by a complete Space Transportation System (STS) permitting travel throughout cislunar space, will be much more challenging and expensive. Leaving behind the sustenance and protection of Earth opens a level of interleaved technical problems quite beyond anything we have tackled so far. Logistically, sustaining large groups for long times inevitably demands some form of Controlled Ecological Life Support System (CELSS). Long microgravity stays might require prophylaxis, whether biochemical or inertial, against bone demineralization; and protracted travel at and beyond geosynchronous orbit (GEO) demands careful shielding against both constant and acute radiation fluence. Solving just these problems reliably and elegantly will keep us busy well into the next century.

Yet, once those problems *are* solved, even primitively, they will cease to pose the dominant obstacle to space civilization. We can safely assert that before multitudes of people begin living in space, more ancient architectural issues will have superseded the technical dilemmas of putting and keeping them there. Establishing an offworld urbanism that can provide the spectrum of amenities, stimulation, and cultural support that people require of cities anywhere presents a really tough problem, beside which our incipient engineering challenges pale. The social complexities introduced by hundreds, thousands, or even millions of people living in space must come to dominate everything else. Technically on the verge of being able to keep communities alive on the Moon, we have barely begun to prepare for solving the total architectural problem engaged by doing so.

Extant but unconcerted preparation takes three forms. First, and least useful, are utopian images arising from contemporary "colonization" studies, which attempt to paint a picture of space civilization by projecting inconsistent and peculiar details. In presenting rather fixed images, they reveal more about their creators than about life in space. Second are the uncounted ideas explored in vignetted detail by science fiction. Albeit often technically bankrupt, these bring to the study of human futures beyond Earth the important advantage of having been conceived by writers generally driven to explore implications and meaning, rather than ways and means. Finally, but unwittingly, the profession of terrestrial architecture is better prepared for solving the eventually important problems of living in space than is space engineering. Only dedicated human planners supported by millennia of professional experience can hope to avoid the mannerist traps of simple visions, while still tapping the vibrant storehouse of potential futures, to realize viable and inspiring cities in space.

We must exorcise the common presumption that architecture has a "humanizing" role to play in engineering that urban landscape. We accomplish more by reversing the notion: Space engineering will in fact be but a new tool in the ancient panoply of architectural practice. Lunar urbanism must after all follow the human needs of its citizens, according to principles that no new technology, no new environment, no new gimmicks are liable to change deeply. Engineering realities of building on the Moon will provide the vocabulary but neither the diction nor syntax of lunar urbanism. Recognizing that human space engineering must eventually be absorbed by the inclusive profession of architecture allows us to see just how it will expand that profession.

Any offworld urban design will require attention to all the "conventional" architectural and planning subjects listed earlier, plus advanced CELSS, radiation management, gravitational biology, interorbital elemental mining, biomass production, material recycling, and of course the full complement of traditional disciplines peculiar to spacecraft engineering, including astronautics, propulsion, vacuum thermal management, attitude control, teleoperation, vibration and noise suppression, artificial intelligence, and redundant safety. Finally, actual planetary architecture must address further the dominant issues of launch and landing, alien planetology including local geology, weather, diurnal cycle and gravity level, and wilderness preservation. Clearly anyone intending to become conversant enough in the components of lunar architecture to perform it rigorously, responsibly, and well has an awful lot to learn. As a professional culture, we are far from ready to take on the task we so glibly imagine. Just putting people on the Moon is indeed child's play compared to establishing a mature and noble lunar urbanism.

The sudden technical and environmental enrichment with which space will infuse the second oldest profession heralds a great leap forward in human culture. For the five millennia of its civilized history, architecture has worked within a fairly parochial range of conditions. Space bursts those archaic boundaries, substituting an unprecedented set of freedoms and restrictions. Old planetary constants become parameters. Gone will be the easy dialogue between indoors and outdoors that humans have always enjoyed. Interior "exteriors" must arise, since the true exterior is lethal. The harsh rules of space and its startling allowances will change altogether the relationship between people and their environment.

By being forced unequivocally to rethink human living, we can remake urbanism beyond Earth if we proceed carefully, starting afresh with the 5000-year history of civilization as practice. Anticipating the most emphatic environmental transformation our species will undergo fuels our incessant designer's hope of improving the human condition. The promise of a pristine, indeed unsuspecting, realm affording utterly new opportunity lures us to try, yet again, generating a new standard of firm, commodious, and delightful urbanism. The Moon provides our first and most priceless chance to prove to ourselves that we can be wiser than the historical evidence shows. Most critically, the clarity with which we can treat human living on an alien world may, at long last, teach us how to protect Earth as the uniquely precious planet it is.

Given time and trial, of course, even the fuzzy problems of lunar human living would approximately sort themselves out, as they have done on Earth. We would hope, however, that foresight could limit error through planning, even though space is an utterly novel arena; not only should we aim to design there an urbanism better than any found on Earth, we must aim to do it hundreds of times quicker than the luxurious five millennia we had here. Otherwise the extravagant cost in human suffering, material depletion, and environmental destruction will be unconscionably high.

LUNAR REALITY

Having defined the scope of architecture and urbanism, and established why access to space must affect their evolution, we can look more closely at their necessary expression on the Moon. Outright lunar prophecy is a specious goal, and certainly premature. So rather than portraying arbitrary details that might

characterize one possible future, we limn instead some factual boundaries that contain all the possibilities. The abundance of misleading images of lunar civilization means that certain basic principles remain unobvious, so we outline the most probable rules that will constrain what lunar architecture must be. Not all these facts will dominate lunar life until real urban growth supplants the first vehicular and outpost phases. Nor will they necessarily remain dominant for more than a few centuries, as they neglect unpredictable material progress.

Lunar urbanism will be *densely populated* at virtually all stages of its evolution. The modern "cottage" culture allowed by Earth's environmental largess cannot be afforded in a place where every cubic meter of vital volume must be hewn (or poured, or sealed, or assembled) and sustained. Nor can suburban "homestaking" really make extensive logistical sense on the Moon. Resources for construction and life support will generally not be dissipated on anything except the densest of cities; lunar society will be almost fully urban.

The overwhelming majority of lunar civilization will depend on indigenous manufacturing techniques. Offworld imports must inevitably be rate-limited. Thus common objects will be made locally, not because supplying them from space is impossible, but because it is impractical. A specialized computer might come from space, but the chair in which the programmer sits, the snack she munches, the scrap paper on which she jots notes, and the light by which she sees must all somehow be produced on the Moon.

This pervasively local origin of lunar culture, with its corollary need to fashion a human environment from the bottom up, will excite and occupy designers for generations and prevents us incidentally from divining a complete image of it now. Some conclusions are unavoidable, though. Simplicity will favor urban transportation machines like bicycles over powered vehicles—if a few kilograms of composite can provide mobility and exercise unobtrusively, elaborate centralized transit systems are likely to be justified only for interurban traffic.

We can expect most surface buildings to be made primarily of lunar concrete reinforced with local metal, serving both structural and shielding needs with minimal industry. We can expect alloys of titanium and aluminum to be used as commonly as are steel and plastic on Earth, and we can expect glass to be everywhere. Among the easiest materials to fabricate from lunar sources, glasses of varying purities will make up everything from tunneled cavern linings and architectural elements, to structural and optical fibers. We must expect that ubiquitous products will be made as quickly, cheaply, and simply as possible from available resources. This might well mean a built landscape dominated by poured, masonry, fired, and vitreous materials. Again, these are not all the Moon makes possible, but they will be the most expedient.

Lunar architecture must be an *interior* architecture. Heavily shielded havens are required during anomalously large solar proton events (ALSPEs, or flares), and cosmic rays (which Earth's atmosphere attenuates) irradiate the lunar surface semi-isotropically and continuously; the best long-term countermeasures are not yet known. It may well be that, when not actually working, people living in space will quite voluntarily limit their unshielded exposure. No modern myth seems more immortal, yet more hollow, than the persistent image of miraculous crystalline pressure domes scattered about planetary surfaces, affording their suburban populace with magnificent views of raw space (and incidentally baking them in strong sunlight).

However, the natural landscapes of the Moon's surface and the antisolar sky will be especially attractive to human sensibility. A lunar lifestyle may evolve that restricts recreational viewing to very special times, spurring ritual, behavioral, and special surface architectures for that purpose. Primarily subterranean, then, lunar cities would be heavily top-shielded by concrete superstructures, by regolith overburden, and perhaps even by areas of untouched wilderness overlying tunneled city caverns. The planetary surface, both natural and engineered, will be the single most important architectural boundary on the Moon.

That boundary must in general also contain *atmospheric pressure*. While the enclosures inside lunar cities can be structurally rather conventional, every square meter of the hermetic city wall surface itself must withstand over 100,000 newtons of force exerted by the air within it. In fact, a regolith overburden with sufficient weight to counteract this pressure would exceed by many times the thickness required for safe shielding alone. Pressurized, lunar cities will in effect be spaceships; no other single feature argues more strongly for an economical, underground urbanism there.

Lunar life need not be troglodytic, though. Many ages of architecture, three of which provide contrasting programmatic examples, have been conceptually or explicitly interior. The urbanistic Roman Empire was conceived and executed as a sequence of controlled volumes and views that regarded all the natural landscapes it conquered and absorbed, from the Middle East to the British Isles, as alien. Imposing the same planning schemes everywhere, Romans created their own universe around themselves, civilizing it with gods of their convenience and arranging in it the ordered landscape of their choosing. Virtually all outdoor spaces in Roman cities functioned as urban "rooms" within which the public rituals of Roman society could be played out. The Roman invention of concrete allowed enclosed volumes of a truly public scale never before seen, and the legacy both of those volumes and of the street facades that surfaced and announced them remains alive today.

In the western medieval millennium following the Roman Empire, northern cold and frequent warfare conspired to produce a genuinely interior environment. Often little more from the outside than a densely shielded pile, medieval architecture peered out of halls and chambers through tiny slits recessed in thick masonry walls. The intellectualism of Christianity encouraged introspection, and even ornament shrank largely off the stone architecture to cloak the people instead. To the east, the old Roman extravagance became Byzantine piety, still with enormous and lavishly ornamented interior spaces but now in the service of religious mystery rather than a secular civic public. Eventually, belief inspired the West to refine its masonry construction technology to recover volume, stretching the old Roman basilica upward and flooding it with light from above. Gothic religion came to sustain an interior architecture as potent, grand, and influential as anything Roman.

Most familiarly, twentieth century North America has evolved the inclusive interior mall to compensate the automotive consumption of its natural landscape. Reverting indoors, public attention is occupied and stimulated by the mall's manufactured landscape. The consistency of its style is place-independent, making Toronto and Los Angeles essentially the same. Driven by capitalism instead of religion or conquest, this enclosed, pedestrian-scaled, and transient strip architecture will also find expression in lunar interiors.

The civic pride, inspiration, and commercialism whose built expressions we have just reviewed briefly will be among the old and new motives guiding lunar civic building. Referring eclectically to the rich human past, a pluralistic twenty-first century lunar culture will embody its own aspirations in the public interiors it builds. All types of lunar interiors will share two distinctive differences from Earth's, however. First, they must accommodate a larger scale of human movement. Although details await experience, a natural gait in lunar gravity will stride longer and hops will rise higher. Human "factors" will have a new problem to solve. Interior supporting structures, governed by economy, will be much more slender than Earth allows. Lunar architecture will therefore be lighter and seem more expansive than Earth architecture, despite its pressurized closure, its exterior shielding, and its urban crowdedness.

Lunar life will be *nonsterile*. Human beings are elaborate ecological hosts, having evolved in the septic biosphere of Earth a web of commensal and truly symbiotic interactions with other organisms. Our understanding of these relationships is too shallow, and utter sterilization too impractical anyway, to plan seriously a sterile offworld ecology. Pathogen management will be a difficult but real problem. Lunar cities themselves will host life as well. Some bacteria that metabolize by corroding metal and can live in environments extreme in temperature, pressure, radiation, and toxics will exploit niches in space. Feral pet and research animals will eventually coinhabit lunar cities, and it is inconceivable that urbanism could grow off Earth without bringing along the venerable cockroach (pigeons should be avoidable). Expansion will be too fast and quarantine too porous to prevent eggs and spores from colonizing the Moon with us.

Finally, the Moon must be a place of unprecedented demarcation between *wilderness* and human use. The ancient fixture of a town wall to distill urbanism from the countryside will recur there, not so much to protect inhabitants from the space environment, but rather to protect the natural lunar environment from human destruction. Fragile though the biosphere of Earth may be in the face of "development," we are nonetheless deeply spoiled by its resilience. The encroachment of living things, relentless weather, and finally even the inexorable tectonics of Earth's geology condemn most signs of human action here to transience. Left alone, denuded forests and ravaged desert ecostructures can eventually recover despite appalling erosion and even toxic pollution.

The lunar wilderness, however, is truly fragile and effectively irrecoverable, despite its inanimate nature. Micrometeorite "gardening" takes millions of years to remake just centimeters of regolith. The forces that allow reclaiming strip mines and ruins on Earth simply do not exist on the Moon; the first trek through a pristine region of the Moon's unique "magnificent desolation" ruins its ineffable wilderness value practically forever. Surface exploration, strip mining, and construction will be facts of human activity on the Moon. So, sooner or later, will be human demands for utter preservation of untouched wild regions. The small planetary size of the Moon, which makes preventing its total use more urgent, also will aid that effort since its close horizon isolates areas visually. Wilderness appreciation cannot be participatory on the Moon the same way it is on Earth. The solace and renewal afforded by contemplating wilderness will induce radically new forms of urban design and specialized architectures to accommodate that human need on the Moon.

The few fundamental properties of lunar architecture and urbanism reviewed here grow directly out of facts as intrinsic to the Moon as weather is to the Earth. By accepting them as boundary conditions, our projections of the incipient, built human lunar environment can be more apt and more useful for planning our future. No rule, after all, prevents rigorous designs from being as exciting, as romantic, and as inspirational as specious ones. Many people, whether professional designers, authors, illustrators, engineers, explorers, leaders, or planners, are thrilled by thinking about living in space and on the Moon. Now is the time to inject realism into those thoughts. By starting from a few accurate principles—that lunar urbanism will be primarily *densely populated, interior,* and *nonsterile*; that it and the civilization it reciprocally defines will be pervasively *indigenous* in its materials and themes; and that lunar *wilderness* is irreplaceably precious—those who do plan can contribute meaningfully to realizing responsibly one of the grandest projects ever imagined in human history.

EARTH-BASED ANALOGS OF LUNAR AND PLANETARY FACILITIES

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Antarctica contains areas where the environment and terrain are more similar to regions on the Moon and Mars than any other place on Earth. These features offer opportunities for simulations to determine performance capabilities of people and machines in harsh, isolated locales. The Sasakawa International Center for Space Architecture (SICSA) plans to create a facility on Antarctica for research, planning, and demonstrations in support of planetary exploration. The Antarctic Planetary Testbed (APT) will be financed and utilized by public and private organizations throughout the world. Established on a continent owned by no country, it can serve as a model for cooperation between spacefaring nations. APT science and technology programs will expand knowledge about the nature and origin of our solar system, and will support preparations for human settlements beyond Earth that may occur within the first quarter of the next century. The initial APT facility, conceived to be operational by the year 1992, will be constructed during the summer months by a crew of approximately 12. Six to eight of these people will remain through the winter. As in space, structures and equipment systems will be modular to facilitate efficient transport to the site, assembly, and evolutionary expansion. State-of-the-art waste recovery/recycling systems are also emphasized due to their importance in space.

BACKGROUND

The Presidential Directive on National Space Policy released on February 11, 1988, has established, as one goal, "to expand human presence and activity beyond Earth orbit into the solar system." Earlier, the National Commission on Space appointed by President Reagan recommended that a permanent lunar outpost and human visitation of Mars be realized early in the next century.

Enormous program costs required to establish a lunar base or to undertake manned missions to Mars will probably be too expensive for even the wealthiest individual nations to justify. International cost-sharing will enhance economic feasibility and also help to ensure that mission purposes will benefit the world community. Antarctica, an international territory, is an ideal place to demonstrate that such cooperative programs can work.

Exploration of the solar system includes research to yield an improved understanding of planet Earth. The Antarctic Planetary Testbed (APT) facility will support scientific investigations of causes and effects of weather patterns and atmospheric changes that influence our human destiny; forces and evolutionary processes that shape the composition and distribution of planetary resources; and ways to accomplish social progress and prosperity while also protecting fragile ecosystems (Table 1). Such issues are of vital importance to all world populations. Accordingly, APT research should involve scientists from many nations. Living and working together under remote, rugged conditions, the culturally mixed crews can demonstrate that cooperation under difficult circumstances is both possible and essential.

The Sasakawa International Center for Space Architecture (SICSA) is contacting science and technology leaders from many countries to invite their participation. These experts represent government, university, and private organizations spanning a broad range of disciplines and resources.

TABLE 1. Key APT purposes.

Social and Life Sciences
- Psychological and social dynamics experiments involving mixed/international crews under severe, isolated conditions.
- Adaptation and performance assessments under harsh environmental conditions.
- Simulations of food production for self-sufficiency under isolated conditions.

Earth and Planetary Sciences
- Atmospheric, weather, and meteorological studies applicable to Earth/planets.
- Geological, geophysical, and physical chemistry research experiments.

Technology Demonstrations
- Partially closed-loop life support systems.
- Waste reclamation, treatment, and recycling systems.
- Advanced power generation and distribution systems.
- Excavation, mining, and material processing systems/procedures under harsh conditions.
- Construction/assembly systems and procedures under harsh conditions.
- Automation/robotic system versatility and reliability under harsh conditions.

Training Ground for Planetary Missions
- Crew observation for candidate selection and team assignments.
- Crew preparation for long-duration planetary missions under simulated conditions.

International Model
- Create and demonstrate participatory agreements that prepare the goundwork for future international initiatives.
- Encourage participation of international government and private sector organizations in advanced mission planning.
- Demonstrate economic and mission benefits to be gained through international investment and participation.
- Serve as tangible expression of commitment to future planetary initiatives.

SOCIAL RESEARCH EMPHASIS

An important APT function will be to serve as a psychological and social research laboratory. In addition, realistic space mission simulations will afford opportunities for crew training and selection.

Extended lunar surface missions and long manned voyages to Mars and other planets will pose great psychological and physiological demands on crews. Abilities of the crew to work well as individuals and as members of a team under these prolonged, difficult circumstances is an urgent concern.

U.S. and Soviet space missions to date demonstrate that social interactions are often complex and problematic. Successful team efforts require that individuals like, respect, and adjust to one another on a very personal basis. Learning to depend upon each other's judgement and technical knowledge is also essential. Such vital group "chemistry" is difficult to predict based exclusively upon psychological profiles of crew candidates.

Most available data pertaining to group dynamics under extended, harsh, isolated conditions is anecdotal and unreliable. While many Soviet space station crew experiences have been of relatively long duration, very little information revealing scientific details about crew interaction, adaptation, and performance has been released. Submarine isolation data is not directly applicable since neither the characteristics or size of the crew populations are comparable. Lessons derived from conventional Arctic and Antarctic experiences have similar limitations due to differences in population groups and the nature of their activities.

APT research and demonstration programs will select crew populations and activities to match real mission conditions and objectives as closely as possible. The participants will be international in composition to reveal insights about the ways cultural differences and preferences can be successfully accommodated.

SPECIAL EARTH SCIENCE BENEFITS

Use of the APT facility for research that is not exclusively space-related can help to cover costs for program implementation and operations. While several nations currently maintain research stations in Antarctica, the APT will be unique as an international base, affording living and work accommodations for a wide variety of cooperative ventures.

A representative APT use is to provide a laboratory for field measurements of seasonal changes in the Earth's upper atmosphere, the stratosphere in particular. Purposes will be to advance our understanding of physical, chemical, and meteorological processes that influence perturbations in ozone above Antarctica that were first observed by the British above their Halley Bay Station during the mid 1970s. Since that time, the October mean ozone level measured at Halley Bay has dropped between 40% and 50%. Potential enlargement of the 12-million-km^2 "hole" is viewed with alarm because atmospheric ozone is responsible for screening out more than 99% of the solar ultraviolet radiation that reaches the Earth's atmosphere. APT research can focus international attention on natural and man-made ozone influences and countermeasures.

APT research can also direct international resources and concerns to other issues of global importance. The Antarctic continent is a major forcing system driving the Earth's weather systems. Accordingly, APT research can investigate and monitor air and ocean transport of radioactive particulates and toxic chemicals, magnetospheric phenomena and their relationship to the solar wind magnetosphere system, and influences of ice and other surface features upon past, present, and future climatic conditions.

APT studies of the Antarctic plate can expand knowledge about the evolution of the Earth's crustal and upper mantle structures. This can lead to a better understanding of the way mineral resources are distributed throughout our planet, whether or not they are to be exploited.

TECHNOLOGY DEMONSTRATION APPLICATIONS

Severe environmental conditions on the Moon and Mars will pose hardships for people and machines. Failure to validate procedures and systems under realistic conditions is likely to be costly in terms of human life and/or failed missions. The APT facility will offer a valuable environment for realistic simulations and assessments (see Table 2).

TABLE 2. Antarctic-planetary analogs.

Environmental Characteristics
- Antarctica, which averages approximately 8000 ft above sea level, is the Earth's highest and driest continent with atmospheric pressure most like Mars.
- Antarctica receives relatively high levels of solar radiation similar to conditions on the Moon and Mars.
- Antarctica, having temperatures as low as -100°F and even colder at the South Pole, has similarities to the Moon and Mars.
- Coastal Antarctic winds range from a 15 mph average to 200 mph and more. Snow storms in these locations have similarities to dust storms on Mars.
- Antarctica and the Moon have locations with long days/nights that affect surface operations; as many as 3 months of darkness/extreme cold on Antarctica and 14 days on the Moon.

Geological Features
- Antarctica is a large, mostly virgin land mass, about the size of the U.S. and Mexico combined. It has a variety of landscape features from which to select sites for planetary mission simulations.
- Antarctica's rock bed terrain includes areas with sterile soils devoid of any life forms, similar to conditions on the Moon and Mars.
- Polar ice caps on Antarctica and Mars are rare in our solar system. Earth and Mars are the only planets possessing these features.

Program Priorities
- Antarctica, like the planets, belongs to no nation. Current and future treaties governing its use can provide a model for cooperative international space initiatives.
- The Antarctic ecosystems must be respected and protected. Similar conservation priorities apply in planning orbiting and planetary habitats.
- Antarctica's remoteness imposes space-like living, work, and resupply constraints. Facilities should be easy to construct, and self-sufficiency should be optimized.

The APT initiative will also provide incentives to advance valuable technologies for terrestrial uses. For example, critical APT performance and reliability requirements under extreme climatic and working conditions will offer demanding tests and testimonials for companies developing commercial products. The APT's remote location and extended-duration crew duty cycles will encourage innovations to achieve high-yield, efficient food and energy production. APT's space-applicable methods to treat and recycle wastes will demonstrate that human settlements can be nonpolluting and environmentally responsible.

A key technology development and demonstration objective will be to realize a high level of self-sufficiency through local growth and processing of food sources. Severe limitations upon environmentally conditioned volume and manpower will demand careful selection of nutritious, rapid-growth plants and animals that are easy and efficient to attend. Hydroponic agriculture and controlled pond fish and shrimp farming are candidate approaches. Organic wastes will be recycled for reuse to the extent possible.

Another priority will be to implement and evaluate autonomous power generation and storage systems. Candidate technologies include biomass systems that produce gas from organic wastes, fuel cells that produce electricity through a reverse osmosis process, wind turbines, and small nuclear generators.

APT operations will provide challenging applications for robotic and other automated systems. Experiments will include excavation to obtain and process mineral resource samples, *in situ* material processing, and construction techniques.

SITE SELECTION/DEVELOPMENT

An important APT planning and implementation priority is to minimize man-made disturbances to Antarctica's environment and ecosystems. This will be accomplished through careful site selection, avoidance of large transportation and construction equipment requiring extensive roadbeds, limitations on crew population size, and emphasis upon reprocessing, reuse, and control of waste materials. Similar considerations apply in planning orbiting and planetary habitats.

The site selection process will correlate key lunar and martian mission simulation objectives with environmental, geological, and programmatic features of various candidate locations. Final selection will be made following studies to determine prudent restrictions on placement, development, and operations to protect wilderness areas. These investigations will be conducted by an international body of scientists and planners committed to ensuring that terrestrial as well as planetary objectives are recognized.

The APT site development plan will adhere to comparable guidelines for hardware and practices appropriate to transportation, construction, and operational procedures on a planetary surface. For example, construction equipment and elements will be sized to be transported to and within the site using relatively small vehicles. Building components and interior systems will be modular, designed for simple and rapid assembly by small crews. This modular approach facilitates evolutionary expansion of site structures: additions, upgrades, reconfigurations of equipment, and changeouts of components for periodic and emergency servicing.

APT will demonstrate state-of-the-art technology to collect, treat, and recycle waste materials to the maximum extent possible. This will help to reduce storage and resupply requirements and prevent the creation of undesirable contaminants. These priorities are also of importance in planning long-term manned space missions.

CONSTRUCTION PRECEDENTS

The initial APT development will be constructed by a crew of approximately 12 people during the Antarctic summer. Six to eight members of this international group will winter-over to establish permanent base-camp operations. Later expansion stages may ultimately produce a small settlement of about 50 people. Tables 3 and 4 present key facility construction guidelines and elements.

There are presently more than 40 manned scientific stations in Antarctica. The largest is the U.S. McMurdo Station, a scientific base and transit center that can accommodate more than 200 people during the winter and more than 1000 during the summer. The Antarctic stations vary in size, facilities provided, and construction.

During the International Geophysical Year, 50 Antarctic stations were operating, including 47 year-round facilities. The cost of establishing and maintaining these stations ran as high as \$1 million/person for the Amundsen-Scott base in 1957. Construction approaches differed with national preferences: British and Norwegian expeditions favored conventional frame buildings, the French used prefabricated pressed steel huts, while American, Australian, and Soviet huts were typically built on rock sites using prefabricated aluminum or plywood panels clamped together to form flat-roofed box-like buildings.

Typical heating systems apply ducted hot air, sometimes using waste heat produced by diesel power generators. Trash management often involves sealing garbage in empty oil drums that are left on sea ice. Some larger stations have heated and insulated sewage pipes.

The largest interior base in Antarctica, the Byrd Station, provides power with a nuclear reactor. The base has 15 buildings (including a hospital) with winter accommodations for 40 people.

TABLE 3. Design and construction guidelines.

Safety
- Design and select materials to reduce fire hazards in the dry Antarctic climate.
- Securely anchor building to withstand high wind speeds.
- If located in a locale with snow, provide means to raise the structure or another device to prevent blockage of entries.
- Design for snow loads (if appropriate).
- Provide emergency health care equipment and supplies.
- Provide a safe haven with emergency rations separate from the main crew living facilities.
- Provide backup power and communication systems.
- Design for easy maintenance and repair of all life/safety-critical systems.
- Provide means for emergency crew evacuation by air and/or land.

Economy
- Emphasize modular and easy to assemble construction systems.
- Size modules and other construction/equipment elements for transport by most economical means.
- Provide economical, local energy source heating and power systems.
- Provide means to treat and recycle waste materials to the extent possible.

Mission Simulation Applicability
- Size and configure facility modules to conform with shuttle payload limitations.
- Provide means to reconfigure interior spaces and equipment for changing demonstration requirements.
- Simulate and provide facilities for simulating construction/assembly procedures to the extent possible.
- Duplicate space habitat environment and functions to the extent practical.

TABLE 4. Initial facility elements.

Building Systems and Equipment

- Modular enclosure(s) of design and size to afford simple and rapid construction.
- Modular interior equipment systems enabling easy change-outs and expansion.
- All elements designed for transport by relatively small air and surface vehicles.
- State-of-the-art systems to collect and recycle waste materials.

Living Accommodations

- Crew quarters for 12 people (summer) and 6-8 people (winter).
- Means to reconfigure living spaces for planetary simulation experiments.
- Galley and wardroom comparable to size and menu provisions of space station.
- Basic exercise, toilet, shower, and laundry equipment.
- Small health maintenance facility for routine and emergency medical care.

Research Accommodations

- Facilities for human, animal, and plant life science research.
- Utility interfaces to accept standardized space station-like experiment racks.
- Data buses and computing systems to control and monitor experiments.
- Laboratory space with work benches, information resources, and storage.
- Maintenance and parts room with basic tools and calibration equipment.

Grounds and Ancillary Structures

- Greenhouse/biosphere for plant growth.
- Staging area and equipment for Earth and planetary science experiments.
- Mining and material processing/sample return simulation area.
- Space construction and assembly simulation area.
- Vehicle repair and storage facilities.
- Helipad and fuel storage depot.

CANDIDATE POWER SYSTEMS

Important power system selection objectives are to minimize dependence upon imported fuel sources, environmental pollution, and maintenance. Systems with potential value for planetary applications will be tested on a continuing basis and used when feasible.

Biomass systems can process organic wastes through anaerobic biochemical fermentation to produce gas for combustion generators. Basic equipment includes a digestor tank and pipeline. An attractive feature of this process is its ability to beneficially use by-products of daily activity that normally present a disposal problem. Since a minimum 95°C temperature must be present for anaerobic processing to occur, the digestor must be well insulated and may require special heating.

Fuel cells convert water into constituent hydrogen and oxygen atoms through reverse osmosis for storage, then back again producing electricity as needed through a chemical reaction. Fuel cells, unlike dry cells, do not require recharging. They will continue to operate as long as hydrogen and oxygen are supplied, producing low-voltage, high-amperage direct current (DC). Practical voltages require stacks of these cells. Alternating current (AC) is obtained by adding a power converter (inverter). Normal conversion efficiencies range from 40-60%. Air pollution is 1000 to 10,000 times less than produced by traditional combustion systems.

Wind turbines represent a potential power option in Antarctica since winds are usually present. While oscillation between maximum and minimum wind speeds is typically high, the frequency between them is short. Electricity can be stored in batteries during calm periods. Wind turbines are easy to transport and require little maintenance. Rotor icing can occur when exposed to blizzards unless electrical resistance heaters are provided. The "fuel source" is locally abundant and free.

Nuclear power generation is clean and reliable but may be most applicable for use in facility operations that exceed the APT in size (e.g., the Byrd Station). Current systems generally require replacement after about two years of use.

PRIORITY TECHNOLOGIES

As in space, high transportation costs, restrictions on cargo payload volumes, and limitations on periods of site accessibility pose serious logistic constraints. These constraints impact the design of facilities and major equipment elements, resupply of consumables, and crew rotation cycles. Stringent conservation measures are necessary to conserve fuel for power and heating (a dominant logistic burden) and to protect the environment.

The APT can serve as an operational testbed for new or improved waste management recovery technologies that can be beneficial in space and remote terrestrial settings. Representative types of systems that are being given consideration include: (1) vapor compression distillation to recover potable water from urine and other sources; (2) thermoelectric integrated membrane evaporation to distill and recover water under a partial vacuum; (3) vapor phase catalytic ammonia removal to recover water from urine using high temperatures; (4) reverse osmosis to remove suspended macromolecules, ionized salts, and certain low-molecular-weight species; and (5) multifiltration, which flows waste water through a series of particulate filters, adsorption beds, and ion exchange resin beds to recover potable water.

The APT will apply advanced power technologies developed by different nations when and where possible. New NASA programs such as the Civilian Space Technology Initiative (CSTI) and Pathfinder program are supporting aggressive efforts to develop advanced power systems for the future. Improved systems created through these enterprises may offer clean and efficient options to replace the fossil-fuel-driven systems used throughout Antarctica today.

Current research involving photovoltaic (PV) systems is aimed at improving solar conversion efficiency above 20% and increasing capacities of electrochemical storage systems. Major studies involve refinements of indium phosphide, amorphous silicon, and gallium arsenide solar cells. (Gallium arsenide arrays have attained efficiencies of 23.5% with projected specific powers of 88 W/kg.) Special emphasis is also being directed to improving battery and fuel cell technologies. The "next generation" battery appears to be the Na/S battery that is being developed by the Air Force's Wright Patterson Aero Propulsion Laboratory (PAPL). Fuel cell improvements are aimed at developing single unit regenerative H_2O_2 cells.

Advanced solar dynamic (ASD) systems can be competitive with PV technologies in very small sizes, but are even more attractive for higher power applications (50 to 100 kWe) where increased efficiency is important. NASA's aggressive ASD technology development program for space station *Freedom* is aimed at providing a factor of 5 or 6 increase in the specific power that will be afforded by the initial PV system. (The 773-m^2 ASD system on space station *Freedom* will be about one-fourth of the array area size of the PV system.)

NASA's advanced nuclear system developments are based upon use of the SP-100 reactor. Special research and development emphases are advanced Stirling and thermoelectric energy conversion systems and space environmental effects safeguards. Radiation protection possibilities for future space applications include shields manufactured from lunar soil. An objective is to enable 80 W/kg space power systems in 500 to 800 kWe modular units.

TRANSPORTATION TO APT SITE

The APT's remoteness will create access problems that are similar in principal to circumstances encountered in planning future planetary bases. This isolation and the requirements it imposes upon the APT's development and operations are intentional, bringing levels of realism to the program that will maximize its value.

Important factors influencing transportation alternatives are (1) the specific APT site location, (2) minimum payload size requirements for initial base establishments and resupply, (3) seasonal weather and ice conditions restricting schedules, (4) emergency rescue accommodations, and (5) environmental safeguards governing acceptable modes for personnel and cargo delivery to the base. Minimization of damage to the natural setting is highlighted as a priority concern.

The maximum size for a cargo item is assumed to correspond with capacities afforded by the shuttle orbiter's 60-ft-long, 14-ft-diameter payload bay. This restriction is imposed to support compliance with existing standards applied to space systems.

APT transportation scheduling will be strongly influenced by severe weather conditions. Short summer "access windows" are analogous to space mission constraints. Large freight ships with ice-strengthened hulls can travel to Antarctica from about mid-December to early March. Other marine freighters, under escort by Coast Guard icebreakers have access to some Antarctic ports from late November to April. Air travel is less dependent upon seasonal weather conditions but is more expensive. Popular Antarctic aircraft are the Lockheed C-130 Hercules, Lockheed C-141 Starlifter, and Antonov An-22 Cock.

Access to the APT site from the nearest ship port and/or airfield constitutes the most challenging problems. Helicopters offer versatility and require minimum site accommodations but are hampered by extreme weather conditions. Accordingly, supply and crew rotation events are planned to be limited primarily to warm summer months.

APT ORGANIZATION OPTIONS

The APT program is conceived as an initiative that involves international organizations in its management and/or use. The specific management structure and operating rules remain to be determined based upon future sponsor interests in compliance with appropriate government laws and policies.

Two general sponsorship and organization approaches are being explored: an internationally financed initiative governed by a Board of Owners; and U.S. Government-sponsored and -owned operation that invites international use.

The multinational Board of Owners approach has a precedent in other international space organizations such as Intelsat and Inmarsat. These organizations provide at least one body that decides issues on a one-nation, one-vote basis (traditional international law) and a second body that apportions power and profits based on share ownership (a flexible, more capitalistic and less statist approach). Intelsat, created in 1965 largely as a U.S. initiative, serves as a successful and continuing model for organizations that desire to develop and share resources and other benefits in a multinational extraterritorial format.

The U.S. Government-sponsored approach envisions a U.S. facility that would invite international use in exchange for financial and in-kind service support. Key organizing sponsors might be NASA and the National Science Foundation (NSF). This approach, provided that such sponsors are interested, will probably be the easier of the two to implement and manage, but is limited as a model for future international partnerships in space.

Program financing for both organizational approaches can be provided through a combination of government grants and user fees. Properly located, outfitted, and operated, APT will represent a valuable real estate investment as well as an important asset to advance world progress and benefits in space (Table 5).

TABLE 5. Benefits for APT participants.

Government Sponsors
- Reflects a national/agency commitment to advance progress towards solar system exploration.
- Facilitates and demonstrates development of advanced systems for space and terrestrial uses.
- Encourages and directs private sector participation and investment in space initiatives.
- Brings nonaerospace organizations and technologies to light that represent new capabilities.
- Produces new knowledge and experience to reduce risks and costs for future space initiatives.
- Fosters and symbolizes a commitment to international cooperation.
- Offers potential model for future international space initiatives.

Commercial Organizations
- Provides opportunities to demonstrate and publicize current technologies and capabilities.
- Provides means for organization to better understand and prepare for future government and commercial space business opportunities.
- Creates a service market for APT construction and operations support.

Research Organizations
- Affords extended access to Antarctica for space and other research.
- Provides opportunities and means to conduct unique research.
- Provides opportunities for networking and participation with other government and private research organizations and sponsors to advance science objectives.
- Presents occasions to share experiences and knowledge with people from other nations/cultures.

INFLATABLE HABITATION FOR THE LUNAR BASE

N93-17442

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Inflatable structures have a number of advantages over rigid modules in providing habitation at a lunar base. Some of these advantages are packaging efficiency, convenience of expansion, flexibility, and psychological benefit to the inhabitants. The relatively small, rigid cylinders fitted to the payload compartment of a launch vehicle are not as efficient volumetrically as a collapsible structure that fits into the same space when packaged, but when deployed is much larger. Pressurized volume is a valuable resource. By providing that resource efficiently, in large units, labor intensive external expansion (such as adding additional modules to the existing base) can be minimized. The expansive interior in an inflatable would facilitate rearrangement of the interior to suit the evolving needs of the base. This large, continuous volume would also relieve claustrophobia, enhancing habitability and improving morale. The purpose of this paper is to explore some of the aspects of inflatable habitat design, including structural, architectural, and environmental considerations. As a specific case, the conceptual design of an inflatable lunar habitat, developed for the Lunar Base Systems Study at the Johnson Space Center, will be described.

INTRODUCTION

As NASA plans future missions involving long-term human presence on the Moon, we must strive to provide a safe, productive working environment, while at the same time minimizing the cost of transporting the lunar base elements. Inflatable structures, also known as expandable or pneumatic structures, have great potential for achieving both of these goals.

An inflatable is a flexible pressure vessel that may be folded compactly for transport and then deployed to its full size on delivery. It offers the twin advantages of a large operational volume and a small transportation volume.

HISTORY

The use of inflatable structures in space goes back to the beginning of the space program itself. In the early sixties, the Echo satellite program sent enormous mylar balloons into orbit for communications experiments. Packaged, the Echos fit into a 1-m-diameter sphere, but when deployed they were 30 m in diameter and were easily visible in the night sky.

During the Apollo program, work was done on a number of inflatable manned space systems by the Goodyear Aerospace Corporation through NASA's Langley Research Center. The studies included materials testing and selection and the construction of inflatable prototypes. The configurations studied included a collapsible lunar shelter to be carried on the Apollo LEM to extend mission durations; a space station module, of which a full-scale prototype was built and pressurized to 5 psig; and a collapsible airlock (*Tynan et al.,* 1971). Though all these projects showed promise, insufficient funding was available to pursue the concepts beyond the prototype stage.

Meanwhile, fabric structures were finding new applications on Earth as well. In 1962, a West German architect, Frei Otto, published *Tensile Structures,* a landmark text describing the attributes and design considerations of pneumatic and air-supported structures (*Otto,* 1962). Interest in these structures grew due to their light weight and low cost, and the availability of increasingly stronger, more durable materials. Very large spaces could now be spanned at a fraction of the cost of rigid structure. The World Exposition of 1970, in Osaka, Japan, marked a watershed in the history of tensile structures. Many of the pavilions were pneumatic or air supported. The U.S. pavilion was covered with an enormous fabric roof made from a material derived from the Apollo spacesuits. This pavilion was the model for many domed stadiums and arenas erected throughout the country during the seventies and eighties.

Inflatable habitation for use in space merges the civil technology of large tensile structures with the aerospace technology of pressurized vessels. There is a sound experience base from which to launch an aggressive inflatable structures program.

ADVANTAGES

Transportation

Space habitation design is severely constrained by the dimensions of the launch vehicle payload bay. The high cost of space transportation drives the subsystem designer to utilize every cubic centimeter available. The result of this process is often a highly efficient, extraordinarily expensive, unique piece of hardware. If this volume constraint could be relaxed, allowing greater use of off-the-shelf technology, the result might very well be cheaper manned space systems.

Cost savings could be realized in launch operations as well by allowing a more flexible manifest with fewer vehicle-dependent payloads. The largest single element of a space station or lunar base is the habitation. If this element could be separated into its basic components, the options for launching it would expand considerably. It might be launched on a large vehicle, if one were available, or on a series of smaller vehicles. This flexibility in launch operations can be attained by launching the pressure vessel separately from the habitat components, and integrating them in

place. This approach requires that the pressure vessel be collapsible, so that it may be packaged efficiently (a rigid vessel occupies the same space whether it is empty or full). A fabric structure is the simplest way to make a collapsible pressure vessel.

Some advantage may be claimed for the rigid module because it carries its air with it. The air required to pressurize the inflatable habitat must be supplied separately, requiring space on a launch vehicle. However, transport of air will be part of the routine logistics system for a lunar base. It will be supplied to the base continuously to make up for losses (chiefly through the airlocks, which lose 10% of their air by volume each time they are used). There must also be an emergency repressurization supply on hand at all times. An efficient system for the transport of volatiles to the lunar base will be required, perhaps employing cryogenic technology to transport them as liquids. (The development of such a system is particularly likely if the transportation system uses cryogenic fuels.) The advantage to the program of getting the first load of air "free" inside a habitation module will therefore be minimal.

Operations and Growth

The large space available in an inflatable would facilitate rearrangement of the interior to suit the needs of the moment and would allow space for working with bulky equipment. For example, a structure to be erected outside could be put together and tested in a shirtsleeve environment before actual deployment. In this way, latent defects in the equipment could be identified before costly EVA time was wasted.

Using the same transportation, an inflatable habitation system can provide habitable volume in larger units than can a rigid system, thereby reducing the number of EVA-intensive surface construction operations as the lunar base grows. The high-volume inflatable can be "expanded" by adding more equipment and furnishings inside, as they are needed.

Habitability

The greatest advantage of large inflatable habitats may be the most difficult to quantify: habitability. Habitability is the sum total of those qualities that make an environment a pleasant place to live and a productive place to work. Studies have shown that personal space, for work and for leisure, is an important factor in the pyschological wellbeing of isolated groups (*Stuster,* 1986). The inflatable would not only provide a large volume, it would provide perceptible volume. A base built up out of rigid modules might be expanded indefinitely by adding more modules, but no single element could ever be larger than the payload bay of the launch vehicle that lifted it from earth. At no point in the lunar complex could a person perceive (or utilize) a volume larger than that of a single module. The space inside the inflatable, on the other hand, may be divided up in any way desired or left open to create large chambers.

DESIGN ISSUES FOR THE INFLATABLE LUNAR HABITAT

The design of an inflatable habitat for the Moon is influenced by a number of factors. It is influenced architecturally by the activities carried on within it, it is influenced structurally by the forces acting on it, and it is influenced by the environment in which it operates.

Architectural Design Issues

As used here, the term architectural refers to those factors that influence the volume and form of the habitat. The physical and psychological needs of humans in a confined environment must be balanced against the physical limitations of a pneumatic structure.

Volume. Habitable volume means a space to work that is pressurized, climate controlled, and protected from radiation. The amount of habitable volume needed depends on many factors: the number of crew members, what they will be doing, and how much discomfort and inconvenience they can be expected to endure. A tour at the lunar base would be the dream of a lifetime for scientists and researchers, no matter how poor the living conditions might be expected to be; but once they have been at the base for a few days, minor inconveniences may become major irritations, to the point of affecting performance. Providing flexibility in the design, so that the crew can implement their own solutions to any problems of space or interior arrangement that arise, is a primary goal in engineering the lunar habitat.

The concept of open volume is a key element in this philosophy of flexible design. There should be enough space to rearrange the interior and to expand the capabilities of the base without major exterior construction. As discussed earlier, inflatable structures are a way to provide large amounts of volume without a great impact on the transportation system.

Form. Inflatables can be made in virtually any shape, but for this study it was decided to limit consideration to those shapes not requiring reinforcing structural elements such as hoops or cables. The objective was to establish a simple baseline for comparison with more complex designs. Given this restriction, the three simplest shapes for an inflatable are the sphere, the cylinder, and the torus (Fig. 1).

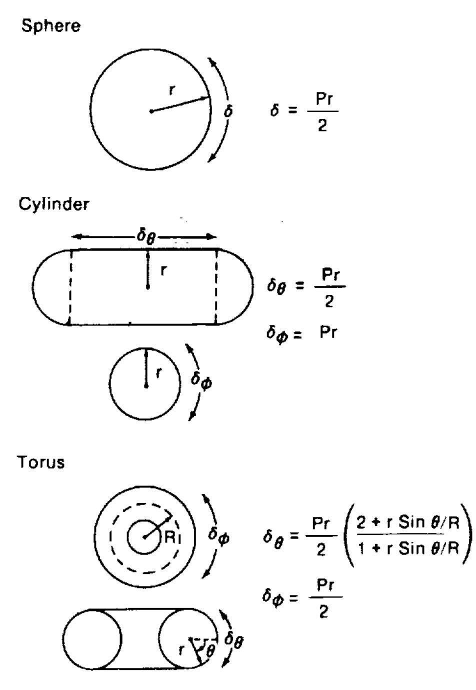

Fig. 1. Pneumatic geometries and associated stresses.

The spherical shape is the most volumetrically efficient, with the least surface area and mass for a given volume (Fig. 2). Stress is uniform throughout the membrane. The chief drawback is the architectural inefficiency of the doubly curved walls.

The cylinder is somewhat more convenient architecturally than the sphere, because the walls are curved in only one direction. With a vertical orientation, one could have vertical walls rather than the arched walls in the sphere. However, for a cylinder of similar dimensions to the sphere, mass and membrane stress are higher. (The circumferential stress in the barrel of a cylinder is twice that for a sphere of the same diameter.) To reduce this, one might make the cylinder more slender and lay it on its side, but this would largely eliminate its architectural advantage over both the spherical inflatable and the rigid space station-type module.

The torus is basically a cylinder wrapped into itself, thus saving the mass penalty of endcaps. It provides a "race-track" configuration that may have safety advantages if the inflatable is compartmentalized, but it suffers from the same compound wall curvature as the sphere. If a large minor diameter is chosen (to gain the open volume that is a presumed advantage of inflatable structures), the result is similar to a sphere with a "pucker" in the middle. From the inside, there will be a column in the middle of the structure that serves no purpose. This effect could be minimized by reducing the minor diameter and making the torus more ring-like, at the expense of open volume.

Structural Design Issues

Structural design issues arise from the nature of the structure itself, relatively independent of the environment. A pneumatic structure required to be pressurized to 101 kPa imposes certain constraints on its designers, regardless of its secondary functions or its environment. Chief among these considerations are the membrane stress, the leakage rate, and the puncture resistance.

Membrane stress. A pneumatic structure is stressed primarily by its internal pressure, overshadowing external forces such as gravity. The internal pressure creates stress in the wall of the inflatable called the membrane stress, measured in N/m. It depends on both the internal pressure and the local radius of curvature. In a sphere, the membrane stress is given by

$$\text{stress} = (\text{pressure} \times \text{radius})/2$$

There are similar equations for the stress in a cylinder or torus. The wall material must be strong enough to resist this stress, yet flexible enough to fold and package efficiently.

In general, the larger the inflatable, the larger the radius of curvature, and the higher the membrane stress. To reduce the membrane stress in a very large inflatable, structural reinforcements such as cables or hoops may be added. These take up some of the stress and reduce the local radius of curvature. As noted earlier, these elements complicate the design.

Leakage. Aside from resisting the membrane stress due to the internal pressure, the structure must also be sealed against leakage. The wall material must be impermeable to air, and care must be taken in the design of hardware interfaces (such as hatches or windows) to limit potential leak paths.

Puncture resistance. An inflatable structure is no more inherently vulnerable to puncture than a rigid pressure vessel. The difference is that an inflatable, if unsupported, will collapse of its own weight when all the air is removed. The internal framework needed to support the floors, walls, and equipment inside the habitat will also support the relatively lightweight pressure envelope. (In the Johnson Space Center concept, the structure must also support the regolith shielding in the event of a loss of pressure; in fact, this potential load condition drives the design, dwarfing the normal load imposed by the interior habitation elements.)

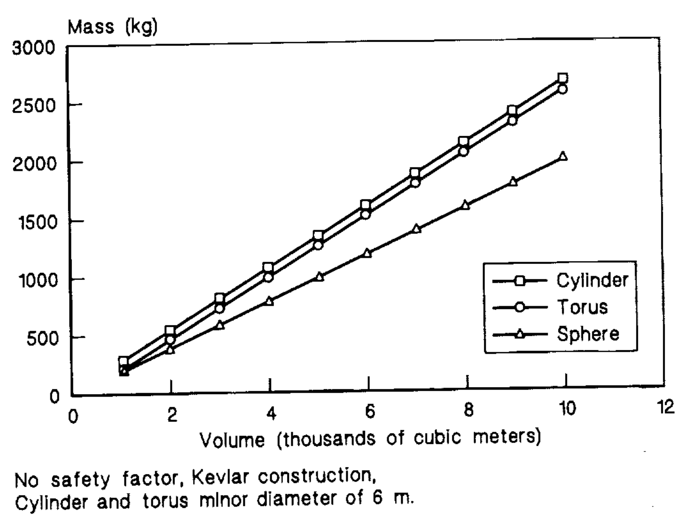

Fig. 2. Mass vs. volume for sphere, cylinder, and torus.

Environmental Design Issues

When designing a lunar habitat, the effects of the lunar environment over a long period of time must be considered. Radiation, temperature extremes, vacuum, and meteoroids all act to erode the integrity of the structure over time.

Radiation. The surface of the Moon, not protected by a thick atmosphere and strong magnetic field as is Earth, is exposed to higher levels of radiation from space. The radiation hazard comes from two sources of different character: galactic cosmic radiation ("background" radiation that permeates space at roughly constant levels) and solar energetic particle events that produce extremely high levels of radiation for short periods of time.

Many of the materials considered for use in inflatable space structures show sensitivity to radiation exposure. If a material cannot be found that has sufficient resistance to long-duration radiation exposure, then the radiation protection provided at the base should include the entire inflatable habitat. To deal with the radiation problem, it has been suggested that the habitat be covered with lunar soil or buried. This concept is as valid for inflatable structures as it is for rigid ones. The 101.4-kPa internal pressure is capable of supporting 40 m of soil on the Moon, assuming a soil density of 1.6 g per cubic centimeter. Obviously, though, some form of reinforcement should be provided in case pressure is lost and the inflatable is no longer able to support the soil load. Some precaution should also be taken against abrasion of the habitat material by the lunar soil, as vibrations from activities inside are transmitted through the envelope. A protective layer of abrasion-resistant fabric, or coating, could be employed.

Temperature and vacuum. The habitat material must be insensitive to temperature swings from 100 to 400 K (darkness to full sunlight) in vacuum. While the material will stay at room temperature and pressure during normal base operation, it may be exposed to more extreme conditions during transport and deployment, or in the event of a major malfunction in which the base is completely shut down for some period of time. In addition, the low thermal conductivity of polymers in general means that the wall material may be subjected to large thermal gradients, with room temperature (293 K) inside and 100 or 400 K outside.

Meteoroids. If the base is properly shielded from radiation, it is more than adequately shielded against micrometeoroids. Only a few centimeters of soil are required for this purpose. (Impacts large enough to penetrate several meters of soil are extremely rare.) In any case, the problem is the same for both inflatable and rigid structures.

THE INFLATABLE LUNAR HABITAT

The inflatable habitat, as currently envisioned, consists of a spherical pneumatic envelope with an interior structural cage to support the floors, walls, and equipment, and to hold up the envelope if pressure is lost (Fig. 3).

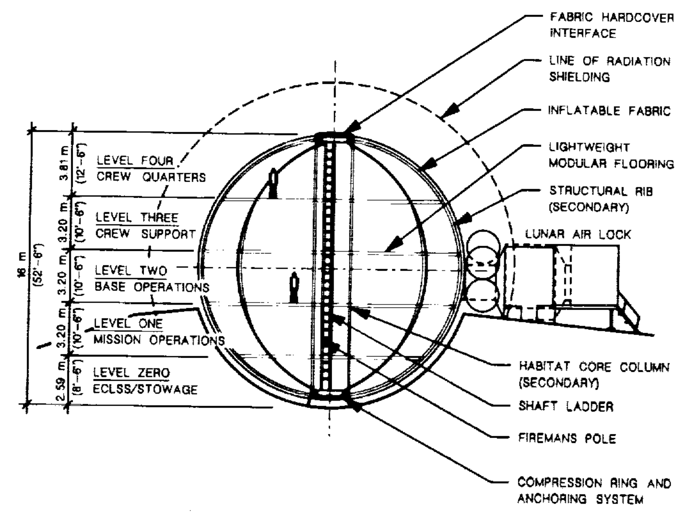

Fig. 3. Cross section of the inflatable lunar habitat.

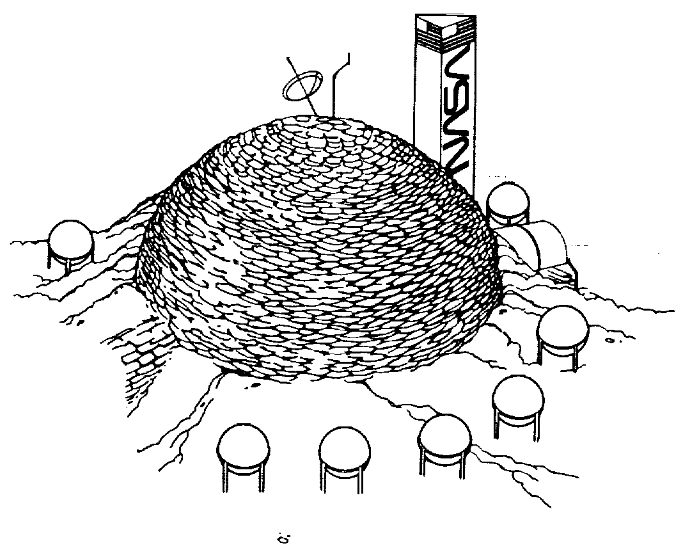

Fig. 4. The inflatable habitat "sandbagged" for radiation protection.

The sphere, when inflated, will be 16 m in diameter, containing 2145 cu m of open volume. The internal pressure is assumed to be 101.4 kPa (standard sea-level atmospheric pressure on Earth). This pressure was chosen, not because it was deemed best for the lunar base (there may be advantages to operating at a lower pressure), but because it represented a maximum. The spherical shape was chosen for its simplicity and efficiency and will be compared with other shapes in trade studies to come. The contained volume was arrived at by taking the total volume of the space station, dividing by the crew of 8, then multiplying this volume/crew member by the 12 crew members anticipated for the lunar base. There are four floors in the current design, with a total of 594 sq m of floor space. (The bottom floor will be curved like a bowl and is expected to be used as a maintenance bay or storage space.) An open shaft 2 m in diameter runs from the top to the bottom for transfer of personnel and equipment from floor to floor.

The inflatable envelope will consist of a high-strength multi-ply fabric, with an impermeable inner layer and a thermal coating on the outside. The analysis done so far has concentrated on sizing the structural layer alone, assuming it to be the primary contributor to the mass per unit area of the wall material. The material selected for use in making mass projections for the envelope is Kevlar-29, a high-strength aramid fiber made by the DuPont Chemical Company. DuPont literature indicates that a broadweave fabric of Kevlar-29, with a thickness of 0.114 mm, has a breaking strength of 525 N/cm (*Dupont Co.,* 1976). Based on this material, the envelope is anticipated to mass 2200 kg, with a structural safety factor of 5. (Keeping all other parameters the same, the mass is directly proportional to the safety factor.) The thickness of the structural layer is estimated at 5 mm, giving a material volume of 4 cu m. Assuming a 10:1 ratio of packaged volume to material volume (meaning that for every cubic centimeter of material there are 9 cu cm of empty space in the package), the packaged volume of the inflatable will be 40 cu m.

In the JSC concept, the habitat is covered with 3 m of lunar regolith in the form of "sandbags" (Fig. 4). This is sufficient to attenuate even the largest recorded flare to safe levels. The regolith imposes a maximum load of 7.8 kPa on the surface of the habitat, easily supported by the internal pressure of 101 kPa. However, placing the regolith is very labor intensive, taking up much of the crew's time during the early missions. An alternative might be to provide a few inches of shielding to protect the habitat from galactic cosmic radiation, with a separate, smaller, "storm shelter" to protect the crew in the event of a serious solar flare. In this case, it was felt that the operational simplicity of a completely shielded base, where no special operations would be required in the event of a flare, would justify the difficult construction task. Also, the long-term effects of exposure to cosmic rays are not well understood, and it is better to err on the side of caution in safety issues.

The interior structure of the habitat will consist of curved beams running along the inside of the envelope, like lines of latitude and longitude on a globe. A combination of radial and concentric beams will support the flooring. This interior

framework will support all the equipment and furnishings of the base. It will also support the regolith shielding in the event of a loss of pressure, and this load actually sizes the structure. (The interior equipment and furnishings are expected to weigh about 60 kN; the regolith weighs 1500 kN.) A rough estimate of the mass of this framework puts it at 16,300 kg, including primary structure (9000 kg), flooring (6000 kg), and walls (1300 kg).

There is a spectrum of assembly options for the internal structure, ranging from hand assembly at one end to fully automatic deployment at the other. A detailed design effort must be initiated to determine where in this spectrum the optimized configuration falls.

CONCLUSION

Inflatable habitation holds great promise for human presence on the Moon and in space. By making it possible to use smaller launch vehicles to lift lunar base infrastructure (the largest single element of which is currently projected to be the habitation), inflatables will allow greater flexibility in launch operations, while improving habitability.

The feasibility of large inflatable structures for space habitation has been demonstrated, but much work must be done to bring the concept to fruition. Materials have been developed that are strong enough for these applications (Kevlar, for example), but numerous other factors, such as temperature sensitivity and radiation resistance, must be examined. Fabrication techniques must also be refined. This can only be done through extensive laboratory testing and use of prototypes on Earth and in space.

In conclusion, a direct quote from the Langley engineers' 1971 paper, "Expandable Structures Technology for Manned Space Applications" (*Tynan et al.,* 1971), is appropriate:

> Tests on full-scale models of representative concepts show that they are sound structurally, have satisfactory deployment characteristics, and in most instances possess remarkable packaging ratios . . .
>
> With proper attention to the selection of materials, effects of the vacuum environment, electromagnetic radiation, and temperature extremes can be minimized . . .
>
> The most difficult problem faced by the expandable structures proponent, however, is getting the "breakthrough" acceptance of such a structure for a manned application to an operational flight program. Understandable prejudices against expandables for manned occupancy can only be overcome by actual use of the concept, at least on an experimental basis, in the space environment.

REFERENCES

DuPont Co. (1976) Characteristics and Uses of Kevlar 29 Aramid. *Preliminary Information Memo No. 375.* DuPont Company, Wilmington, Delaware. 8 pp.

Otto F. (1962) *Tensile Structures,* Vol. I. Massachusetts Institute of Technology, Cambridge. 320 pp.

Stuster J. W. (1986) *Space Station Habitability Recommendations Based on a Systematic Comparative Analysis of Analogous Conditions.* NASA CR-3943. NASA, Washington, DC. 209 pp.

Tynan C. I. Jr., Williams J. G., and Osborne R. S. (1971) *Expandable Structures Technology for Manned Space Applications.* AIAA Paper 71-399. American Institute of Aeronautics and Astronautics, New York. 15 pp.

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PRELIMINARY DESIGN STUDY OF LUNAR HOUSING CONFIGURATIONS

N93-17443

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A preliminary design study assesses various configurations for habitation of the lunar surface. The study assumes an initial 4-man habitation module expandable to a 48-man concept. Through the numerous coupling combinations of identical modules, five basic configuration types are identified. A design model presents each configuration in light of certain issues. The issues include circulation, internal and external spatial characteristics, functional organizations, and future growth potential. The study discusses the attributes, potentials, and unique requirements of each configuration.

INTRODUCTION

Numerous studies on lunar bases have concluded with a configuration depicting the necessary scientific parameters for operation and human habitation.

Mathematical laws actually permute an overwhelming number of configuration combinations. The near-infinite choices coupled with human habitation requirements present a dilemma that can best be resolved through architecture.

Architecture enables the selection of potential options that satisfy human needs: a sense of place, a sense of privacy, a fit into the environment, and a sense of well-being.

BACKGROUND

The preliminary design study concentrates on configurations and assumes the scientific parameters developed: module size determined by transport system, materials selections, radiation shielding, temperature control systems, and a closed ecological life-support system. For the purpose of demonstration, the configurations are composed of physically identical and interconnected modules. They establish a complex that can house a population expandable from 4 to 48 people.

The study allows the modules to be configured vertically as well as horizontally. Vertical arrangements present spatial qualities such as privacy. They introduce the potential for connections to experimental below-grade chambers and additional circulation loops.

THE MODULE

Each module shell is identical in composition. The basic shape is cylindrical, with a length twice the width. The interior orientates in any direction. In the horizontal position the plan form is rectilinear, and in the vertical it is circular.

The module is constructed with three openings, one on each end and the third on the side. The side connector demonstrates a greater number of combinations through reflection, translation, and rotation (Fig. 1). Such possibilities become extremely important in determining performance, placement, and siting of a lunar base. The module's sides can be orientated in any direction in the horizontal plane. The introduction of the vertical plane increases the number of possibilities; for example, when eight modules are related horizontally and vertically, the variations in combinations are innumerable. Basic configuration types need to become apparent. Only after identifying the attributes of the basic configuration types can the variations of other techniques be architecturally appreciated (Figs. 2 and 3). A balance between mathematical possibilities, scientific demands, and architectural needs can be attained.

BASIC CONFIGURATIONS

The study's design model categorizes the configurations into five basic groups: linear, courtyard, radial, branching, and cluster. Each configuration responds differently to the circulation and the internal/external spatial characteristics for a lunar base.

A configuration can provide a unique quality of space: open space, inner circulation loops, and privacy. The organization and selection of such qualities is based on the functional and operational procedures of the lunar base. A lunar base may require two or three shifts working and sleeping different hours. A

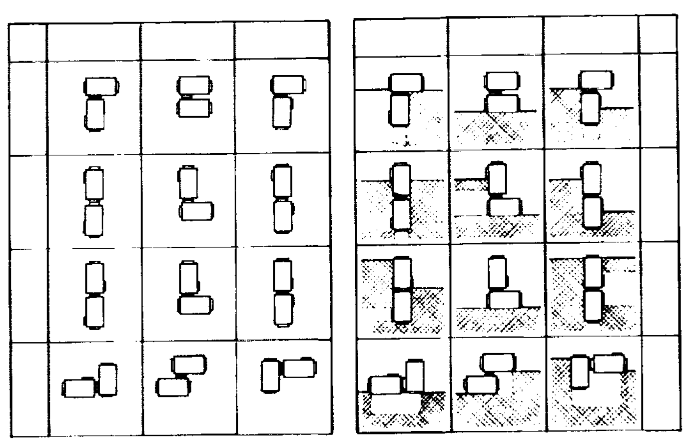

Fig. 1. Some possible combinations of two modules, each with three coupling options. Left is horizontal plane (plan view); right is vertical plane (section view through regolith). Note that varying the relation of configuration to lunar grade increases number of arrangements, especially egress routes.

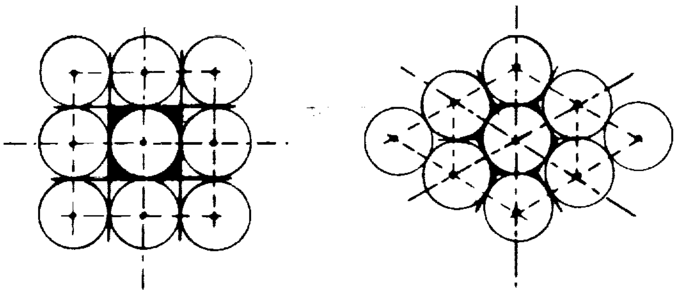

Fig. 2. Two close-packing options for circular modules.

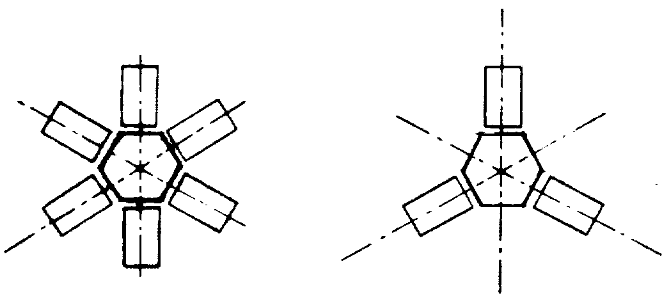

Fig. 3. Plan view of multiple coupling techniques.

configuration that provides three private crew areas, separately located, would permit a congenial working atmosphere, as opposed to the configuration with only one major crew area off a main corridor.

Each configuration provides reliability and safety through redundancy. Dual egress is necessary from any portion of the configuration. Airlocks must be located for easy access by the crew with emergency airlock systems as needed. In some configurations, a secondary safety circulation configuration loop is necessary.

Each of the five configuration types with their distinctive characteristics is presented in Table 1 and discussed in detail below.

Linear

The linear configuration is the simplest of all configurations. It is the repetition of modules with one primary circulation path. It can be stepped, undulating, or spiraling (Fig. 4). The internal distances from one end to the other are maximized. The spatial characteristics are primarily public-type spaces that are noisy and conducive to space-sharing with the circulation path (Fig. 5). Externally the configuration covers minimal area, yet requires maximum enclosure. Expansion possibilities are limitless with the configuration being altered to another form (Fig. 6). One of the main problems with the linear system is safety; a secondary circulation system that requires complicated looping techniques is necessary.

Courtyard

When a linear configuration closes on itself, its basic characteristics change: the area coverage becomes greater and the enclosure minimizes. The courtyard is a unique identifiable space. There still exists one primary circulation path; however, it now forms a closed loop, an internal dual egress system distinguishable from the other basic configurations (Fig. 7). Varying functional organizations are possible including a two-directional corner or nodal point condition (Fig. 8). An additional attribute is the courtyard area itself. It allows complete access to all modules for repair and maintenance and in the future can be altered to usable space (Fig. 9).

Radial

The radial configuration is centralized space with linear extensions in more than two directions (Fig. 3). The central area provides a major functional space with secondary areas radiating. These secondary spaces can be private, quiet areas or main circulation routes to additional functional zones (Fig. 10). The major concern with this configuration is that access to each arm is through a central zone. In terms of safety, this presents a concern with egress technique. A secondary emergency circulation system may be necessary to loop the radiating arms together. One advantage of the radial configuration is the provision of an easily accessible central functional zone. When the radial configuration is combined with another configuration type (i.e., courtyard or branching), it allows for distinct zoning possibilities.

Branching

A linear growth system that expands with secondary circulation paths from a main circulation path characterizes a branching configuration. These branching areas provide transition areas: public from private, noisy from quiet, and primary from secondary (Fig. 11). Functional zoning is diverse, providing multiple options. Also there are numerous expansion options, including in-fill between the secondary systems (Fig. 12).

Cluster

Cluster configurations have no dominant circulation patterns. They have the most open sense of circulation allowing for flexible, close proximity of functions and close packing of modules (Fig. 13). Generally a large central area is created, and a closed circulation loop is obtained. Private and/or quiet areas can be separated from public and/or noisy activity areas. With modules abutting one another, any necessary repairs or maintenance can cause a potential problem.

CONFIGURATIONS IN THE VERTICAL PLANE

So far the configurations discussed are with the circulation movement in the horizontal plane. With the introduction of a module in the vertical plane, different criteria are established: the relation of the configuration to grade and to itself. The circulation movement may be one configuration in the horizontal and another in the vertical. For example, a branching configuration may branch horizontally, yet with the introduction of vertical modules and a second plane of horizontal modules, a courtyard configuration could also appear.

TABLE 1. Characteristics of five basic configurations.

	Linear	Courtyard	Branching	Radial	Cluster
Basic Characteristics					
Length/width ratio	maximum	maximum	average	average	minimum
Area coverage	minimum	maximum	maximum	minimum	average
Enclosure	maximum	minimum	maximum	maximum	minimum
Circulation					
Type of circulation route	primary	primary	prim.+second.	multi	multi
Circulation intersections	none	none	multi	one primary	multi
Travel distances	maximum	maximum	average	minimum	minimum
Closed vs. open loop	open	closed	open	open	open
Dual egress	no	yes	no	no	no
Safety egress possible?	complex	exists	possible	possible	possible
Spatial Characteristics					
Primary space	yes	yes	yes	yes	yes
Secondary spaces	no	no	yes	yes	yes
Quality of space	noisy	noisy	noisy/quiet	noisy/quiet	noisy/quiet
Public vs. private	public	public	public/private	public/private	public/private
Space shared with circulation	yes	yes	not necessary	not necessary	not necessary
Type of space (interior)					
open or enclosed	enclosed	mainly enclosed	open/enclosed	open/enclosed	open/enclosed
vary size/shapes/locations	minimal	minimal	max. at nodes	central max.	maximum
vary circulation path type	minimal	minimal	maximum	average	maximum
Functional Organization					
By definition of configuration	decentral	decentral	central	multicentral	central
Adjacency or interconnect	yes	yes	yes	yes	yes
One-sided organization	yes	yes	yes	yes	yes
Multisided organization	no	yes	yes	yes	yes
Nodal organization	no	minimal	yes	yes	yes
External Characteristics					
Maintenance/repair	accessible	accessible	difficulty	difficult at central node	inaccessible
Emergency egress safety	complex	dual egress	secondary system necessary	secondary system necessary	dual egress
Single module disfunction	inoperable	operable	part operable	part operable	operable
Direct exposure sol-radiation	maximum	minimum	maximum	maximum	minimum
Shade formed by configuration	maximum	maximum	maximum	maximum	minimum
Sol-radiation/shade ratio	average	minimum	average	average	maximum
Regolith coverage required	maximum	minimum	average	average	minimum
Expansion/Future Growth					
Limitless growth horizontal	unidirectional	no	multidirectional	unidirectional	no
Future in-fill	no	yes	partial	partial	partial
Vertical expansion	yes	yes	yes	yes	yes

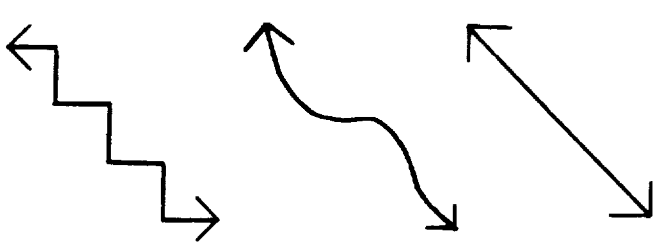

Fig. 4. Linear circulation paths.

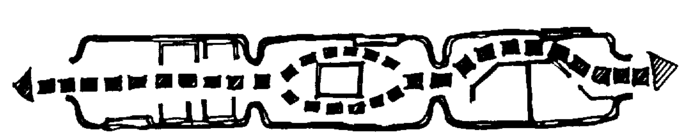

Fig. 5. Schematic plan of linear configuration's spatial characteristics.

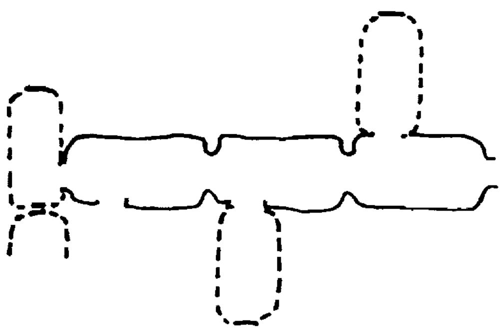

Fig. 6. Schematic plan of expansion potential of linear configuration.

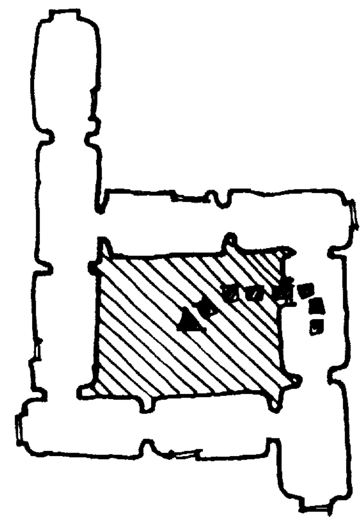

Fig. 9. Schematic plan of courtyard in-fill zone.

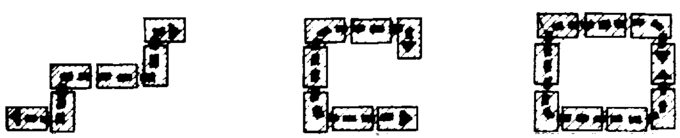

Fig. 7. Courtyard configuration is formed when linear path closes on itself.

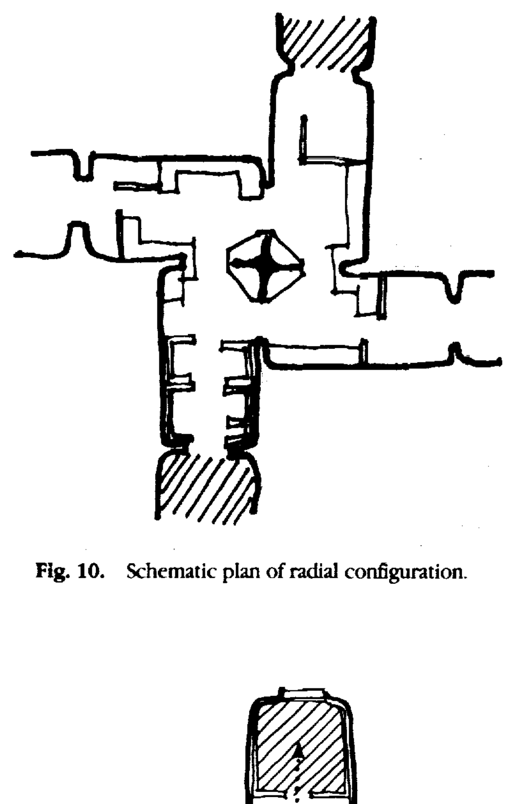

Fig. 10. Schematic plan of radial configuration.

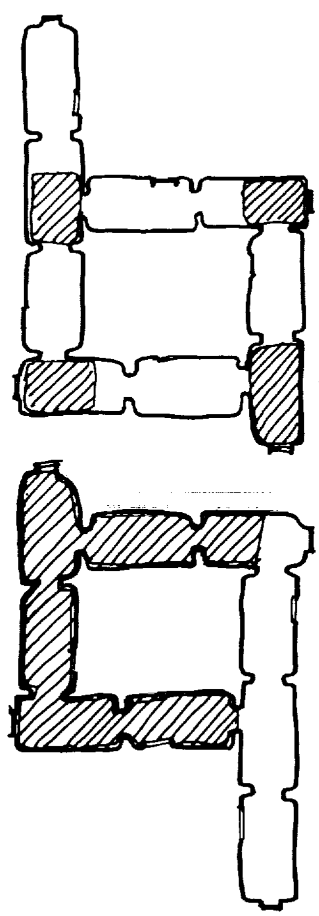

Fig. 8. Plan of courtyard functional organization options.

Fig. 11. Schematic plan depicting portion of a branching configuration.

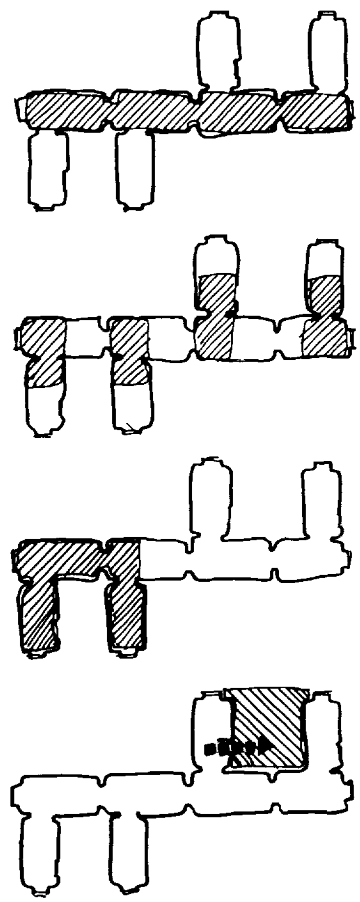

Fig. 12. Schematic plan of functional options for branching.

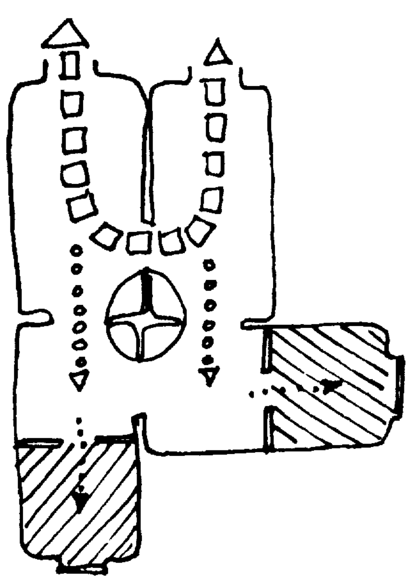

Fig. 13. Schematic plan of a cluster configuration.

Once a vertical module is introduced, it allows the configuration to expand in a different dimension. The expansion may be to connect the horizontal portion with a below-grade chamber, extend the base into a hill to a similar shelter deep in the regolith, or it may step the horizontal configuration to adjust to the lunar site.

EXPANSION OF THE BASE

In time it may be necessary to expand the initial configuration. The expansion can follow its natural progression of growth, or it may deviate. The deviation is dependent on the purpose of expansion, the construction materials, and the site conditions.

The expansion can take the form of an addition to the existing module, in-fill or underground expansion, or a separate facility. An addition to the existing module can be the coupling of new modules or an experimental space to the facility. Expansion by in-fill requires minimal additional material (Figs. 9 and 12). The adjacent modules become the exterior walls of the new area. Underground expansion, after initial testing, requires less additional material, only a circulation connection and interior finish. A separate facility connected through a circulation corridor to the existing requires a separate life-support system.

Each of these expansion types can be composed of similar modules or by an experimentally accepted construction method. Examples are actual on-site construction, transported fabric systems, and lightweight composites.

CONCLUSION

Independent of the lunar base's construction technique is the configuration's composition. Five basic configurations have been identified. From these basics innumerable arrangements can be derived. Each has advanatages and disadvantages. The optimum habitation will be a combination of the five basics, and its functional and operational requirements will be dependent on population sizes and activities.

Scientific parameters, typology, and combinatorics all play a key role in the selection of one design over another, but there still exists a human need that must be satisfied. This can only be accomplished by an architecture that combines science and art.

PREFABRICATED FOLDABLE LUNAR BASE MODULAR SYSTEMS FOR HABITATS, OFFICES, AND LABORATORIES

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MOBILE ARCHITECTURE

In spite of the advanced building technologies available today, concepts for mobile architecture are still limited to certain styles and applications. Prefabrication of buildings as complete units to be produced in a factory assembly line is only possible for certain types of stationary and mobile structures. However, the concept of a multistory, prefabricated, foldable, modular, structural unit has paved the way for almost unlimited sizes of building complexes to be prefabricated, preassembled, and shipped to a site for fast erection.

As a new approach, the prefabricated, modular, multistory, structural system was introduced for the first time for the purpose of pilgrim accommodation in the holy city of Makkah, Saudi Arabia. Characterized by its own unique patented construction and adaptable to special purposes, this concept is thought of as a new trend of mobile architecture. Several applications, such as residential apartments, offices, schools, and hospital systems, were proposed in multistory, foldable structures with the proper modifications to fit each purpose.

The overall size of each of these complexes depends upon required functions. Transportation of the individual prefabricated modules can be via trucks, trailers, or they can even be self-driven. It is also possible to launch these foldable units in special space vehicles, opening the door for a new era of human settlements on other planets. The units would be equipped with adequate facilities for initial settlements.

LUNAR SETTLEMENT

Prior to the establishment of lunar scientific and industrial projects, an innovative architectural approach for a self-contained lunar base is very much needed. The aim is to provide essential protection from solar flare radiation and to house preliminary facilities and personnel for the very initial stage when manual construction operations may be difficult.

In view of the various problems connected with the initial lunar settlement, the basic characteristics of such an approach were anlayzed as follows:

1. Independence from the expensive transportation of large and heavy building materials, site preparation machinery, and construction equipment and mechanisms.
2. Providing self-opening and site self-adjustment means to avoid using heavy and complex lifting mechanisms and site preparation machinery.
3. Self-erection of the entire base to minimize manual construction operations that require sizeable labor forces.
4. Utilization of local resources for galactic radiation flux protection, rather than terrestrial means, which must be imported, or the use of elaborate operations for tunnel digging or regolith massshielding.
5. A self-sufficient and self-contained lunar base of initial settlement, which is the precursor for additional research and development leading to a lunar city.
6. Providing conventional, habitable, and usable architectural spaces suitable to the 1/6-*g* environment.
7. Providing a modular, flexible, and expandable interconnected network for different future activities of a lunar base.

Evolving from the same philosophical approach of space settlements (i.e., space stations), it is believed that a prefabricated, self-contained system, providing all necessary life-support, is most appropriate for the initial stage of lunar colonization and manned settlement.

LUNAR BASE CONCEPTS

The first habitat and work station on the lunar surface undoubtedly has to be prefabricated, self-erecting, and self-contained. The building structure should be folded and compacted to the minimum size and made of materials of minimum weight. It must also be designed to provide maximum possible habitable and usable space on the Moon. For this purpose the concept of multistory, foldable structures was further developed.

The idea is to contain foldable structural units in a cylinder or in a capsule adapted for launching. Upon landing on the lunar surface, the cylinder of the first proposal in this paper will open in two hinge-connected halves (Fig. 1.1), while the capsule of the second proposal will expand horizontally and vertically in all directions (Figs. 2.2, 2.3, and 2.4). In both proposals, the foldable structural units will self-erect providing a multistory building with several room enclosures. The solar radiation protection is maintained through two regolith-filled pneumatic structures as in the first proposal (Fig. 1.4), or two regolith-filled expandable capsule shells as in the second one, which provide the shielding while being supported by the erected internal skeletal structure.

LUNAR BASE MODULE—FIRST PROPOSAL

The typical lunar base module proposed here is a cylinder 11 m in diameter and 40 m in length connected to a lunar shuttle vehicle.

The cylinder will longitudinally open to form two hinge-connected halves (Fig. 1.1). The self-opening mechanism will initially depend on lunar gravity for the two halves to open, and then the book-like opening will be completed by means of a built-in tension cable system. Horizontal stability is maintained by the same cable mechanism, assisted by a hydraulic footing system.

Each half of the cylindrical module contains four foldable structural units. Two units will form a pair that lean against each other when unfolded (Fig. 1.2). The typical structural unit is 9 m long and 3 m wide and consists of one rigid and four collapsible floors (Fig. 1.5). The typical floor-to-floor height is 2.5 m. The lower rigid floor of each structure provides two room enclosures,

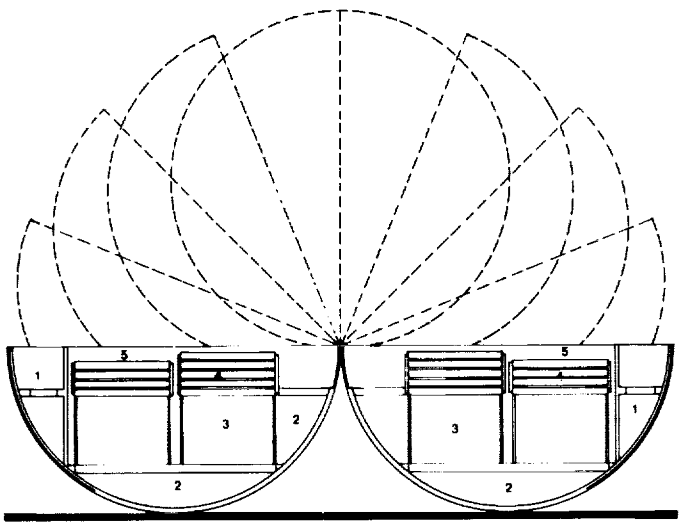

Fig. 1.1. Hinge-connected cylinder halves. Cross section showing folded structures. (Legend: **1.** Ground regolith pumped between two layers of pneumatic structures and flowing to shield module sides. **2.** Infrastructure and utility space. **3.** Rigid floor and fixed installations. **4.** Foldable structural units and room enclosures. **5.** Pneumatic structures or capsule shells.)

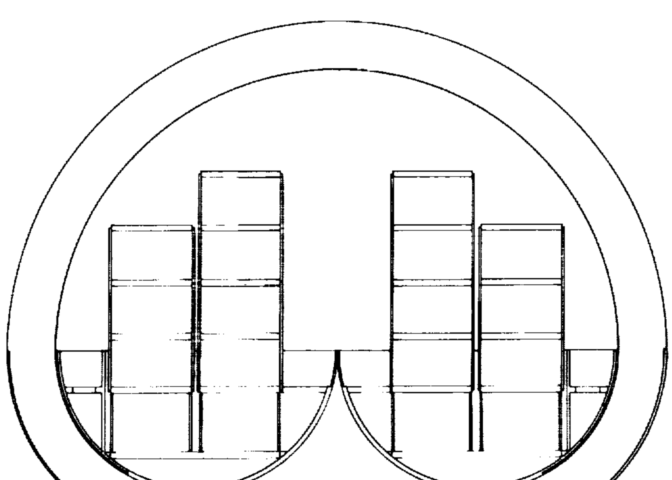

Fig. 1.2. Cylinder cross section. View during foldable structure self-erection.

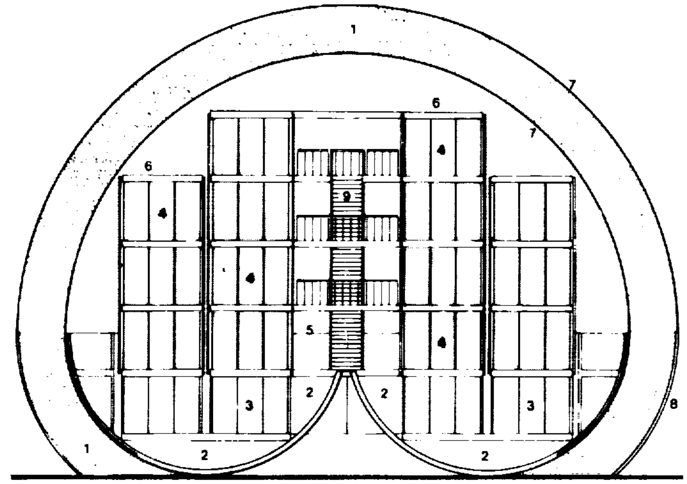

Fig. 1.3. Lunar base module interior elevation: (Legend: **1.** Ground regolith pumped between two layers of pneumatic structures and flowing to shield module sides. **2.** Infrastructure and utility space. **3.** Rigid floor and fixed installations. **4.** Foldable structural units and room enclosures. **5.** Central hall. **6.** Usable space. **7.** Pneumatic structures or capsule shells. **8.** Side shields to allow regolith to flow down. **9.** Staircases to be assembled.)

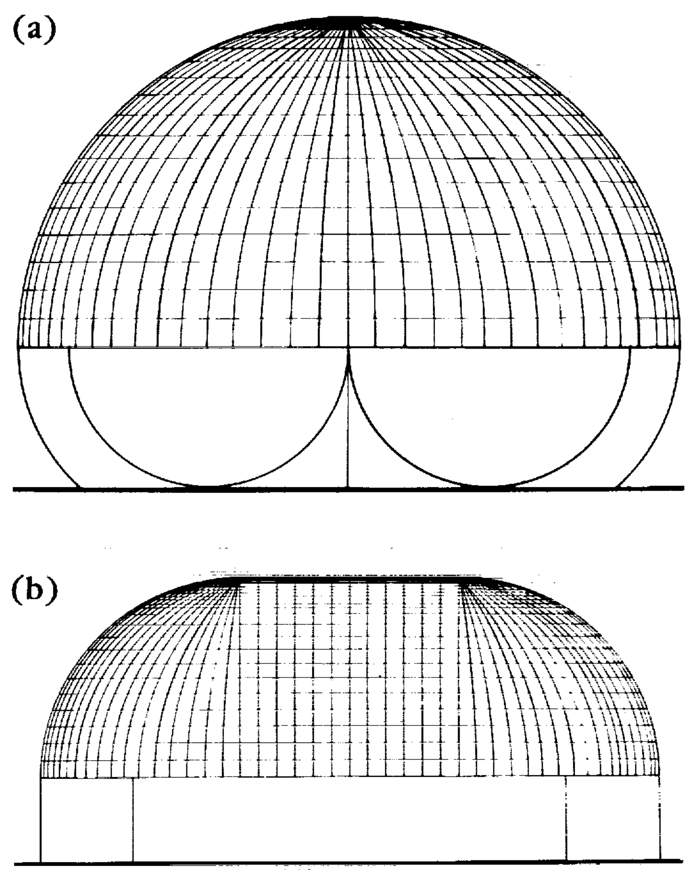

Fig. 1.4. View of cylinder after pneumatic structure inflation: **(a)** side elevation; **(b)** long side elevation.

while two additional rooms will be provided in each of the upper collapsible floors. Lightweight wall panels will unfold from the ceilings to form enclosed rooms (Fig. 1.3). Following the self-erection of the structural units (Figs. 1.2 and 1.5c), prefabricated stairwell components are assembled inside the base for vertical access (Fig. 1.6). To facilitate the unfolding process of the structural units without any additional external lifting facilities, the self-erection mechanism is built in.

The size of the lunar base module can be varied depending upon the launching capacity. Similarly, the size of the foldable structures can be varied to accommodate the internal carrying capacity of the lunar base module.

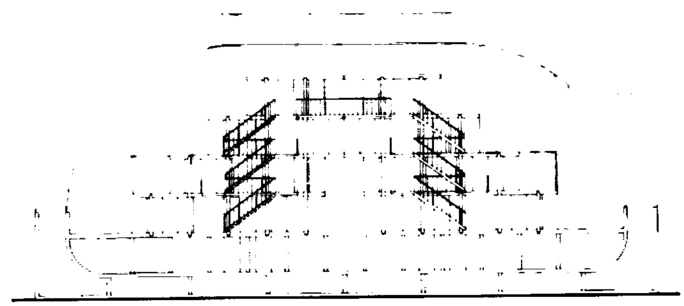

Fig. 1.6. Lunar base longitudinal section.

FIRST PROPOSAL COLLAPSIBLE STRUCTURE DESIGN

The aluminum structural units consist of several collapsible floor slabs 3.2 m wide, 6.5 m to 12.5 m long, and 15 cm thick. The preferred arrangement would have four collapsible floors plus a base platform.

(a)

(b)

(c)

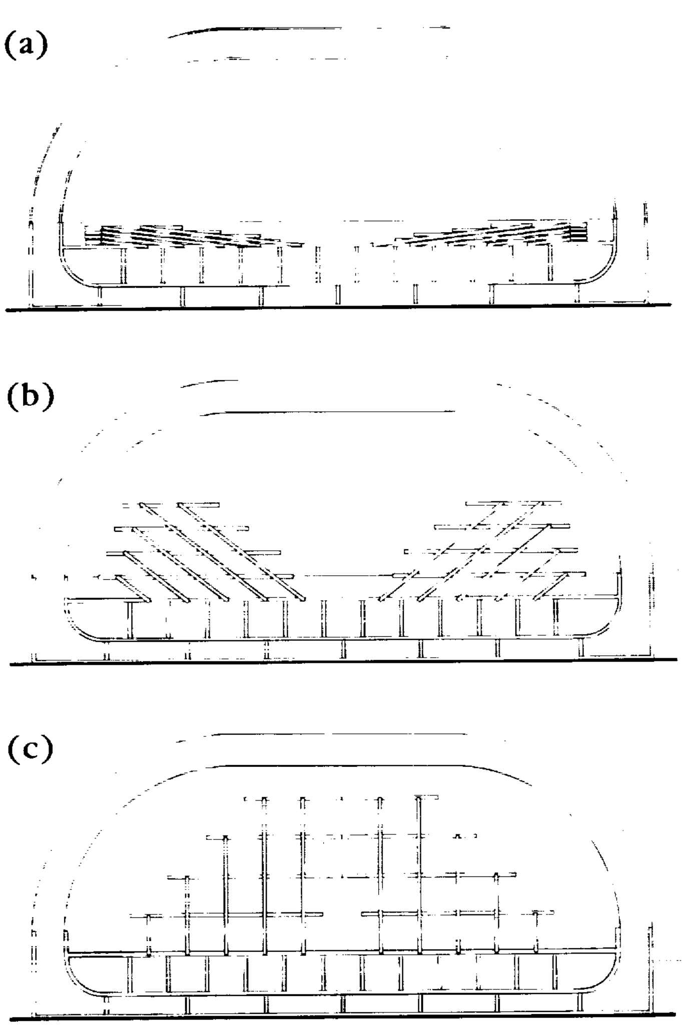

Fig. 1.5. Cylinder longitudinal section: **(a)** view before foldable structure self-erection; **(b)** view during foldable structure self-erection; **(c)** view after foldable structure self-erection.

The floor slab consists of two main parallel longitudinal beams of 6.5 m to 12.5 m in length and 15 cm in height connected by secondary beams 3 m in length. The decking of these slabs could be achieved through suitable lightweight, rigid material, such as fiberglass, with proper final finishing.

Adjacent floor slabs are connected to each other with supportive columns connected to the longitudinal beams by pivotal hinged joints. The hinge connection allows for scissor-like rotations (Fig. 1.5b). Whenever the unit is folded inside the cylinder halves, the aluminum slabs stack neatly above each other in a horizontal position (Fig. 1.5a).

In the unfolded position, a supportive column is connected to the floor slab every 2.5 m. In the same way, every column is connected with a main longitudinal floor beam every 2.5 m in height. This scissor-like grid of vertical columns intersecting with horizontal floor slabs (as seen from the side elevation of the unit; Fig. 1.5c) allows for the rotation of columns and maintains the horizontal stability of the floor slabs, thus achieving the system's foldable and collapsible nature.

A self-winching mechanism is attached to the top floor slabs of each pair of foldable units. A cable connects the pair of winches to form one lifting mechanism. When the cable is wound around the winch, the folded structures will automatically be raised to their upright positions. Other self-uprighting mechanisms could also supply the required forces to raise the structures.

The rigid floor upon which each foldable unit is erected is integrated structurally with the cylinder half where it is contained to form a base platform (Figs. 1.1 and 1.5a). This base platform will permanently accommodate laboratories, work stations, fixed utility machinery, and infrastructures in eight 3×5-m rooms. For the purpose of seating astronauts and crew members during the trip, some of these rooms will be used as seating compartments. Some spaces between the rigid floor and the cylinder half will temporarily be used as storage areas for the furniture and equipment of the upper collapsible floors. The rigid floor is reached through a permanent staircase leading up to the main central space of the lunar base (Figs. 1.3 and 1.6).

Corridors formed between the rooms of the base platform are 2.5 m wide and 2.3 m high. Through the use of square telescope sliding corridors, adjacent lunar base modules can be interconnected. This width and height will allow for a future motorized circulation network when many more modules are interconnected within the future lunar city layout.

An erection sequence for a module, culminating in the installation of regolith shielding, is illustrated in Fig. 1.7. Models on a simulated lunar surface are shown in Fig. 1.8.

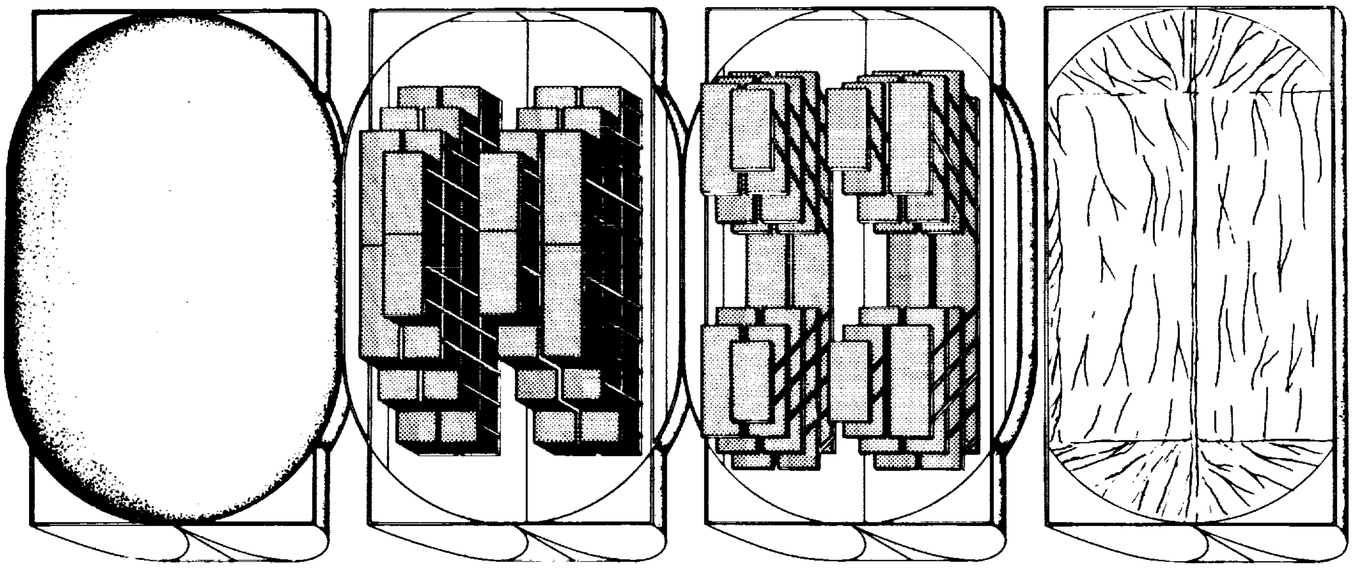

Fig. 1.7. Prefabricated lunar base modular systems (habitats, offices, and laboratories).

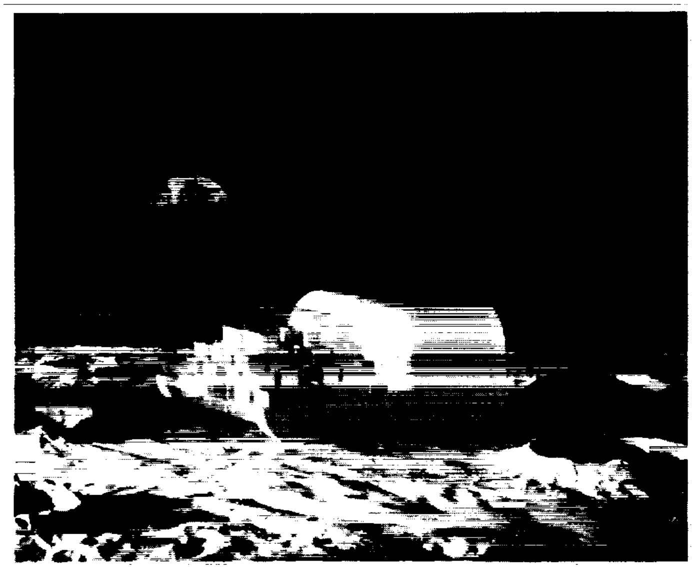

Fig. 1.8. Models of modules on a simulated lunar surface.

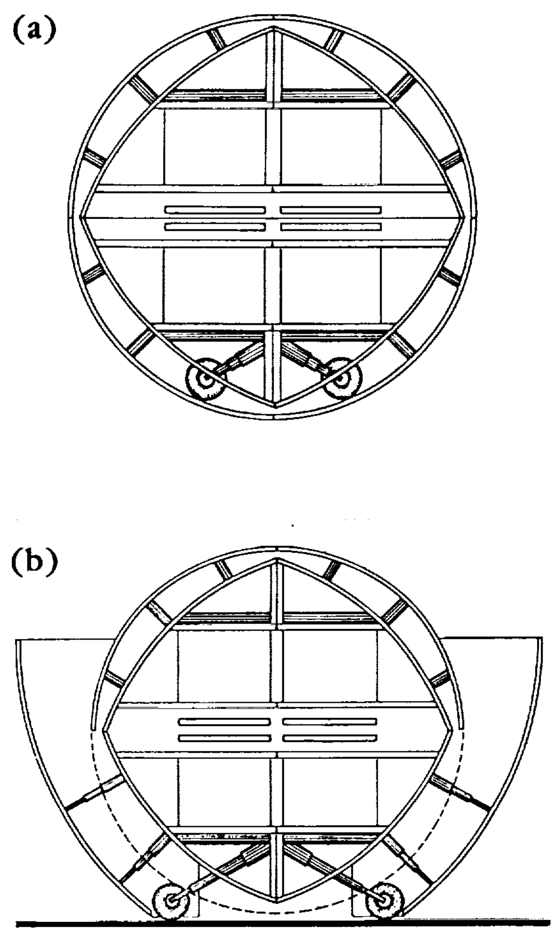

Fig. 2.1. Section of folded capsule **(a)** before landing; **(b)** after landing.

LUNAR BASE MODULE—SECOND PROPOSAL

The typical lunar base module in this proposal is a capsule 12 m in diameter (Fig. 2.1) and 30 m in length with two half-sphere ends. The exterior cylindrical and spherical envelopes consist of two expandable shells. The idea is to have this capsule expanded and amplified to provide larger habitable and usable space on the Moon. The outer shell will expand to make a 42-m-long capsule with a diameter of 24 m (Figs. 2.3 and 2.4). The inner shell will provide an internal sealed capsule 18 m in diameter and 36 m in length. The 2-m to 3-m cavity between the two expanded shells will be filled with either ground regolith or bags that contain the lunar soil.

The original capsule is longitudinally split into four quarters and recombined through an internal skeletal structure and a set of horizontal and vertical hydraulic pistons. In its folded position, it contains two rigid floors with three folded collapsible floors in between (Fig. 2.1).

Prior to landing, the lower parts of the external capsule shell will slide and open to allow for proper landing with the assistance of the lunar shuttle vehicle jets (Fig. 2.2a). Immediately after landing on the lunar surface, the capsule will expand in two opposite horizontal directions (Fig. 2.2b). The two halves, 6 m apart, will maintain their structural connection and stability. The two half-sphere ends will then expand horizontally in the other directions (Fig. 2.2c).

The self-erection of the folded structure will make it rise and complete the vertical expansion of the capsule, providing a multistory building structure (Figs. 2.3 and 2.4).

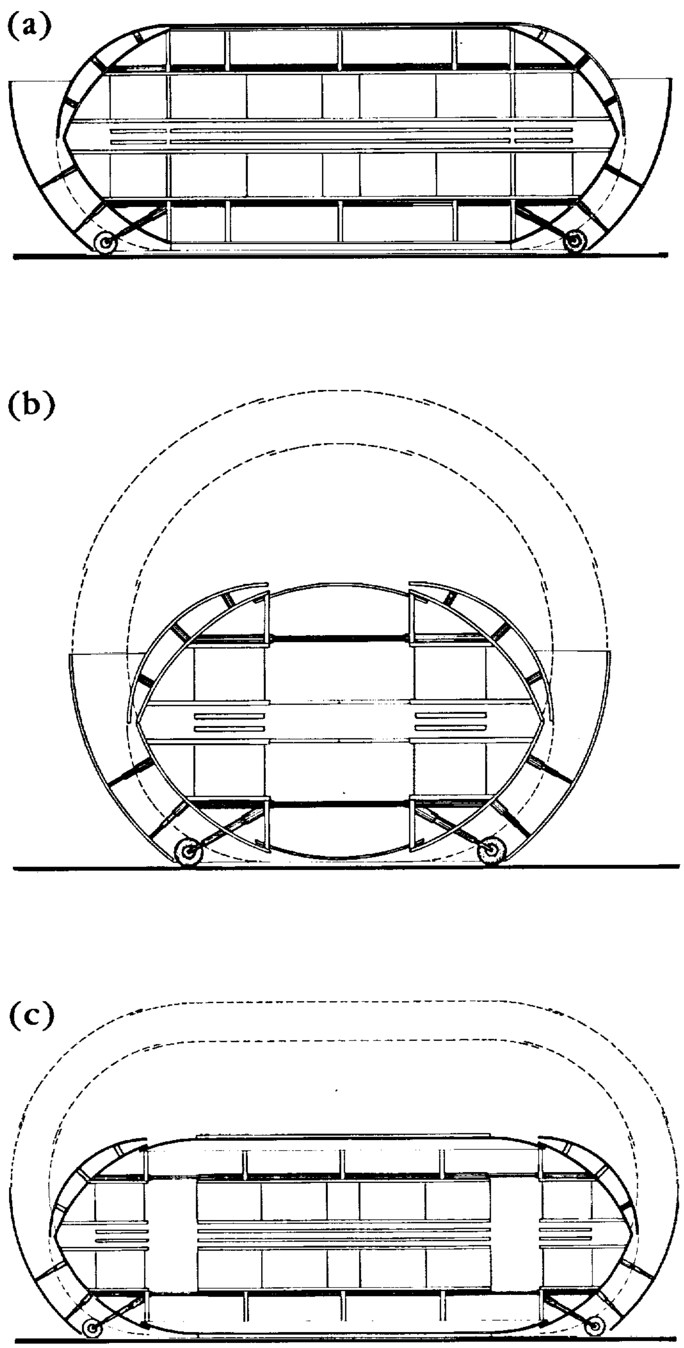

Fig. 2.2. **(a)** Longitudinal section before horizontal expansion. **(b)** Section showing capsule during horizontal expansion. **(c)** Longitudinal section after horizontal expansion.

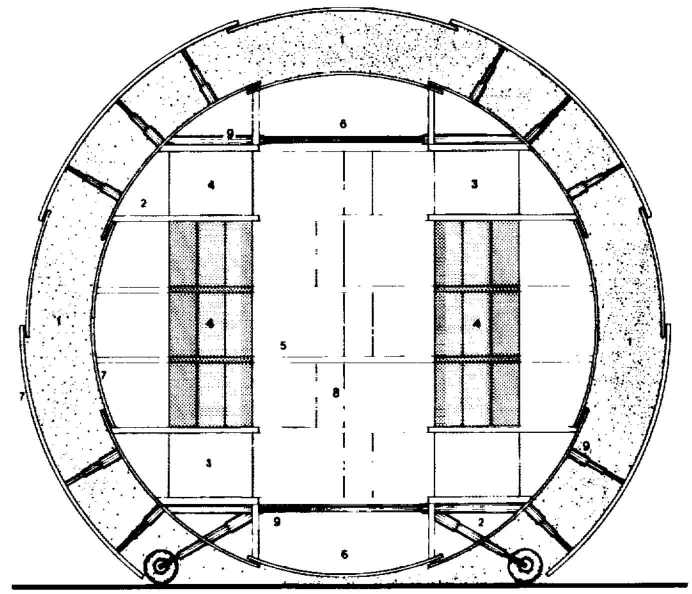

Fig. 2.3. Cross section. (Legend: **1.** Ground regolith pumped between two layers of pneumatic structures and flowing to shield module sides. **2.** Infrastructure and utility space. **3.** Rigid floor and fixed installations. **4.** Foldable structural units and room enclosures. **5.** Central hall. **6.** Usable space. **7.** Pneumatic structures or capsule shells. **8.** Staircases to be assembled. **9.** Hydraulic piston.)

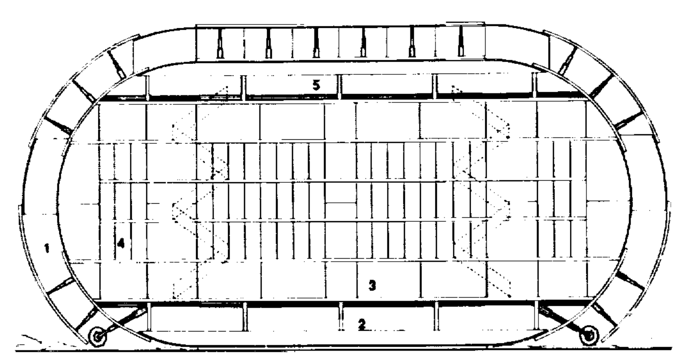

Fig. 2.4. Longitudinal section showing foldable structures after self-erection (vertical expansion). (Legend: **1.** Ground regolith pumped between two layers of pneumatic structures and flowing to shield module sides. **2.** Infrastructure and utility space. **3.** Rigid floor and fixed installations. **4.** Foldable structural units and room enclosures. **5.** Usable space.)

SECOND PROPOSAL COLLAPSIBLE STRUCTURE DESIGN

The aluminum structural units of this proposal are integrated with the internal shell of the capsule. They consist of two rigid floors 6 m to 10 m wide (Fig. 2.1a) and 24 m to 28 m long (Fig. 2.2b) in the folded position. After the expansion of the system, these floors will become 13 m to 14 m wide (Fig. 2.3) and 31 m to 34 m long (Fig. 2.4). There are three collapsible floors folded between the two upper and lower rigid floors (Figs. 2.1 and 2.2b). The vertical unfolding of the structure will generate the space of these three collapsible floors of 16 m to 17 m in width (Fig. 2.3) and 30 m to 34 m in length (Fig. 2.4).

The rigid floor slabs consist of 15-cm-thick main and secondary beams. The collapsible floor slabs consist of two main parallel longitudinal beams 24 m in length and 15 cm in height connected by secondary beams 3 m in length. The decking of these rigid and collapsible slabs could be achieved through suitable lightweight, rigid materials such as fiberglass, with proper final finishing.

Rigid floor slabs are connected by rigid, supportive columns connected to the longitudinal beams. In the case of the collapsible floors, the supportive columns are connected to the beams by pivotal hinged joints. Each column will bend in two halves with the help of a middle pivotal hinged joint. This bending ability of the column will give the structure the chance to collapse and fold. The self-erection mechanism of the system is achieved through a set of hydraulic pistons that fulfill the vertical expansion providing the total of five typical floors 2.5 m in height.

COLLAPSIBLE FLOOR ROOM ENCLOSURES

Single-room enclosures are created in each of the collapsible floors of the foldable structures in both proposals. This could be achieved through the use of wall panels stored in compartments in the floor/ceiling slabs. One example is to have the panels unfold from a ceiling slab to be fixed to a floor slab providing a typical 3×4-m or 3×5-m room. These rooms could be easily elongated, combined, or even redesigned to fit required functions. They will be furnished as habitats, offices, work stations, or medical facilities.

The wall panels are prefabricated out of lightweight solid or transparent synthetic materials, as desired. Foldable furniture could also be built in for greater packing density and added compatibility of the entire structure. The floor will be finished with suitable material for each purpose. Ceilings will accommodate the lighting system, air conditioning, and any other utility infrastructure. A total of 60 single-room enclosures could be available in the first proposal, occupying 860 sq m. There are an additional 255 sq m of usable terrace space, which is the space between the upper floor roof slab and the curved pneumatic canopy. Adding the corridors and the open space results in a total usable area of 1520 sq m. In the second proposal, a total of twenty-four 3×3-m and 3×4-m single-room enclosures in the rigid floors is possible and 36 single-room enclosures of the same area in the collapsible floors. There are an additional 1880 sq m of usable open space in different floors. The total area provided in this proposal is 2640 sq m.

STAIRS

Stairs are necessary for personnel movement between floors. Following the self-erection of the foldable structures in both proposals, staircases will be positioned between the two inner pairs of the structural units. Due to the lightness and design simplicity of the stair components, manual assembly could be performed by the crew members inside the lunar base modules.

GALACTIC RADIATION FLUX SELF-PROTECTION

The galactic radiation flux self-protection concept requires ground regolith to be pumped in and to fill a space created between the two outer layers of the pneumatic structure in the cylinder proposal or the cavity between the internal and external shells in the capsule proposal. This concept utilizes the lunar soil as a shield; thus, expensive and heavy shielding does not have to be transported from Earth. While being affixed to the exterior edges of the longitudinal halves of the cylinder, the two layers of the pneumatic structure are inflated prior to the self-erection of the structural units. Ground regolith is then pumped into the cavity between the two layers until it is completely filled. The load created from the regolith shielding canopy is transferred by the erected skeletal structure down through the vertical supports to the lunar surface. The cylinder halves are designed to accept a flow of ground regolith to fill their sides insuring complete shielding. Ground regolith could be provided by a special vehicle designed to drive on the lunar surface and collect, grind, and pump regolith through special pipes in order to fill the cavity in the inflated pneumatic structures. The outer layer of the pneumatic structure could be designed so that it is expandable in order to increase the size and thickness of the cavity if additional shielding is required. Another way to achieve the same goal is to collect regolith in bags and stack them in the cavity between the pneumatic structures. The same approach could be followed in the second proposal, utilizing the cavity between the two capsule shells for galactic radiation flux self-protection.

MODULE VARIATIONS

Additional future modules of both proposals could be entirely devoted to specialized functions such as hospitals, medical faciliities, specialized laboratories, and prefabricated plants and factories. By eliminating some floor slabs and keeping only the skeletal structure, agricultural fields could be created in larger span enclosures. In the same way, aquariums and zoological gardens could be established.

CONTINUATION OF RESEARCH AND DEVELOPMENT

The initial lunar settlement of prefabricated foldable modules as proposed in this paper is thought of as providing a pioneer settlement that will pave the way for later phases of lunar-based construction capabilities. The proposed designs will accelerate the planning process for early manned presence on the Moon; therefore, it is worth further investigation to test various aspects and concepts outlined above. A full-scale prototype could be built for testing the self-opening, the site self-adjustment, and the self-erection mechanisms with respect to the 1/6-*g* lunar environment. In addition, various materials can be experimented with to provide a structure that uses lightweight materials to provide sufficient structural strength.

CONCRETE LUNAR BASE INVESTIGATION

N93-17445

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This paper presents results of structural analyses and a preliminary design of a precast, prestressed concrete lunar base subjected to 1-atm internal pressure. The proposed infrastructure measures 120 ft in diameter and 72 ft in height, providing 33,000 sq ft of work area for scientific and industrial operations. Three loading conditions were considered in the design: (1) during construction, (2) under pressurization, and (3) during an air-leak scenario. A floating foundation, capable of rigid body rotation and translation as the lunar soil beneath it yields, was developed to support the infrastructure and to ensure the airtightness of the system. Results reveal that it is feasible to use precast, prestressed concrete for construction of large lunar bases on the Moon.

INTRODUCTION

Scientific and industrial operations on the Moon require suitably shielded structures. Small structures for limited scientific activities may be constructed from structural elements prefabricated on Earth and transported to the Moon. However, large industrial structures on the Moon must be constructed using *in situ* lunar materials. A potential material for such construction would be concrete.

Feasibility studies on lunar cement production (*Lin*, 1987) and experimental studies on lunar concrete (*Lin et al.*, 1987) provide scientific evidence that quality concrete can be produced from lunar materials. Concrete, a thermally and chemically stable construction material, can withstand the harsh lunar environment.

One of the innovatively designed elements that contributes to the feasibility of constructing a large concrete lunar base is curved panels. A curved panel, with its concave side facing outward and supported by columns along its straight edges, will develop compressive stresses when subjected to internal air pressure. Concrete performs very well under compressive force. This design will make better use of concrete than a hoop prestressed concrete tank does. Other innovative concepts such as tension-compression columns and a floating foundation contribute greatly to the successful design of the proposed concrete lunar base. Engineering data on lunar concrete needed in structural analyses were taken from test results obtained in *Lin et al.* (1987).

OBJECTIVES

The objectives of the study are to perform structural analyses and to evaluate the economic and technological feasibility of a preliminary design for lunar concrete structures.

STRUCTURE

Figure 1 shows the proposed three-story concrete structure with a 24-ft story height and a 120-ft diameter. The reason for choosing this particular height is to provide work space suitable for industrial operations and flexibility of converting it into two 12-ft stories if more living quarters are needed. The core structure located at the center of the system is a self-contained cylindrical unit with an inside diameter of 20 ft. This unit provides astronauts with safety shelters in case air leaks occur in the base.

Figure 2 shows major structural elements of the proposed lunar base. These elements, based on structural performance, can be classified into four types: compression, flexural, tension-flexural, and tension-compression. According to types, the structural elements include curved panels, floor/ceiling slabs, roof/ground floor slabs and exterior columns, core cylinder, and tension-compression columns respectively.

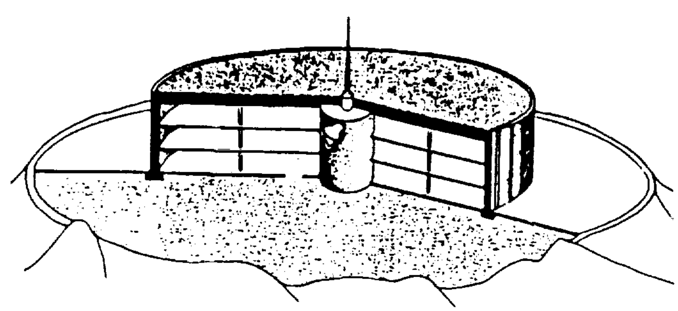


Fig. 1. Proposed three-level concrete lunar base.

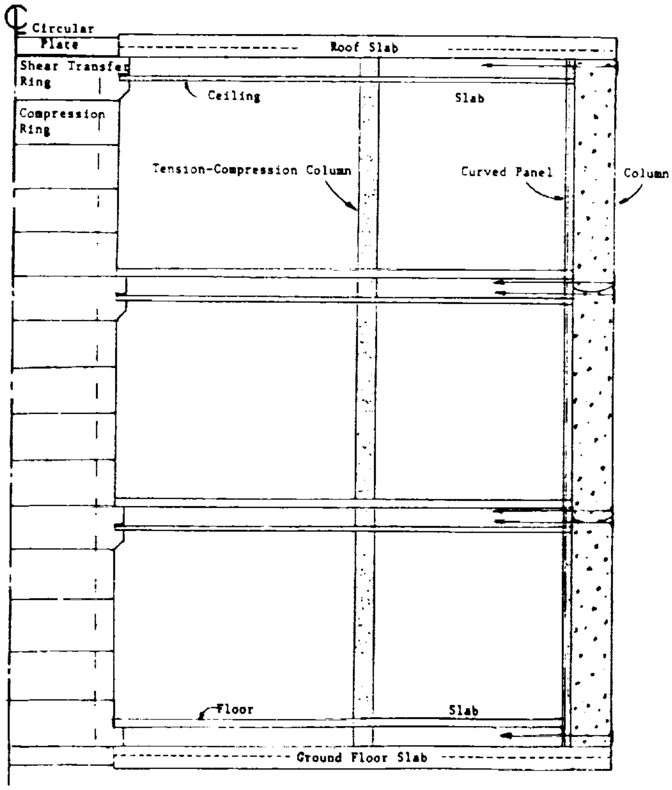


Fig. 2. Major structural elements.

The determination of element sizes considers the workable space in the future concrete plant and the lifting capacity of future lunar robots (assumed 20,000 lb). The space shuttle's external fuel tank measures 27 ft in diameter and 154 ft in length. Two fuel tanks will make an ideal concrete plant. With a presumption of an external tank utilization, a maximum width of 24 ft for the proposed structural elements is recommended.

LOADS

Because there is no atmosphere on the Moon, wind loads can be neglected in the design. Although thousands of moonquakes occur each year, their energy is generally less than a Richter magnitude of 2. Consequently, the effect of seismic loads of this scale is considered insignificant. However, the largest moonquake recorded is approximately a magnitude of 4 (*Williams and Jadwick,* 1980). The occurrence frequency and locations are not reported. All design loads, except internal air pressure, were multiplied by a factor of $\frac{1}{6}$ to account for the weak lunar gravity.

For industrial operations, a live load of 600 psf (100 psf on the Moon), the heaviest live load recommended by the Uniform Building Codes (*The National Building Code,* 1985), was used. The air volume in the proposed lunar base, weighing 10 kips on the Moon, was also included in design computations.

The dead loads include a suggested 18 ft of lunar soil (300 psf) on the roof for radiation shieldings, and 25 lb/cu ft of concrete.

Load factors of 1.7 for live loads and 1.4 for dead loads recommended by ACI 318 building codes (*Building Code Requirements for Reinforced Concrete,* 1983) were applied in the design to increase the safety factor. To be conservative, the internal air pressure was treated as live load.

One of the major concerns in designing lunar structures would be the internal air pressure due to the vacuum condition on the Moon. One-atmosphere pressure, equivalent to a 2100-lb force acting on a square foot area, exceeds by 20 times the live load required by the Uniform Building Codes for an auditorium structure on Earth. Subsequently, a factored air pressure of 3600 psf was used in structural analyses.

The proposed lunar base is of a precast, prestressed concrete structure. Prestressing will be applied to the structural elements by means of a post-tensioning procedure.

Three loading conditions were analyzed to ensure structural safety. The conditions and their design loads are (1) during construction: factored dead loads only; (2) in service: factored dead and live loads, air pressure, and prestressing; and (3) in the event of air leaks: factored dead and live loads and prestressing.

CONCRETE DATA AND DESIGN GUIDES

Concrete data needed in the design computations were taken from test results obtained by *Lin et al.* (1987). They include the compressive strength of 10,970 psi, flexural strength of 1206 psi, modulus of elasticity of 3.1×10^6 psi, and thermal expansion coefficient of 2.9×10^{-6} in/in/F.

Lunar ilmenite is a candidate material for manufacturing steel. However, no data on lunar reinforcing steel and strands are presently available. For convenience, the existing steel data commonly used in design were assumed in the computations.

The design procedures basically followed the ACI 318-83 building codes (*Building Code Requirements for Reinforced Concrete,* 1983) for reinforced and prestressed concrete elements. The PCI Design Handbook was used as a guide in determining prestressing and selection of tendons.

ANALYSES AND PRELIMINARY DESIGN

Connections between precast concrete members are often the most important factor influencing the structural analysis of the proposed lunar structure. Careful considerations for connections were exercised to avoid stress concentrations and air leaks. Suitable elastomeric pads have been considered to cushion members at joints. An epoxy coating or sealant that hardens without oxidation can be applied on the interior surfaces of concrete elements to prevent air permeation. The use of elastomeric pads and sealants requires an in-depth study and is beyond the content of this program.

The use of pads at joints will undoubtedly allow connected elements to rotate. A member that rotates at supports may be treated as a statically determined structure. The computations for moments and shears for a simply supported member yield conservative results.

Most structural elements for the proposed infrastructure are prestressed concrete members. For a prestressed member, the moments induced by the post-tensioning are directly proportional to the eccentricity of the centroid of tendons with respect to the neutral axis of the members. When members are joined together by post-tensioned tendons, some degree of joint rigidity will develop. The moments on members in the vicinity of the joint due to post-tensioning may no longer be directly proportional to the tendon eccentricity (*Post-Tensioning Manual,* 1982). Moments that are induced by the primary moments are called secondary moments. A secondary moment may affect the computational procedures for deflections. This study does not take into account the secondary moments and deflections.

The design of prestressed concrete members was carried out based on the concept that if there are no tensile stresses in the concrete, there can be no cracks. To achieve zero tensile stresses in concrete members, high-strength steel can be used to carry tension while concrete carries compression to form a resisting couple against the bending moments. A familiar equation (*Lin*, 1967) that satisfies these assumptions is presented as the following

$$f = \frac{F}{A} \pm \frac{Fey}{I} \pm \frac{My}{I} \qquad (1)$$

where f = stress, F = prestress force, e = eccentricity between cgs of prestress and neutral axis, M = moment due to loads, y = distance from neutral axis to the point where stresses are to be calculated, and I = moment inertia of the cross section.

The load balancing method (*Lin*, 1967) was also used in determining prestresses to balance the applied loads. The balancing load, v, produced by a parabolic tendon can be expressed

$$v = \frac{8Fh}{L^2} \qquad (2)$$

where h is the sag of tendon profile and L is the span length.

Designing prestress to counteract loads or bending moments for structures on Earth is straightforward. However, designing lunar structures that may experience air leaks in which the air pressure diminishes and the prestress in concrete remains could be complicated. The retained prestress may violently break the members if concrete is not properly reinforced. The following briefly describe the designs of structural elements (details of designed elements are included in *Lin et al.*, 1988).

Panels

Figure 3 shows a plan view of a reinforced concrete panel simply supported by columns at both ends. A curved panel subjected to a factored atmospheric pressure (3600 psf) on its outer surface will develop compressive stresses. The reaction at supports can be expressed as R = 3600 ϒ, where ϒ is the radius of the panel. With this equation and section 14.5.2 of ACI 318-83, a 6-in-thick panel with a 20-ft radius was selected. The minimum reinforcement provision (section 7.12 of ACI 318) yields the required reinforcing steel.

Roof Slabs

The roof system of the proposed infrastructure consists of 33 identical fan-shaped slabs. Each slab is composed of three 12 × 16-in radial joists and a 10-in-thick deck with variable widths.

Dead loads include the 18-ft-thick lunar soil and concrete weights, while live loads include envisioned solar energy collectors on the roof and the 1-atm internal air pressure applied to the bottom surface of the slab. Figure 4 provides the loading information. The linear variation of loads is attributed to the

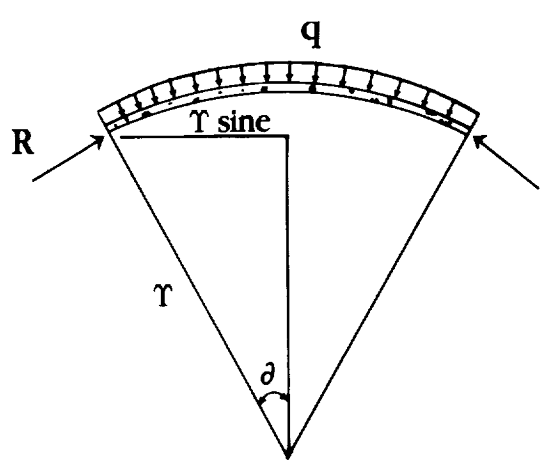


Fig. 3. A plan view of a curved panel.

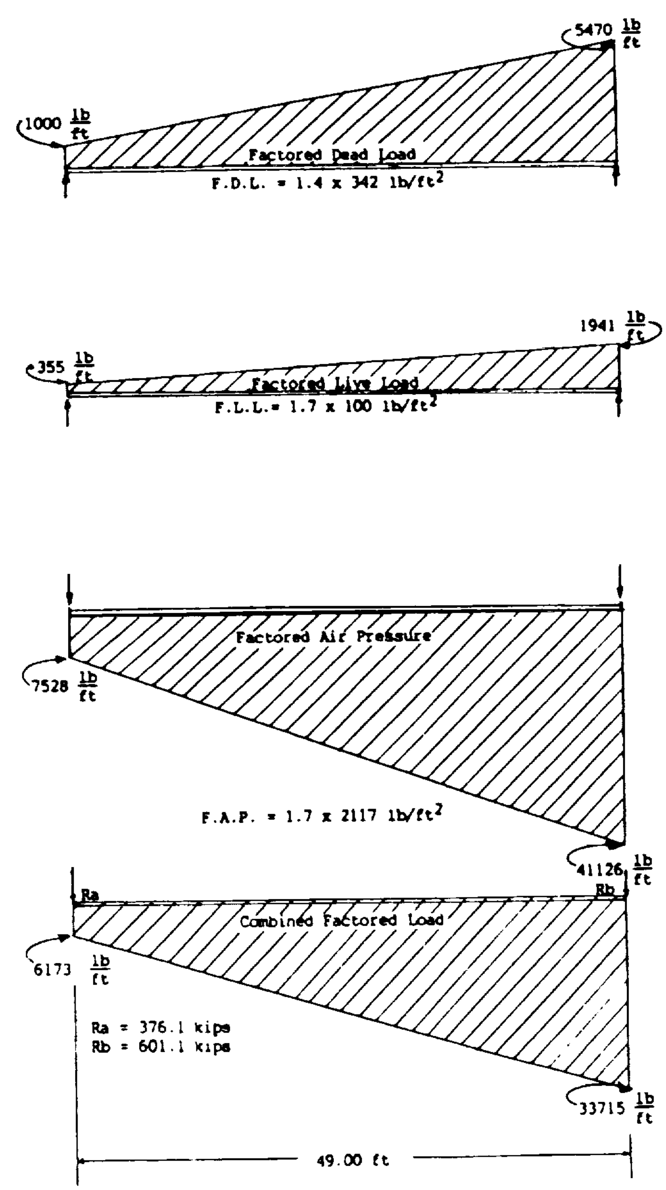


Fig. 4. Loads on roof slabs.

variable width of the slab. Because the air pressure is much greater than the sum of dead and live loads, the bending moments due to the combined loads are dominated by air pressure and are too big for a practically sized slab to resist. To resolve this problem, possibilities of using in-span restraints to confine upward deflections of the slab were investigated in hopes of reducing the bending moments. It was found that a downward force of 407 kips at the maximum bending moment location, 27.2 ft from the interior support, will decrease the maximum bending by 63%. The 407-kip force will be supplied by the so-called tension-compression column, which will be discussed in a later section. Figure 5 shows the computed moment diagrams of the slab without and with a tension-compression column (see the parabolic curve and triangular curve respectively).

It is tedious to determine the critical section of a roof slab with variable widths subjected to discontinuous nonlinear moments. To simplify the computations, ratios of the bending moment to the moment inertia at selected sections were computed. By comparing the calculated ratios, the section at 16 ft from the interior support was found to be the most critical and was used in the design. The designed roof slab is shown in Fig. 6.

Ground-Floor Slabs

The behavior of the ground-floor slabs during service is similar to that of the roof slabs. However, the bending moments developed in the ground-floor slabs are greater than those of roof slabs. The increase in bending moments results from there being no 18-ft-thick lunar soil placed on the ground-floor slab to counteract the air pressure, and the live and dead loads act in the same direction as the air pressure. Accordingly, deeper joists and more prestressing strands were required for the ground-floor slabs. Figure 7 shows a designed ground-floor slab.

Ceiling Slabs and Floor Slabs

Like the roof system, the ceiling and floor systems each consist of 33 one-way slabs. However, their loading conditions are quite different. Because they are completely enclosed by the infrastructure, the air pressurization in the lunar base has no effect on them. The equilibrium condition of air pressure on these slabs accounts for much smaller loads and simplifies the design procedures.

Nevertheless, ceiling slabs were designed to carry horizontal steel tendons (25 psf) during construction, while floor slabs were designed to carry heavy industrial equipment (100 psf). With these assumed live and dead loads, the maximum bending moments and shears for both types of slabs were calculated.

The strength design method recommended by ACI building codes was followed to calculate nominal moment and shear strengths, and to determine the principal reinforcement and stirrup details.

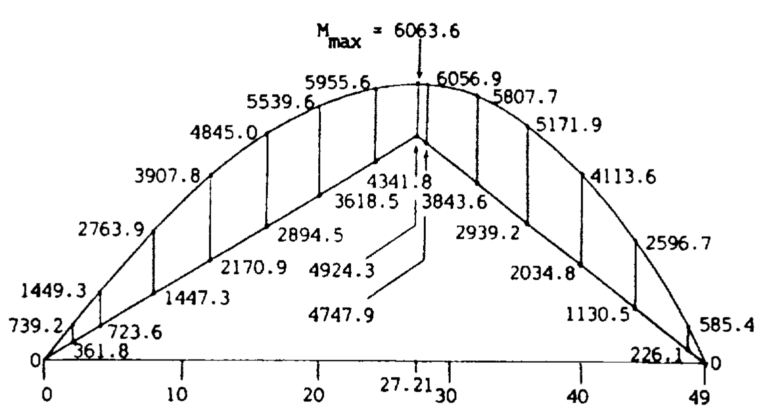
Tension-Compression Columns

In the case of air leaks, the air pressure in the structure diminishes, but the 407-kip concentric prestressing force remains on the roof and ground floor slabs. This force is sufficient to cause either slab to collapse if no other schemes were designed to counteract the force. To solve this engineering problem, a concrete column with unbonded tendons was designed to restrict slab deflections. This design enables the column to perform dual functions. The column acts as a tension member in a pressurized condition and as a compression member in the event of air leaks. Since there are no documented references on such columns available in concrete structural textbooks, they are referred to as "tension-compression columns."

As discussed, the tension-compression columns were designed to decrease the bending moments and to minimize the thicknesses of the roof and ground floor slabs. Conceivably, the use of tension-compression members offers economic benefits to the proposed lunar base construction.

If a tension-compression column were used for each roof slab, the spacing between successive columns would be about 5.5 ft, which is too small for industrial operations. For this reason, curved beams about 22 ft in length, extending across three slabs, will be installed along the circular path at a distance of 27.2 ft from the outer face of the core structure, on the roof slabs and under the ground floor slabs, to provide a force transfer mechanism to the tension-compression columns. This design will increase the spacing between columns to about 25 ft.

The design of the proposed tension-compression columns to resist tension is straightforward. The required tensile force of 1220 kips ($=407\times3$) in each column determines the needed number of strands (30 strands of 0.6-in diameter). In an air-leak scenario, the restraining force immediately transfers to the



X	M_2
0.00	0.0
2.00	361.8
4.00	723.6
8.00	1447.3
12.00	2170.9
16.00	2894.5
20.00	3618.5
24.00	4341.8
27.21	4924.3
28.00	4747.9
32.00	3843.6
36.00	2939.2
40.00	2034.8
44.00	1130.5
48.00	226.1
49.00	0.0

Difference $(M_1 - M_2)$
0.0
377.4
725.7
1316.6
1736.9
1950.5
1921.1
1613.8
1139.3
1309.0
1964.1
2232.7
2078.8
1466.2
359.3
0.0

Fig. 5. Moment diagrams.

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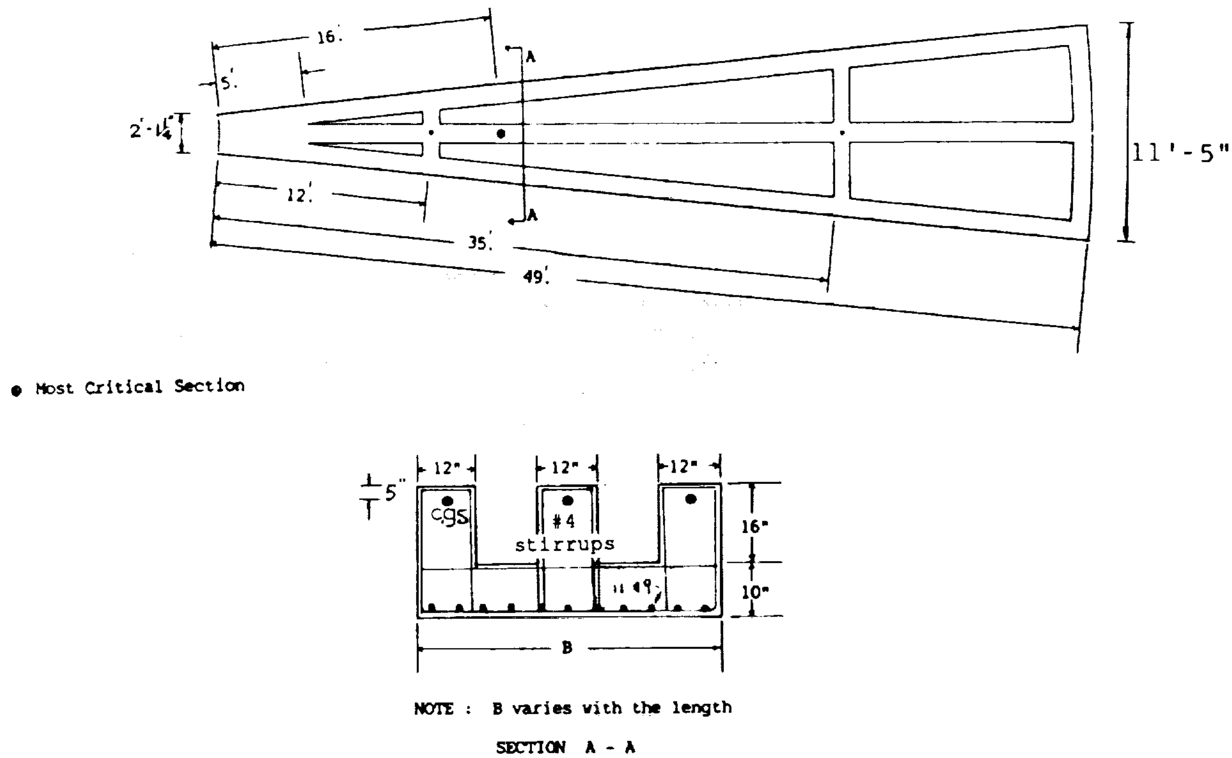

Fig. 6. A designed roof slab.

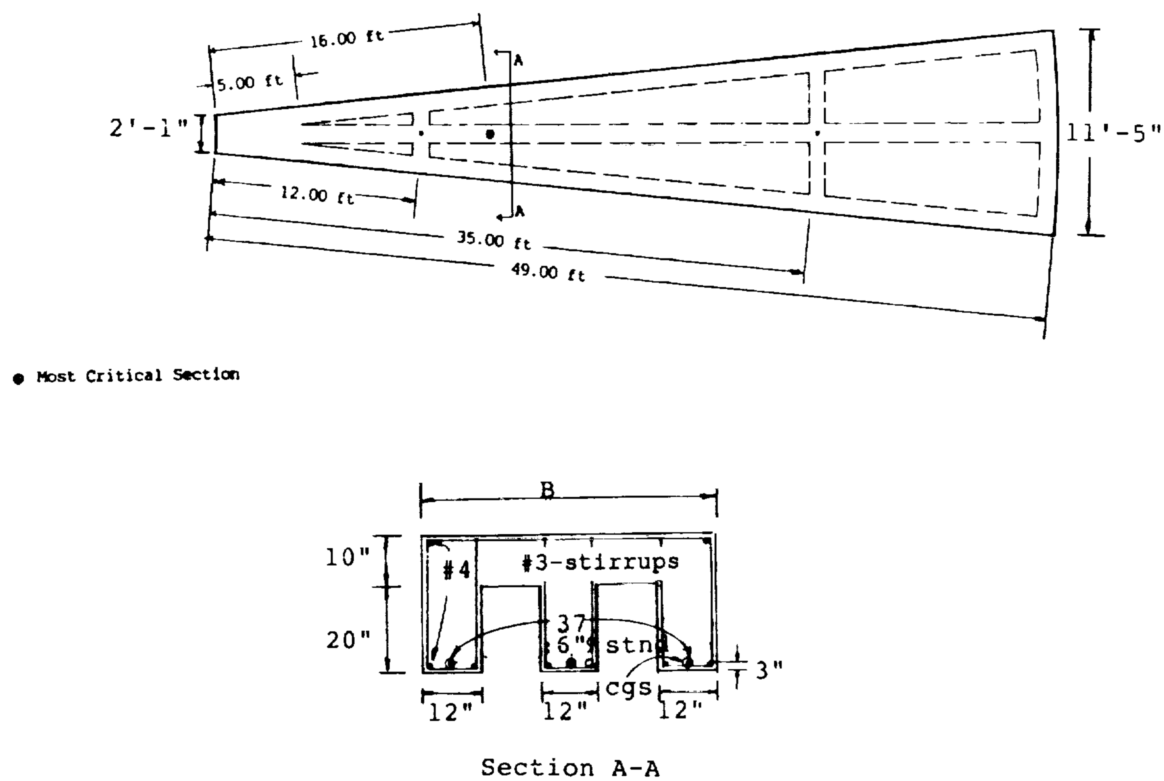

Fig. 7. A typical ground-floor slab.

column. The column must be capable of resisting the sum of the 1220-kip force and the projected live and dead loads that it may have to support. A 24-in circular column was selected from tables on pages 4-40 of the *CRSI Handbook* (1984). The designed tension-compression column is shown in Fig. 8.

Exterior Columns

There are 33 exterior columns on each floor of the three-story structure to laterally support the curved panels. In the vertical direction, three columns are aligned to form a 72-ft support for the roof slab and ground floor slab. These three columns are secured in position by prestressing vertical tendons that run through the columns and horizontal tendons that run in between the floor and ceiling slabs (see Fig. 9).

Columns for an ordinary structure on Earth act as compression members. For a pressurized structure on the Moon, the columns behave like a tension member or a beam. The design of the exterior columns followed the beam design procedures.

How columns are tied together in a multistory structure influences the serviceability of the lunar base. A connection mechanism, in which the lower end of an upper column in a semispheric shape sits on the dish surface of the top end of the lower column to form a roller-bearing joint, allows the jointed columns to rotate against each other and nullifies the bending moments at joints.

The prestressing tendons designed to balance air pressure in each column extend about 1 ft beyond the joint and are anchored to the adjacent columns. This arrangement will add safety to the column connections either in normal service or in an air-leak condition. The exterior columns are reinforced with rebars to prevent concrete from cracking due to stress reversals in an air-leak scenario. Section A-A of Fig. 9 gives details on reinforcing bars.

Core Structure

The inner core structure is a self-contained unit capable of withstanding 1-atm pressure. Structurally speaking, the core structure serves as an interior support to slabs so as to reduce slab span. The core structure will be fabricated using ring and plate elements with a weight limit of 10 t.

Based on the structural function, two types of rings were designed: the shear transfer rings and the compression rings. The shear transfer rings are elements that have corbels to support the slabs. All rings have a 20-ft inside diameter and thicknesses of 2 ft and 1 ft for shear transfer and compression rings respectively. All rings except the shear transfer ring at the top are 4 ft in height. Again, elastomeric pads will be used to ensure airtightness at all joints when the core structure is prestressed.

Both pressurized and depressurized conditions were considered in the design. Under the pressurized condition the circumferential stresses in the wall become the dominant design factor. The hoop force per foot of the ring subjected to 3600 psf internal air pressure is 36 kips. To balance this hoop force only one 0.6-in-diameter seven-wire strand is needed. Figure 10 shows the suggested hoop tendons. A complete circular tendon will be made from three segmental tendons, each having a subtended angle of 120°. Tendons that overlap in brackets ensure the continuity of hoop forces.

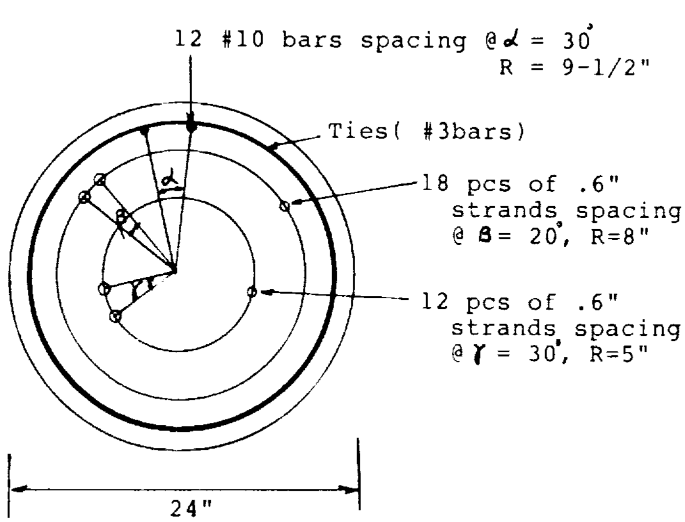

Fig. 8. Cross section of a tension-compression column.

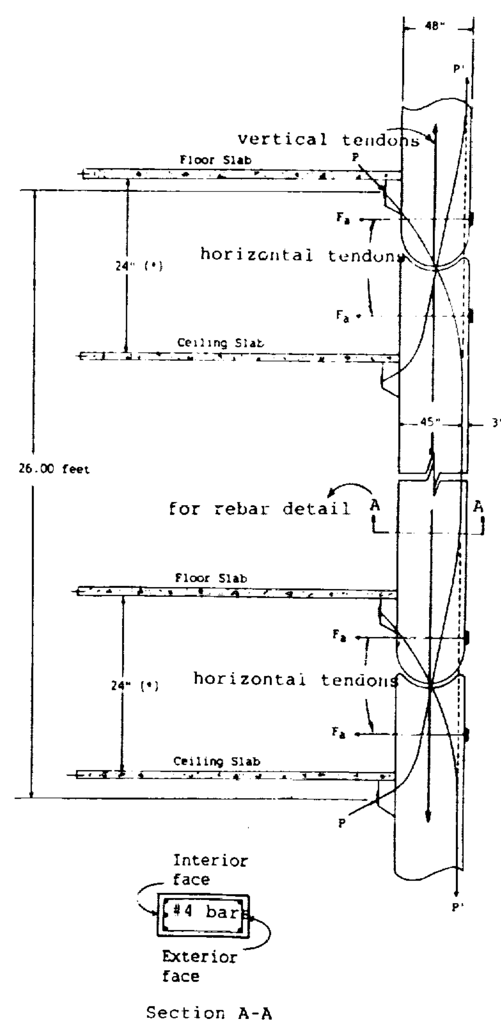

Fig. 9. Exterior columns and connections. Note: (•) - 24" space between the floor and ceiling slabs is to provide room for horizontal tendons.

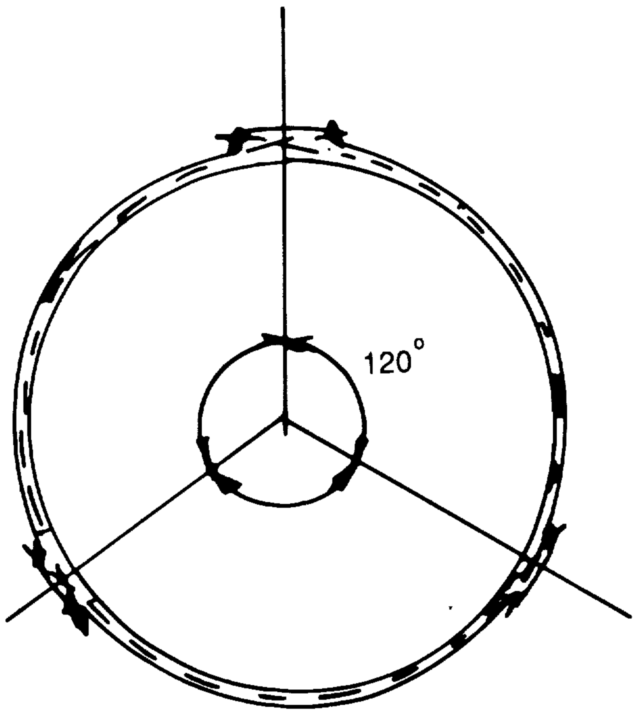

Fig. 10. Hoop tendons.

The vertical strands in the rings that hold the core elements together were determined from factored loads including air pressure, live loads, and dead loads acting on the cover plates. The calculated results show that 38 strands of 0.6-in diameter are needed to hold the cover plates and rings in position to contain the anticipated 1-atm air pressure.

Circular cover plates were designed as a two-way slab simply supported along its perimeter. The plate theory (*Timoshenko and Woinowsky-Krieger,* 1959) was used in calculating the maximum bending moment at the plate center. The two-dimensional balancing method was used to determine the needed strands. It was found that a 12-in-thick circular concrete plate with six 0.6-in-diameter strands for every 11.5-in strip will satisfy the design. The circular plates will be reinforced with conventional rebars to resist bending moments during construction.

ESTIMATE OF MATERIALS

In brief, the designed infrastructure weighs approximately 8500 t on Earth (1400 t on the Moon), of which concrete is about 8000 t, strands 110 t, and reinforcing bars 250 t. To make 8000 t of concrete needs approximately 1000 t of cement and 330 t of water, assuming free moisture in concrete will be captured for reuse. All these materials except hydrogen can be obtained from the Moon. The amount of hydrogen to be transported from Earth is 36 t.

FLOATING FOUNDATION

As reported, the depth of the lunar regolith varies from 0.5 m to 40 m in the mare and highland regions of the Moon. Laboratory studies (*Williams and Jadwick,* 1980) on returned samples show an average bulk density of about 1.66 g/cm^3, porosity of around 45%, cohesion strength of 0.1 to 1.0 KN/m^2, and absolute dryness. Based on astronaut's footprints, applying a pressure of 10 KN/m^2 would normally settle the soil approximately 1 cm. These mechanical properties of the soil make conventional compaction, piling, and mat footing impractical for construction on the Moon.

One of the important requirements for the lunar base construction is to provide airtightness and structural integrity. Therefore, the tolerance for differential settlements of the foundation must be extremely small. To achieve this objective, an innovative concept of a floating foundation for lunar construction was developed to minimize possible relative movements between structural members of the superstructure due to foundation settlements. The proposed foundation structure will be made with precast concrete members. A rigidly assembled substructure that could have rigid body rotations and translations as the lunar soil beneath it yields can support the superstructure like a ship carrying stacks of cargo on the high sea. This design could practically eliminate differential movements among elements of the superstructure.

A truss structure made with precast concrete members is not common. The main reason is that when a truss is subjected to loads, some of its members may develop tensile stresses that could cause cracks and damage to concrete. However, if one can develop a truss, subjected to loads at the top and soil reactions at the bottom, to produce compressive stresses throughout its members, the feasibility of using concrete for truss construction can be realized. Through many trial analyses, one such truss was found and is shown in Fig. 11.

The dish-shaped foundation will be fabricated from four or six trusses. The assemblage of trusses and methods for connection require careful studies. The design of the proposed foundation is left for future study.

CONCLUSION

Precast, prestressed concrete has many material and structural merits for the proposed lunar base construction. The attractiveness of concrete structures lies in the fact that most materials except hydrogen for making concrete, steel, and water are available on the Moon. To build the proposed infrastructure 120 ft in diameter and 72 ft in height requires about 8000 t of concrete, 110 t of prestressing steel, and 250 t of rebars. To make the 8000 t of concrete requires about 36 t of hydrogen. Hydrogen, the lightest element known to man, is the only material that needs to be transported from Earth.

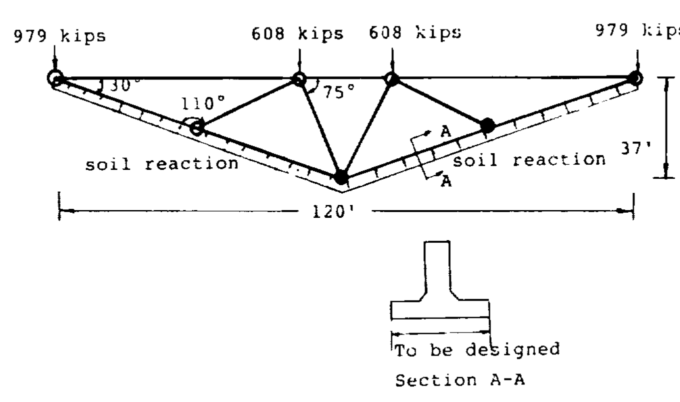

Fig. 11. A proposed floating foundation.

The designed prestressed concrete structure provides 33,000 sq ft of work area, large enough for industrial productions including the assemblage of rockets in a horizontal position. The designed infrastructure is capable of supporting a dish-shaped solar energy collector with a diameter of 120 ft on its rooftop. Photovoltaic solar panels as high as the lunar base structure can be erected along the perimeter of the base. These solar energy installations serve as radiation shields as well as energy supply sources.

It is concluded that to build concrete structures to safely contain 1-atm air pressure on the Moon is technically feasible. This investigation is only a preliminary design; more research and design studies are needed to complete a final design.

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REFERENCES

Building Code Requirements for Reinforced Concrete (1983) ACI Standard 318-83, Detroit, Michigan. 155 pp.

CRSI Handbook (1984) Sixth edition. Concrete Reinforcing Steel Institute, Schaumberg, Illinois. 1024 pp.

Lin T. D. (1967) *Design of Prestressed Concrete Structures,* 2nd edition. Wiley, New York.

Lin T. D. (1987) Concrete for lunar base construction. *Concr. Int., 9.*

Lin T. D., Love H., and Stark D. C. (1987) Physical properties of concrete made with Apollo 16 lunar soil. *Proc. AIAA Conf.* American Institute of Aeronautics and Astronautics, New York.

Lin T. D., Senseney J. A., Arp L. D., and Lindbergh C. (1988) Concrete lunar base investigation. *J. Aerosp. Eng., 3.*

The National Building Code (1985) Engineering and Safety Service, New York.

Post-Tensioning Manual (1981) Third edition. Post-Tensioning Institute, Phoenix. 323 pp.

Timoshenko S. P. and Woinowsky-Krieger S. (1959) *Theory of Plates and Shells,* 2nd edition. McGraw-Hill, New York.

Williams R. J. and Jadwick J. J. (1980) *Handbook of Lunar Materials.* NASA RP-1057.

VERTICAL REGOLITH SHIELD WALL CONSTRUCTION FOR LUNAR BASE APPLICATIONS

N93-17446

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Lunar bases located on the lunar surface will require permanent protection from radiation and launch ejecta. This paper outlines a method of providing physical protection using lunar regolith that is constructed in situ as a modular vertical wall using specially devised methods of containment and construction. Deployable compartments, reinforced with corner struts, are elevated and filled by a moving gantry. The compartments interlock to form a stable wall. Different wall heights, thicknesses, and plan configurations are achieved by varying the geometry of the individual compartments, which are made from woven carbon fibers. Conventional terrestrial structural engineering techniques can be modified and used to establish the structural integrity and performance of the wall assembly.

PURPOSE

An established lunar base located on the lunar surface will require permanent protection from ionizing radiation sources and from engine blast debris or ejecta. Radiation, which includes galactic cosmic radiation and solar cosmic radiation (*NASA*, 1989), is particularly lethal in the form of solar flare radiation produced during periods of intense cyclic activity. The generation of debris and ejecta trajectories caused by vehicle engine blast during launch and landing operations is significant and accommodation and facilities must be shielded accordingly (*Phillips et al.*, 1988). Though debris and ejecta are to some extent directional, radiation is isotropic, indicating that full enclosure of lunar base accommodation and facilities will be necessary for a permanently manned lunar base. The most economical and effective method will be based on mass-shielding using local lunar regolith as the shielding material.

This paper describes continuing studies of autonomous superstructure systems designed to provide the required complete proximity shielding of accommodation facilities located on the surface. A method of providing horizontal shielding above a lunar base complex has previously been described (*Kaplicky and Nixon*, 1985), and this paper outlines an approach toward the configuration of shielding as a vertical wall. It is anticipated that both systems can be integrated and reconfigured as a complete enclosure, given the availability of programmatic data on the architecture of the lunar base.

CONCEPT

Free lunar regolith can be aggregated and constructed *in situ* as a modular vertical wall to surround the lunar base, based on knowledge of the availability of loosely compacted regolith in the vicinity of the lunar surface (*Taylor*, 1982). Loose regolith is deposited into individually deployed columns of compartments that are interconnected and reinforced by lightweight vertical struts. The wall is constructed in a progressive linear sequence from the ground up using a construction method that employs an independent mobile gantry using interlocking struts as reinforcement elements. The wall geometry is tailored to the required shape and size for protection (nominally 2 m thick by 7 m high for the purposes of preliminary studies). Key objectives for the concept include (1) maximum simplification and standardization of required fabricated structural elements, (2) maximum control and verification of construction integrity during assembly tasks and sequence, (3) minimum volume and weight of structural elements/components to be transported from Earth, and (4) minimum crew extravehicular activity (EVA) task complexity, physical effort, and construction time on the lunar surface.

ASSEMBLY

Compartment stacks are transported to the lunar surface in a compact and stowed condition in which the stacks are sandwiched between lightweight pallets. A possible construction for the pallets is a 2-cm-thick alloy waffle panel. Each pallet subsequently acts to help spread the load of the regolith evenly over the prepared regolith ground surface.

In Fig. 1, the stowed stack pallets are retrieved after landing and transported to the construction location using lunar surface vehicles. Field assembly of the shield wall uses a mobile gantry. In Fig. 2, a deployable and independently mobile gantry frame is assembled. Gantry frame design and construction is based on technology derived from the space station beam structure development and already successfully tested in orbit (STS 61B).

In Fig. 3, shield wall stacks are elevated in a bay-by-bay sequence. The folded stack pallets are first placed in the required positions at ground level. The operator moves the gantry over

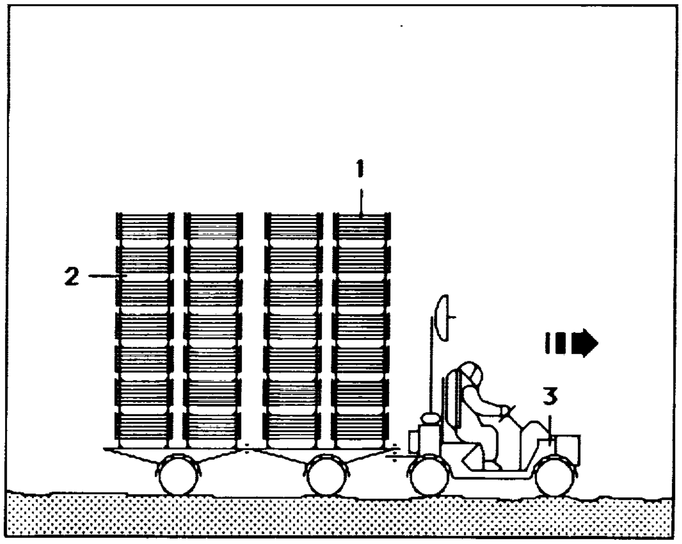

Fig. 1. Retrieval of (1) stowed wall stacks and (2) pallets after landing and transportation to the construction location using (3) lunar surface vehicles.

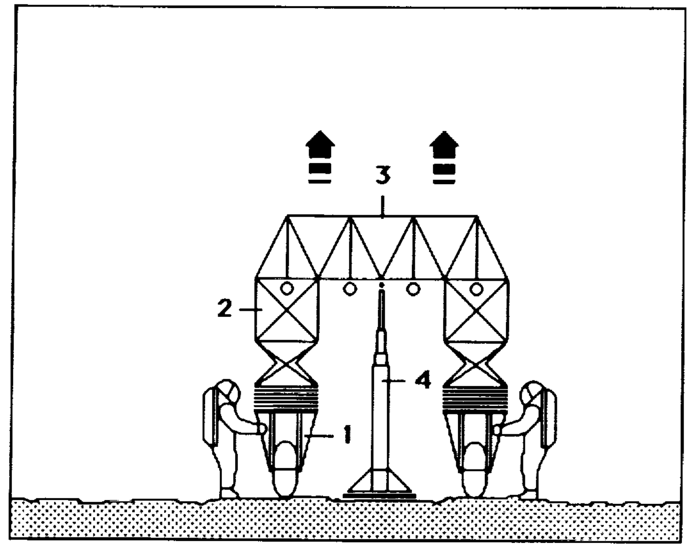

Fig. 2. A deployable and independently mobile gantry frame is assembled: (1) gantry undercarriage, (2) extending framework, (3) crossbar, and (4) lifting mechanism.

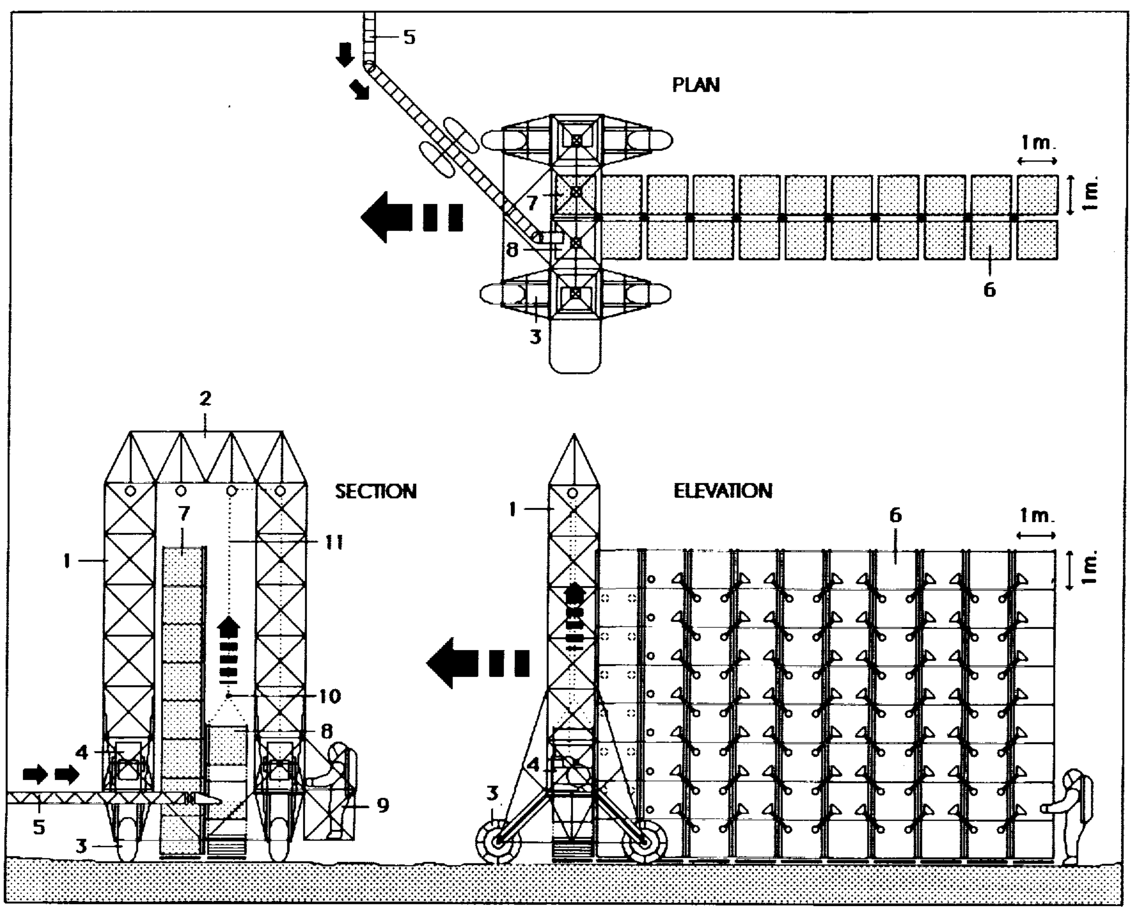

Fig. 3. Shield-wall stacks are elevated in a bay-by-bay sequence: (1) gantry towers, (2) gantry crossbar, (3) gantry undercarriage, (4) gantry drive mechanism, (5) mobile regolith conveyor, (6) completed wall section, (7) completed compartment stack, (8) stack deployment in progress, (9) assembly EVA operator, (10) hook attachment, and (11) hoisting cable.

each stack in turn and attaches a lifting cable and hook to loops attached to the top compartment of each stack. An electric motor at the gantry base winds in the cable to raise the folded compartments vertically. The act of raising the compartments causes them to unfold from the pallet in a controlled deployment by means of integral cables, as shown in Fig. 4. As each compartment unfolds, it is filled with regolith from an independent mobile conveyor and closed and secured with an integral flap. The vertical struts are interlocked with matching struts in the adjacent stack, which is already in place. The compartment is then hoisted up. As each filled compartment is hoisted, vertical struts attached to the corners interlock with similar struts attached to the preceding compartment suspended above. This procedure is repeated until all compartments in a single stack are filled, locked together, and suspended from the gantry.

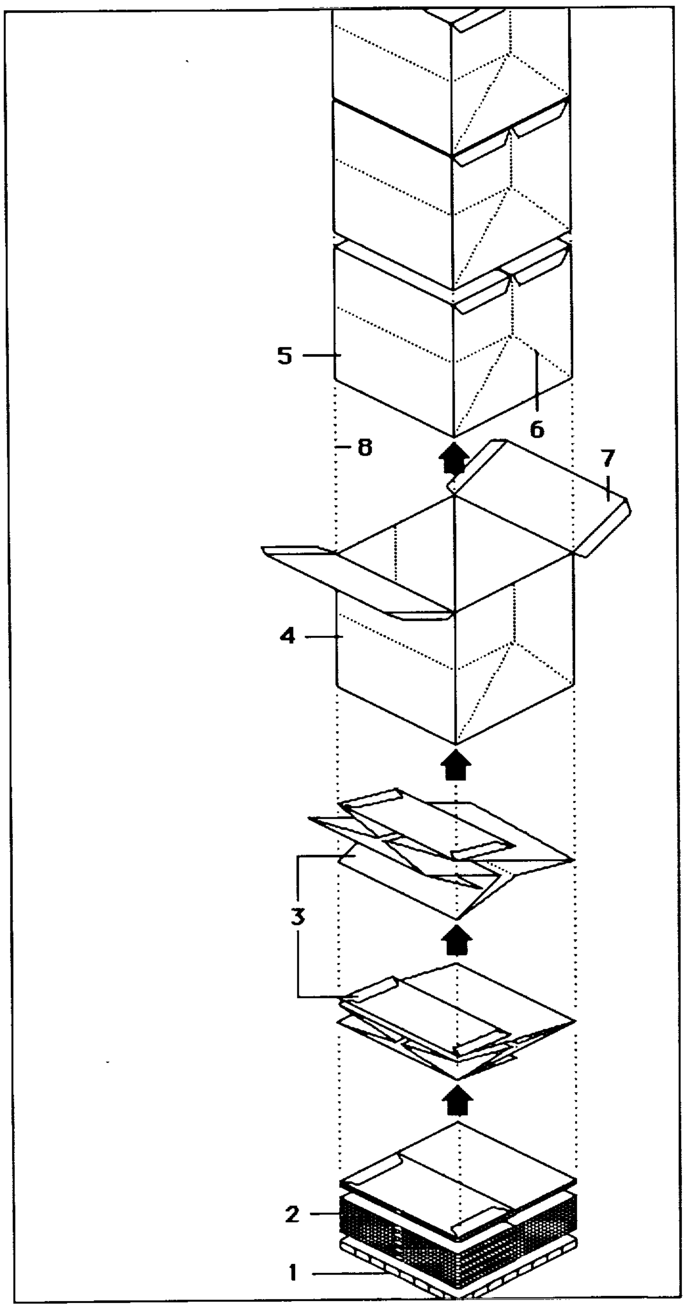

Fig. 4. Raising the compartments causes them to unfold from the pallet in a controlled deployment by means of integral cables, as shown here: (1) stack pallet, (2) compartments stowed, (3) compartments unfolded, (4) compartments deployed, (5) compartments secured, (6) fold lines, (7) flaps, and (8) cable lines.

The entire stack is then lowered to the surface where the filled compartments settle into compression. The hook and lifting cable are released from the completed stack and the gantry moves on to the next stack. Regolith loads transfer vertically downward from compartment to compartment and onto the pallet at ground level where they are distributed onto the lunar surface. The overall form of each stack will experience controlled deformation and settlement but will remain contained and stabilized by the interlocking reinforcement struts. Finally, compartment faces are strapped together to provide longitudinal stability using spring-activated microfastener (hook-and-eye) pads.

Once complete, the shield-wall system is essentially inert and will require no maintenance. Periodic inspections of compartment membrane integrity will be required to check for damage due to meteor or other debris impact. A method of field repair of damaged compartments will be required to rectify such damage.

STRUCTURE

The aim is to design a wall that is stable during and after construction and that can resist nominal horizontal loads such as the accidental impact of a crewmember on EVA. It is assumed, for example, that a substantial meteor hit will involve corresponding repair or rebuilding of the wall.

It is assumed that the loose regolith is of the order of 20 cm thick. This will first be cleared from the area where the wall is to stand so that it will be founded on the more dense rock below. If the surface of the rock is not level, it will be built up with a thin layer of regolith to provide a flat base. It is proposed that the structural elements are designed in accordance with the Uniform Building Code (*International Conference of Building Officials,* 1985) as far as allowable stresses, factors of safety, and resistance against buckling are concerned.

Preliminary studies are exploring the feasibility of using carbon fiber cloth bags that are stiffened with alloy struts to form cubic compartments for the regolith, as shown in Fig. 5. The compartments are assembled into vertical stacks that form the wall. Mechanical connection between the compartments is provided at the corners where the struts, which run vertically along two corners of one compartment, interlock with similar struts above. Battens are also used to stiffen the compartments. Four integral battens are incorporated along the edges of the top face of each compartment to keep the cubic geometry in shape while it is filled with regolith. Battens are also incorporated down compartment corners. Lightweight pallets keep the bottom of the compartment stacks flat. Flaps at the top face of each compartment are folded down once filling with regolith is complete and are held in place with microfastener pads. The compartment stacks behave primarily in compression. Some shear stiffness is provided by the connections between the struts and also by the in-plane shear stiffness of the fabric, which can be improved by adopting a triaxial fabric weave and by introducing a regular pattern of microfastener pads on the wall outer elevations to "stitch" the compartment stacks together after deployment. These

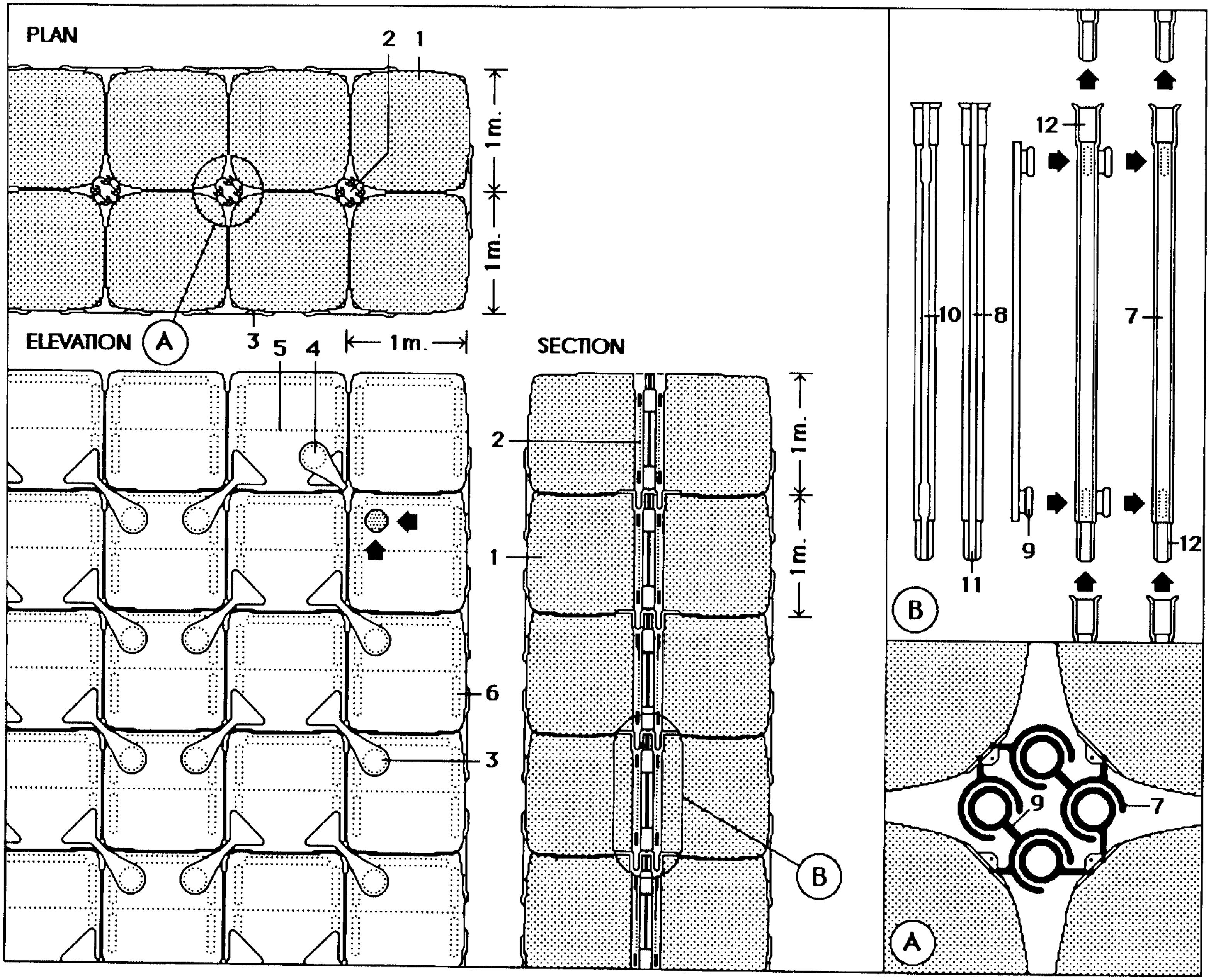

Fig. 5. Preliminary studies are exploring the feasibility of using cubic compartments for the regolith as shown here: (1) filled compartments, (2) interlocking strut assembly, (3) microfastener pad deployed, (4) microfastener pad stowed, (5) compartment bag fold lines, (6) integral battens, (7) typical strut section, (8) typical strut elevation, (9) interlocking lugs, (10) strut with track slots, (11) lug tracks, and (12) strut interlocking ends.

measures are required to improve the stability of the wall. Scaled testing is required to establish an equivalent Young's Modulus for the assembly, so that a factor of safety against instability can be established.

As well as compression, the vertical alloy struts and their connectors are required to be designed to carry the weight of six compartments in tension. This loadcase occurs during construction when six filled compartments are suspended by the gantry during the filling of the seventh, lowest compartment. The carbon fiber fabric will be tested to take the pressure from 7 m of regolith.

INITIAL CONCLUSIONS

The studies demonstrate that (1) vertical shield-walls constructed primarily of raw regolith contained by minimal "delivered" structural elements are a feasible proposition, (2) autonomous "stand-off" shielding eliminates the need for excavation and does not impact systems configurations and operations, and (3) terrestrial civil/structural engineering methodology and analysis procedures adjusted for lunar conditions are appropriate for selective lunar engineering applications.

FURTHER STUDIES

Studies are needed in the following areas to further evaluate the feasibility of the concept: (1) testing of full-scale model with densities scaled down for feasibility of construction, (2) strength and serviceability testing of carbon fiber fabric options, (3) preliminary detail design of gantry and strut system, (4) assessment of safe bearing pressures on the lunar surface, and (5) assessment of the bulk compaction and cohesion of the regolith.

REFERENCES

International Conference of Building Officials (1985) *Uniform Building Code*, pp. 456-467. Whittier, California.

Kaplicky J. and Nixon D. (1985) A surface-assembled superstructure envelope system to support regolith mass-shielding for an initial-operational-capability lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 375-380. Lunar and Planetary Institute, Houston.

NASA (National Aeronautics and Space Administration) (1989) *Man-Systems Integration Standards*. Vol. 1, Rev. A, pp. 5-77 to 5-117. NASA STD-3000.

Phillips P. G., Simonds C. H., and Stump W. R. (1988) Lunar base launch and landing facilities (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 194. Lunar and Planetary Institute, Houston.

Taylor S. R. (1982) *Planetary Science: A Lunar Perspective*, pp. 115-123. Lunar and Planetary Institute, Houston.

EVOLVING CONCEPTS OF LUNAR ARCHITECTURE: THE POTENTIAL OF SUBSELENE DEVELOPMENT

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In view of the superior environmental and operational conditions that are thought to exist in lava tubes, popular visions of permanent settlements built upon the lunar surface may prove to be entirely romantic. The factors that will ultimately come together to determine the design of a lunar base are complex and interrelated, and they call for a radical architectural solution. Whether lunar surface-deployed superstructures can answer these issues is called into question. One particularly troublesome concern in any lunar base design is the need for vast amounts of space, and the ability of man-made structures to provide such volumes in a reliable pressurized habitat is doubtful. An examination of several key environmental design issues suggests that the alternative mode of subselene development may offer the best opportunity for an enduring and humane settlement.

INTRODUCTION

It has been a very long time since the art and science of architecture has been called upon to contribute fundamentally to the transformation of human civilization. Nevertheless, we can see that humankind's ability to expand civilization to another planet will certainly depend upon our success in contriving a very sophisticated built environment—an architecture that is truly appropriate for the Moon. In seeking this goal, it is conceivable that we may be required to dispense with our terrestrial tradition of "erecting" buildings. Ironically, it may turn out that the profession that contributed to the advancement of civilization by giving humankind an alternative to the cave may call us back to that environment.

The definition of architecture here must be stretched a bit beyond the Vitruvian conception of rigid structure, utility, and aesthetics, for these elements hardly begin to address the complexity of creating a fully integrated biospheric medium. When we consider the subject of building a place for man on the Moon, we must take a radical approach, for there are no applicable earthly precedents to guide us. We must think holistically, in terms of integrated systems, for the problems of lunar habitation are interconnected, and they cannot be considered in isolation. Certainly, we cannot think of architecture merely in terms of structure and function. Given the nature of this extraordinary endeavor, it can be posited that the architect, in the truest definition of his profession, will play a central and critical role in determining the real potential of lunar settlement.

A review of the numerous proposals for lunar base construction and habitation reveals a variety of themes. Looking critically at these, we find many innovative proposals that tend to suffer from their concentration on a very limited set of considerations. There has also been a tendency to rely on preconception, a tendency to extrapolate methodologies developed in previous space missions to the realm of the lunar base. Too often, highly logical designs are nevertheless weakened by a reluctance to consider the more intuitive notions of a designer's mind—a shame at this stage of the discussion. There has been a noticeable deficiency in designs that look beyond the early outpost phases of basing, at the question of how a lunar base may evolve—and at how anticipation of this evolution may guide early base planning. A continuous thread linking most of these proposals is that they have been proposed in the absence of a clearly defined program; however ingenious, they are solutions in search of a problem. To solve the problem of radiation shielding, or of thermal stress, or of atmosphere containment—to solve one problem, or another—is not enough. There has been a lack of comprehensiveness in the consideration of architectural issues, and this is because no one has yet been able to propose a workable architectural program that relates all the various factors that must form the basis of any lunar base design. Until this is accomplished, it will not be possible to evaluate fairly any specific proposal.

This paper is aimed at contributing to the discussion of lunar development by offering to the reader some insight into the range of architectural considerations that must shape this program, and to suggest how differing modes of architectural development are able to respond to a spectrum of factors. In so doing we will attempt to define and formally distinguish between two very different modes of lunar basing, these being the categories of surface-deployed superstructures and subselene adaptational environments. We believe that the alternative mode of subselene development, i.e., the exploitation of natural lunar caverns, may

very well yield novel conceptions of the manner in which a lunar base may evolve, and offer a reasonable means of producing a humane lunar settlement.

A Sampling of Critical Architectural Issues

The list of factors that will influence lunar base design is prodigious and spans virtually all fields of human interest. The architecture that we ultimately build, viewed at any stage of evolution, will certainly result as a compromise product reflecting collaboration between many centers of expertise. Matters that seem to go well beyond the purview of architectonic practice will become critical in lunar base design. For many of us, the ability to resist the convention of pursuing narrowly defined technological questions will be an important first compromise.

Another crucial first step will occur when we come to see the architecture of lunar settlement, not in terms of a translation of terratectonic principles, nor in terms of modified off-the-shelf technologies, but rather as a highly specific product of invention. A fitting lunar architecture will require a radical approach, a necessity forced upon us by the distinction of this new planet. We will need to purposefully reconsider the ways we have been conditioned to build on the Earth, and we must be prepared to dispel all preconceptions; we must become preoccupied with novelty. The great promise of this, of course, is not merely implied for architecture on the Moon, but for the quantum improvement of architecture in general.

It should be noted that the most perplexing concerns of lunar base design may relate less to the more widely discussed problems of fractional gravity, radiation flux, and vacuum, and more on the fathomless issues of human behavior and interaction.

With these qualifications in mind, a brief review of several of the more critical architectural considerations is offered.

Lunar Gravity

Of course, one of the most prominent and alien features of the Moon is its fractional gravity, and this will affect the architecture in various ways.

Clearly, structural design will reflect effectively increased load-bearing capacities; however, this must be taken in the context of several interacting factors. For instance, if regolith-mass shielding is to be employed, any inherent load-bearing advantages may be canceled. Although gravitational force will always be a significant factor, even in $1/6\,g$, other factors may govern structural design determinations. Principally, we are thinking about pneumatic forces due to atmosphere containment. Internal air pressure is a variable, and has to be considered a dynamic force. Extreme thermal cycling may force further complication of the structure, thereby reducing the efficiency of spanning systems. The performance of indigenously manufactured structural materials may be compromised by extraordinary design safety factors. Function and safety factors may work to counter any opportunity for material efficiency in spanning members when system redundancy and compartmentalization strategies overrule.

Another effect of reduced gravity concerns anthropometry, space planning, and the dimension of space within a base. The dynamic human dimensional relationship with the built environment is gravity dependent. Intuitive expectations of lunar base spatial requirements can only be modeled hypothetically, and cannot be easily translated from the terrestrial condition. The effect of this problem will contribute to form determination. Also, it seems likely that continued research into this question will result in a modification of present estimations of spatial economy and efficacy.

A third important effect of this issue concerns the health of humans and other animals and plants, and this relates to the largely unknown and potentially deleterious effects of living in a substantially reduced gravitational environment. *Diamandis* (1988) addresses this and points out reasons to doubt that lunar gravity will provide sufficient physiologic stresses over the long term to prevent the same deconditioning that is seen in zero gravity. (Extended stays in zero gravity have led to immunosuppression, muscular atrophy, osteoporosis, cardiovascular deconditioning, and body fluid/metabolite shifts; there is also the strong suggestion that embryogenesis and early development will be adversely affected.) Potentially, these physiologic reactions threaten our ability to adapt permanently to the Moon, and jeopardize as well the option of revisiting Earth. The built environment must be able to accommodate these concerns in several ways. First, a primary method of mitigating physiologic stress will almost certainly depend upon physical exercise, and so the architecture might be designed so as to require the inhabitants to walk long distances between elements of the base. Another means toward the prevention of these physiologic disorders involves the inclusion of some mechanism for providing artificial gravity, as suggested by Diamandis. In both cases, the architecture would need to be capable of providing the requisite spatial volume and three-dimensional sophistication implied by these devices.

Radiation Shielding

It is a well understood fact that the enclosing envelope of any lunar base must be capable of shielding the inhabitants from the intense ionizing radiation that strikes the lunar surface. In the case of surface constructions and modular habitats, it is generally estimated that between 2 and 3.5 m of loosely piled regolith will be required to provide sufficient protection (*Silberberg et al.*, 1985). Considerations of habitat form and exposure are aspects of design that are directly affected by this problem (see *Land*, 1985). Other matters that are called into question include structural complications due to the radiation-shield load; preferences for certain shielding materials (considering the generation of secondary neutrons within the shielding material by cosmic rays, as well as the variable absorptive efficiencies of candidate shield materials); the practicality of fenestration; access to the exterior hull for inspection and repair (see *Kaplicky and Nixon*, 1985); paradoxical limitations on solar access; and the practical considerations of maintenance. The designers of a lunar base are therefore obligated to consider very carefully the ways in which this necessary element will work to shape base architecture.

Atmosphere Containment

The form of a lunar base will be determined by a wide range of factors, but a common denominator in any formula for resolving base morphology will be the restrictions imposed by the physics of atmosphere containment. Without the perfect and reliable confinement of an atmosphere, no lunar base is possible. Having said this, it must also be noted that atmosphere containment cannot be held in isolation as the exclusive determinant of form (as has been a theme in many lunar base proposals). If pressure-

vessel physics were to dominate our thinking, we would be limited to the utilization of spheroids and cylinders, and with respect to the many other requirements that must contribute to the definition of base architecture, these forms are fundamentally problematic.

We should realize that the very knowledge of environmental integrity and dependability on the part of the inhabitants will likely become a key to our adaptive ability, and so there is a behavioral component to atmosphere containment. Therefore, while the structure of a lunar base must be designed for fail-safe reliability, there should also be a sufficient level of architectural sophistication to express this strength to the inhabitants.

The enclosure system should be able to withstand accidental and intentional decompression of the structure, and it may be unwise to rely on structural systems that depend upon internal air pressure for support (since their integrity depends upon the integrity of the atmosphere). It is important that any hull-type structure remain accessible for inspection and repair. Also, once established, a lunar base will likely be in a virtually continuous growth mode, so it is important that the structural system be devised so as not to interfere with base expansion and revision.

Very importantly, as a breathable atmosphere represents an absolutely vital resource that, in theory, could become the subject of political influence or the target of sabotage, appropriate safeguards must be considered and eventually integrated into the architecture. (Similar vulnerabilities will exist for water, food, energy, and other vital resources as well.)

Extreme Thermal Stress

Surface temperatures over the lunar diurnal cycle vary over a range of 500°F (260°C). Structural elements that are subject to exposure to this extreme thermal variation, particularly exposed or uninsulated atmosphere-containing superstructures, must be highly elastic in their design. Material fatigue due to thermal cycling may be a problem and could limit the effectiveness of certain materials. Fully sheltered superstructures, with thermal differentials of perhaps 300°F (149°C) will be subject to lesser but still significant extremes. This will constrain the scale of exposed superstructures, as well as the range of geometries that might be available. It will require the use of proven, high-strength materials, which further implies a very high level of architectonic sophistication, construction difficulty, reliance on high-precision components, and the need for redundancy in atmosphere containment systems. If material fatigue is a significant problem, structure lifetime will be adversely affected.

Environmental Ruggedness

Many recent proposals suggest derivative space-station technology (habitat modules) for use as lunar habitats, others suggest pneumatically supported fabric structures, and still others feature large thin-walled aluminum domes. Considering the nature of activities that are postulated for the Moon (mining, industrial manufacturing, chemical production, transportation node, etc.), and considering that this expansion-oriented permanent settlement will be inhabited, not by a highly trained crew, but by a very mixed population of individuals, these proposals seem inadequately rugged. Accidents, abuse, and misuse are certainties within any human-inhabited environment and must be considered in the formulation of any architectural system. The important and early need for a rugged, abusable, "kickable" environment should not remain understated.

Meteoritic Impact Susceptibility

Recent theses on lunar base design have usually considered the effects of micrometeoroidal impacts on structures and equipment (*Johnson and Leonard,* 1985, and others). Certainly, the issue of micrometeoroidal impacts is important in the design of virtually all types of space structures, and it will be a very important concern in lunar base design. The fact that lunar base design must reflect many of the same problems that have typically concerned spacecraft designers is underscored by recent studies that have shown that the lunar-environment dust flux is substantially denser (as much as 10^2) than interplanetary models (*Grün et al.,* 1984). In particular, we must be concerned with the long-term performance of exposed materials, as well as the potential for puncture impacts.

Lunar planners must have special concern, however, for the far more insidious larger meteoritic bodies, for they pose a potentially catastrophic threat to permanent lunar habitats. Macrometeorite impacts do indeed occur on the Moon with sufficient frequency that they pose a real threat to long-term lunar habitation and they must be considered in the planning of any lunar base (Zook, personal communication, 1988). We are concerned here, not with dust, but with multicentimeter metallic projectiles moving at extremely high velocities. We suggest that it is overly simplistic to dismiss this matter on the basis of a statistical supposition. More realistic would be the adoption of a conservative engineering philosophy, where an evaluation of worst-case scenarios would demand that structural designs be devised on the basis of the assumed certainty of various types of collisions and near-collisions. Considering the indeterminate lifetime of lunar base structures, and given the need for the assurance that the inhabitants will demand, this seems a most reasonable approach.

Political Considerations

The political issues that will have an impact on lunar settlement design are perhaps the most difficult to assess and may be the most critical concerns for lunar base planners.

The scope of concerns here is very broad, spanning the intricacies of international relations, nation building, national security, economics, monetary standards, political theory, law, common heritage, and the definition of property on national and individual scales. All these considerations will interactively affect the architecture of lunar settlement. For a broader discussion of the nature of these matters in the context of space and lunar development, the reader is referred to a number of articles, including *Joyner and Schmitt* (1985), *Finney* (1985), *Dula* (1988), *Gabrynowicz* (1991), and *Robinson and White* (1986).

There are a number of political variables that stand out as being determinative of lunar base architecture. First, there is the realization that current international treaty casts doubt on national prerogatives with regard to the construction and property definition of a lunar base. Then there is the question of the predominating politico-economic system philosophy of the nation or nations involved. The governing system, planning philosophy, functional characteristics, and the rate and direction of future growth for the base will all be guided by this issue. Another pivotal planning consideration here is the question of property definition and individual liberty—by which political model will lunar settlement be guided? A related question concerns vital resource authority and distribution, and the problem of delegating authority for the maintenance of essential life-supporting systems (including the architecture itself). Ultimately, redundancy (or

decentralization) in vital resource storage and distribution systems may come to parallel the importance of structural system redundancy, but for the purpose of making political control more difficult.

Another concern that should not be overlooked is the ability of architectural systems to respond over time to changing needs and functional requirements, especially as they may be directed by political considerations. Vicissitudes in national and international policy may require unforeseeable changes and constant modification of base facilities. Evolution toward settlement autarky will certainly require a transformable architectural system. Basically, the architecture can either contribute to successful polity, or hinder it, depending on the degree of responsiveness to these changing needs.

There is a potential in the holistic view of architectural planning for providing mechanisms that work to protect pluralistic systems and the rights of the individual. Conversely, a faulty design can be an instrument of control. While these concerns may not be obvious in the early outpost phases of lunar basing, they will surely become mandatory for greater settlements. What must be remembered is that the Moon forces a duty on the architecture for which there is no corresponding terrestrial analog, and that is the obligation of providing essential life-support. In such a role, we can be sure that the architecture will be the subject of political influence.

Behavioral Issues

The interior environment of a lunar base presents myriad psychological and sociological design questions and complications, far too many to list here. It should be noted that although space-environment behavioral problems have been studied at great length at NASA and other agencies, much of this work has focused on considerations that relate to space vehicles, zero-gravity environments, and the social interrelationships of highly trained crew personnel. Much of this work has little or no meaning in a lunar setting, and new research efforts will be needed to properly equip base architects with meaningful insight. Let it suffice to say that the development of any baseline lunar base architectural program will remain incomplete without significant novel research in this area, and that many of the architectural proposals produced to date have originated in the absence of this critical information.

We would like to suggest several areas of behavioral research that will directly affect the architecture of a lunar base, and that require detailed investigation. They include the following:

Spatial volume requirements. To determine the human need for space in the totally confined environment of a lunar base. It is possible that this requirement will be highly determinative of planning strategies, and the need for copious internal volumes may force a reevaluation of current postulations of lunar base size.

Environmentally imposed psychological stress. To anticipate any deleterious psychological reactions or stresses that may result from living with the constant potential for environmental failure; to suggest architectural devices that may ameliorate these apprehensive stresses.

Environmental stimulation and diversity. To further assess the human need for environmental diversification; to suggest sources of environmental stimulation that might supplant missing terrestrial stimuli.

Individual spatial requirements, retreat space, and privacy. To evaluate the essential environmental requirements of the individual within the specific context of lunar settlement; and to do so in the context of such crossover concerns as property definition, political philosophy, and fractional gravity anthropometrics.

Earth-diurnal cycle emulation methodologies. To study methods of recreating various psychological and biological environmental cues based on terrestrial conditioning; to evaluate their effectiveness in the lunar setting; and to suggest possible architectural contributions. Key concerns here are environmental lighting and lighting controls.

Architectural semiotics. To consider evolving concepts of lunar base design that depend upon subliminal suggestion or semiotic message in order to bring about some desired effect. Such devices may be useful in the prevention or moderation of environmentally imposed stress, for example.

Spatial Volume

A misconception, we think, concerning the design of lunar bases, relates to the assumption that spatial volume within a lunar base will be a premium and highly economized amenity. This idea, expressed in so many proposals, seems to be an extension of precedent and practice, and may be due to the fact that, with all previous space missions, large spatial volumes have been achievable only rarely, and then only at great expense. This thinking may also be the product of presumptions about the economic and practical limits of large structures. Of course, a lunar base is essentially a static structure and, as such, it represents a novel mode of space development. While the economics of lunar development will be the subject of continuing study, we should probably take care to avoid any premature conclusions about the cost of large-scale development. In any case, the absolute need for copious internal volumes in a lunar base will inevitably present itself, regardless of economic expectations. It will simply be unfortunate if our lunar ambitions are needlessly restrained.

Simply put, we should expect the architecture of a continuously expanding lunar base to be able to accommodate the spatial needs, whatever they are, of the inhabitants. It should be anticipated that the open volumes of these spaces will be quite large. The need for spatial volume over the long term may be equal to the need for other vital elements of life support, and must be considered a design-driving issue. The need for transition from small-volume early outpost spaces, to large-volume greater settlements may present itself very early in base evolution, and this should be considered in any program evaluation. This is a matter that cannot be overlooked or subordinated.

SUPERSTRUCTURAL AND SUBSELENE MODES

As part of this report, we would like to formally distinguish between two fundamentally different ways of approaching the construction of a lunar base. The responsiveness of each type to critical design issues varies, so the distinction is important.

The category of lunar surface superstructures includes the great majority of lunar base proposals to date. Basically, any erect construction, whether assembled, inflated, or landed, situated on or near the lunar surface, fits this classification. Typically, superstructures rest on a prepared foundation (ideally one anchored to bedrock). Habitable superstructures must provide a structural envelope capable of the reliable containment of an atmosphere. In all cases, it is the structural system that must carry

the full range of loads, allowing multiple levels of redundancy and various factors of safety in their design.

Contrasted with this type is the category of subselene development, which involves the environmental adaptation of the lunar subsurface. Within this classification, structural and atmospheric loads may be carried directly by the surrounding rock mantle, with the greatly minimized need for a substantial and sophisticated superstructural enclosure. The direct exploitation of lunar lava tubes (natural caverns) may be considered a particular subtype of subselene development. The use of lava tubes as shelters for superstructural elements (but without closure and pressurization of the tube) can be considered as a hybrid mode of subselene development. A second subclassification might include excavated developments, where self-supporting voids (artificial caverns) are purposefully created. With subselene basing, we distinguish the lunar subsurface as being far more environmentally hospitable to development than is the surface and, therefore, inherently advantageous as a place to put a lunar base.

It may be said that architecture, being a very old profession, tends to enjoy its history and traditions. Certainly, architects enjoy building, and it is understandable that our first visions of lunar basing might demonstrate continuity with the heritage of terrestrial construction. Unfortunately, as we begin to come to grips with the complexities of lunar settlement, predictions of substantial construction and habitation on the lunar surface seem increasingly romantic.

Although detailed evaluations of candidate architectural schemes must await the framework of formal programming, meaningful comparisons of generalized surrface and subsurface basing concepts are possible. The results of our initial studies, which attempt to compare the various attributes of these two modes of development and identify inherent advantages and disadvantages, are shown in Table 1. This study is certainly not conclusive, but it does begin to suggest the applicability of several systems. Even at this stage, however, it seems clear to us that there are deficiencies inherent to all surface habitation schemes, and that the potential of lava-tube-based developments should be investigated further.

Looking at the disadvantages of lunar surface superstructures, it is apparent that there are significant technological issues that will always impose limits on the extent of construction and on other related aspects of architectural design. Even for the smallest surface habitats, the interwoven factors of pressure-vessel physics, thermal stressing of the enclosing skin, radiation shielding, and construction difficulty in a lethal environment present extremely perplexing problems.

The ability to create structures of highly variable morphology is not one of the strengths of this mode of development. The need for morphological complexity, flexibility, and revisability is dictated by functional, behavioral, political, and other considerations, and should not be undermined by inherent structural limitations. Resolving this contradiction will complicate any surface-based design. Further, in order to achieve safe and reliable structures on the surface, additional complication of the structure will be required. Inspection and maintenance needs will add still more complication. The alternative of subselene basing raises the matter of thermodynamic performance, for we must realize that, by comparison, surface structures are inherently poor performers.

As a rule, in order to construct similarly sized environments, with similar safety and performance expectations, we should expect surface-constructed bases to require more sophistication and greater quantities of construction materials. There may also be a need for greater degrees of precision in the manufacture of these materials. Overall surface settlement growth may therefore be inhibited by increased competition for base resources. Considering these limitations, it seems too great a stretch of the imagination to expect a construction sophistication capable of providing the very large internal volumes that are comparable even to small-scale lava tubes. Even if all other problems were to be resolved, failure to accommodate the spatial requirements of the inhabitants would invalidate any exclusive reliance on surface structures.

Finally, with surface-based systems we see many contradictions. For instance, the need for complex architectural form is in opposition to the principles of pressure vessel design, which calls for simplicity; the need for large volumes implies greater hull surface areas, which runs contrary to the issues of radiation shielding, thermal stress, and thermodynamic performance; and the material economy of thin-walled pressure hulls cannot be reconciled with the need for environmental ruggedness and macrometeorite protection.

As we review these issues and contradictions, two strategies of surface construction seem practicable. First, we would expect surface structures to permit an initial and early operational capability on the lunar surface. Early subselene deployment, in the form of lava tubes used as shelter for habitats, may provide an alternative to extensive surface development, and this prospect should be studied actively. However, initial operations from a surface base camp would seem mandatory in light of the need for precursor investigations of lava tubes. In this role for superstructural systems, many of the confounding issues that relate to permanent habitation would not be pertinent, thereby allowing the use of relatively simple structures.

Second, in combination with subselene adaptation, surface constructions will certainly fill many important roles; however, we do not believe these include long-term habitation. Many lunar operations will occur at the surface, requiring both pressurized and nonpressurized facilities. Vestibular surface constructions would be needed for surface access to subselene facilities. Eventually, it may even be desirable for an established subselene base to expand elements of its facilities upward by penetrating the cavern roof.

If surface-constricted superstructures are utilized for long-duration habitation, we may estimate some aspects of their architectural form. In this capacity, those proposals for lunar basing that have indicated a highly compartmentalized bomb-shelter-like environment seem most reasonable. Such an environment would necessarily have few access points, few windows, and be buried under some 7 to 12 ft of regolith. If constructed as a mass structure, possibly in concrete, its walls would probably be quite thick, its spaces forming a chambered matrix. Spatial hierarchy would be based, for a long time, on the distinction between the interior of the base and the inaccessible lunar exterior—there would be no "outside." For all intents and purposes, it would be a man-made cave.

LAVA TUBES

The existence, operational advantages, and favorable environmental conditions of lunar lava tubes were discussed by *Hörz* (1985). Speaking from the perspective of planetary geology, he discussed the theorized origin and formation of lunar lava tubes, and stressed the certainty of their existence. He went on to

TABLE 1. This table summarizes a systems comparison study performed by the authors and identifies the inherent advantages and disadvantages of six generic architectural systems.*

Architectural Systems Comparison

Identification of Inherent Advantages and Disadvantages of Various System Archetypes which are Candidates for Lunar Development

● INHERENT ADVANTAGE ✖ INHERENT DISADVANTAGE
○ POSSIBLE ADVANTAGE ✕ POSSIBLE DISADVANTAGE

	SURFACE				SUBSURFACE	
	HABITAT MODULES S/S-DERIVED OVERCOVERED	THIN SHELL RIGID FRAME OVERCOVERED	MASSIVE SHELL RIGID FRAME OVERCOVERED	SOFT SHELL (INFLATABLE) OVERCOVERED	LAVA TUBE-A (AS SHELTER FOR OTHER STRUCTURES)	LAVA TUBE-B (FULLY ADAPTED HABITAT)
POTENTIAL FOR EOC Does the system have the potential for Early Operational Capability?	●		✕	●	○	✖
SITING LIMITATIONS Does the system inherently limit the site selection process?	●	●	○	●	✖	✖
OVERALL ARCHITECTURAL DESIGN FREEDOM Is the system inherently self-limiting?	✖	✕		✕	○	●
VOLUME-ENVELOPE RELATIONSHIP Does the system offer any inherent advantage or disadvantage with regard to the ratio of constructed envelope to enclosed volume?		✕	✕			●
POTENTIAL FOR PERMANENCE Is there a realistic potential for permanent habitation?	✖		○	✖	○	●
FUNCTIONAL ADAPTABILITY How well will the system adapt to a changing range of functions?	✖	✕	✕	✕	○	●
FENESTRATION Can the system accommodate direct views of the exterior?	●	●	●	○	✖	✖
POTENTIAL FOR VERY LARGE VOLUMES Does system have the capacity to provide copious internal volumes?	✖	✕	✕	✕	○	●
STRUCTURE LIFETIME Is the structure lifetime predictable?	✕	✕	○	✖	○	●
SYSTEM POTENTIAL FOR SUBSTANTIAL RECONFIGURATION Can the structural system be revised easily?	✕	✕	✕	✕	○	●
PHYSICAL EXPANSION POTENTIAL Can the system be expanded easily?	○				○	●
OVERALL RELIABILITY OF STRUCTURE Can we expect reliability of the structural system?	○	✕		✖	○	○
RUGGEDNESS OF STRUCTURE Can the system stand up to rugged use and abuse?	✖	✕	○	✖		●
EXTREME THERMAL STRESS Will the system be subject to extreme thermal stressing?	✖	✖	✖	✖	●	●
COMPLICATION OF STRUCTURE DUE TO MAINTENANCE REQUIREMENTS Will the need for access, inspection, and regular maintenance cause the structure to be more complicated?	✖	✖	✖	✖	●	○
EVA-INTENSIVE CONSTRUCTION OPERATIONS Will the construction method require an extensive human presence?	○	✖	✖	○	○	○
OVERALL ENERGY MANAGEMENT Does the system have any inherent advantages or disadvantages?	○	✕		✕	●	●
BEHAVIOR OF STRUCTURE DURING FAILURE In the event of sudden decompression, will the structure fail catastrophically?	✕	✕		✖		○
DEPENDENCE ON PNEUMATIC PRESSURE FOR STRUCTURAL SUPPORT Does system depend on internal air pressure for structural support?	○	○	●	✕		●

	SURFACE				SUBSURFACE	
	HABITAT MODULES S/S-DERIVED OVERCOVERED	THIN SHELL RIGID FRAME OVERCOVERED	MASSIVE SHELL RIGID FRAME OVERCOVERED	SOFT SHELL (INFLATABLE) OVERCOVERED	LAVA TUBE-A (AS SHELTER FOR OTHER STRUCTURES)	LAVA TUBE-B (FULLY ADAPTED HABITAT)
STRUCTURE COMPLICATION DUE TO ATMOSPHERE CONTAINMENT ISSUES Will the need for redundancy and perfect integrity mandate complication of the structural system? Of the construction process?	✖	✖	✖	✖	✖	○
RESTRICTIONS IMPOSED ON ARCHITECTURAL FORM BY ATMOSPHERE CONTAINMENT ISSUES Will architectural design be directed or restricted by the issues of pressure-vessel engineering?	✖	✖	✖	✖	✖	○
RANGE OF MATERIALS WHICH CAN BE USED Will the system be subject to environmental extremes which will limit the types of materials which can be used for construction?	✖	✖	✖	✖	○	●
DEPENDENCE ON EARTH-IMPORTED CONSTRUCTION MATERIALS/COMPONENTS Does the system depend heavily on earth imports? Can a transition to indigenous materials be achieved easily?	✖	✕	✕	✖		○
INHERENT RADIATION SHIELDING Does the system offer any inherent capacity for radiation shielding?			○		●	●
DEPENDENCE ON REGOLITH FOR SHIELDING Is the placement of large quantities of regolith required for shielding?	✕	✕	✕	✕	●	●
COMPLICATION OF THE STRUCTURE DUE TO SHIELDING Will the need for regolith shielding mandate complication of the structure? Or the construction method?	✕	✕	✕	✕	●	●
PASSIVE MEDIATION OF LOW-G CONDITIONS: PHYSICAL EXERCISE Does the architecture affect the course of living on the moon in such a way as to passively require physical exercise within the base?	✖					●
MACROMETEORITIC IMPACT PERFORMANCE Is the system inherently susceptible to macrometeoritic impact?	✕	✕	○	✕	●	●
CELSS INTEGRATION Is the system receptive to the close integration of advanced CELSS?	✕					●
CELSS EVOLUTION Is there an inherent capacity in the system to allow for the maturation and evolution of CELSS technology?	✕				○	●
POTENTIAL FOR EVENTUAL TERRAFORMATION Does the system offer any potential for interior terraformation?	✖			✕		●
POTENTIAL FOR EXPANSIVE OPEN INTERIORS Can the need for copious open space be satisfied?	✖		✕		○	●
HIERARCHICAL SPATIAL DIFFERENTIATION Is there a capacity for meaningful hierarchical spatial differentiation?	✕					●
ENVIRONMENTAL STRESS Does the system offer any inherent advantages or disadvantages with respect to psychological stress due to the environment?	✕	✕	○	✖	●	●
CULTURE/POLITICAL SYSTEM FUNCTIONAL ADAPTABILITY If the lunar base program is pursued by one nation, or alternatively, by a coalition of nations – how well can the system respond to varying C/P requirements?	○				○	●

* A more detailed comparative analysis of candidate basing systems awaits the framework of a specific lunar base program statement. The reader should note the important advantages that are offered by systems that depend on the sheltering capacity of lunar tubes, suggesting a course that sees an early transition from initial surface constructions to the deployment of facilities within nearby tubes.

suggest how these natural lunar caverns may have superior potential as habitat shelters. In summary, Hörz provided us with the following overview.

First, we know that lunar lava tubes exist. They are observable as being related to the numerous sinuous rilles, or lava flow channels, that are found abundantly on lunar basalt surfaces. These flow channels are believed to be collapsed sections of lava tubes and, in a number of instances, remaining sections of intact tube become apparent with the observation of uncollapsed roof segments. It is noteworthy that while the frequency and global distribution of lava tubes are not well understood, they are subsurface features, and fully intact tubes will not normally be recognizable from surface imagery.

We can also observe that lunar tubes are significantly larger and more sinuous than terrestrial analogs. By scaling various rilles and uncollapsed roof segments, typical widths and depths of tubes can be estimated in the hundreds of meters, with overall lengths commonly measuring a few kilometers. Restrictions and enlargements within the interior of lava tubes may occur (as they do in terrestrial lava tubes), but it is suggested that the relief scale of these features is typically small when compared to cross-sectional dimensions. Figure 1 indicates a number of lava flow features, including one known lava tube (scalloped linear feature at the lower center of the photograph); these observable features may be suggestive of lava tube morphology.

ORIGINAL PAGE
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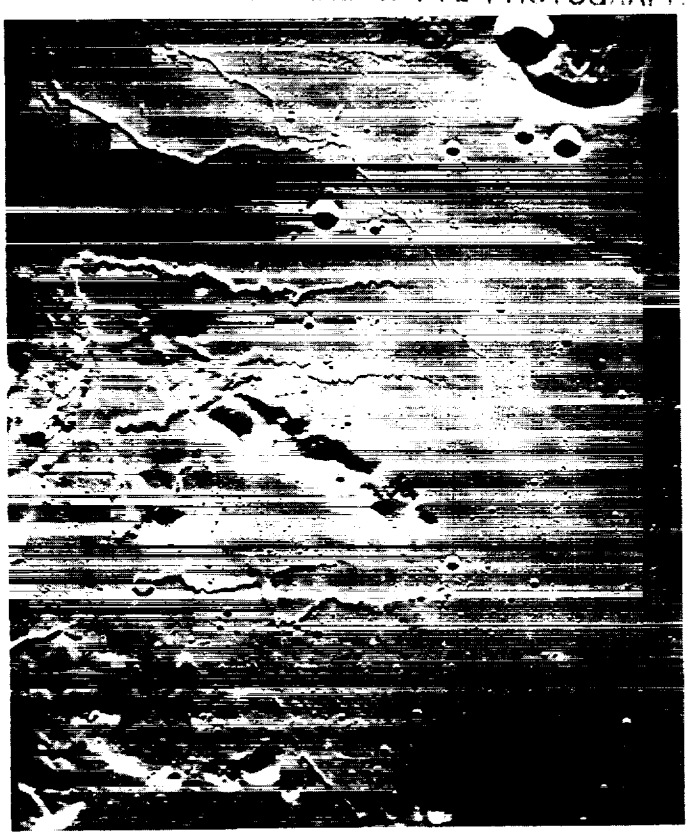

Fig. 1. The morphology of lunar lava tubes is suggested by these lava flow features, some of which may be depressions caused by the collapse of lava tubes. Note the variability of scale and the proximity of craters and mountains. Segments of uncollaped tube segments can be seen at the bottom center. (Lunar Orbiter V, frame M-19.)

Lava tube roof thicknesses seem to be more than sufficient to provide superior radiation shielding and protection from meteorite impacts. Deducing from beam-modeling techniques, basalt "bridges" (lava tube roofs) of at least 40 to 60 m in thickness would be required to span the observed widths of a few hundred meters. If the proportional relationship of roof thickness to cross-sectional dimension in terrestrial lava tubes is any indication, we should expect to see typical roof thicknesses ranging from 0.25 to 0.125 of cross section. Crater impact studies further support these estimates.

Uncollapsed lava tubes are further observed to have sustained substantial and repeated meteorite impacts. It is noted that the expended energy from some of the larger impacts would equate to several tons of TNT (Hörz, personal communication, 1988), and while lava tubes seem well capable of withstanding such a direct shock, similar performance by surface-situated superstructures is difficult to envisage.

Within the large and well-protected interiors of lava tubes, the concerns of material degradation, thermal fatigue, and related exposure problems are moderated or negated, and it becomes possible to utilize a far wider range of materials and electronic devices. The interiors of lava tubes also give direct access to lunar bedrock (a rare condition), and this could be a substantial asset to the operation of heavy equipment, the stabilization of vibrating machinery and scientific equipment, and the founding of structural partitions and building components. It is estimated that the interior temperature of lava tubes remains unaffected by diurnal surface temperature variations, and remains a constant -20°C.

Hörz also mentions a number of possible disadvantages of lava tube basing, most notably the difficulties associated with accessing the tubes, as well as the question of lunar resource distribution and lava tube site selection.

THE PROMISE OF SUBSELENE DEVELOPMENT

From an architectural standpoint, the most profound advantage to be attributed to subselene development concerns the practicality of achieving very large internal environments. It is difficult to conceive any form of human habitation on the Moon—beyond only the earliest outpost bases—that do not provide for very large and even vast volumes of internal space. The permanent transition from terrestrially scaled open spaces to the enclosure of a spatially limited lunar base is simply too much to demand from any human being.

How much space is enough space? In lieu of empirical data on the human need for space in autonomous lunar environments, perhaps the most effective way to appreciate this issue may be by imagining oneself inside a permanent lunar station, confined, where there is no "outside" to escape to. Ultimately, if we cannot answer the need for copious space, it may not be possible for us to adapt to the Moon.

Is confinement to small and unyielding rooms and corridors an acceptable condition in a lunar base? In the context of life on Earth, these conditions would be considered punishing. Even for lunar base volunteers engaged in the most interesting work, dedication and eager expectations may give way to the reality of a very dull and encumbering place. It becomes easy to see how a badly designed and unsympathetic environment can, at the very least, severely weigh on the minds of men and women. The argument for returning humans to the Moon (in lieu of robots) is based on our intrinsic ability to think, to learn, to react, and

to be creative—all aspects of humanity that prisons are designed to defeat. Living permanently on the Moon will not be purposeful if we create places that effectively emulate penal institutions.

In time, research may yield some insight into this question of how much space is enough, and we should not be surprised if current expectations prove inadequate. It can be predicted, however, that if provided with essentially inadequate space, long-term lunar inhabitants will—in short order—seek more realistic designs that are not tied to a misconstrued or Earth-biased economy. Looking forward to the real needs of long-term basing, we should seek only those modes of architectural development that are capable of answering this essential need for space. The practical capacity to provide near-term expansive interior volumes seems to exist presently in lava tubes. Considering the limitations of even the largest plausible surface-deployed structures, it is stimulating to consider the architectural potential of a secure natural cavern with the multi-hundred-meter cross sections and multikilometer lengths that Hörz speaks of.

Indeed, if lava tubes are pursued as habitats, an early developmental problem will exist in that many tubes may be too large for practical purposes. Unfortunately, we are troubled because too little is known about the nature of these caverns, and we are forced to speculate about the dimensions of tubes that have defied detection. It does seem reasonable to expect, however, that a wide range of usable tubes will be found, and that modestly sized tubes could be made available for early stages of development. Eventually, larger tubes could be accessed and adapted. Conceivably, the progress of this adaptation could be staged, beginning with a small tube and advancing therefrom. Most importantly, it should be understood that the need for copious interior volumes can be accommodated by exploitation of a natural lunar feature.

Another beneficial aspect of lava tube exploitation involves the degree of internal complexity and variation that is typical of these features. Ironically, some have suggested that this very issue—the relief scale of restrictions and enlargements—is a negative aspect of lava tube deployments since it may inhibit the installation of various technologies, hinder trafficability, etc. From an architectural standpoint, however, this variability can only be viewed as an asset. Related in a sense to the need for copious space is the need for environmental stimulation, and here spatial variation and greater scales of surficial relief may be seen as features that work to define the environment as an interesting place.

Issues that relate to base morphology and, in particular, the need to vary and revise the form of the base over time, are also well received in lava tubes. With reliance on the surrounding monolith for structure, enclosure, and radiation protection, the number of confounding form-determining factors can be reduced, and the design can be better aimed at the critical functional, behavioral, and political considerations.

We note that the environment within subselene voids is far less threatening than the surface environment and, in a sense, the lunar subsurface is more Earth-like than any other place on the Moon. Furthermore, the basalt mantle surrounding the tube is, in essence, a carvable matrix that can be cut and sculpted into the widest range of architectural forms, such as those suggested in Fig. 2. It is not difficult to imagine the manner in which tube development could proceed: Lava tubes could be enlarged and

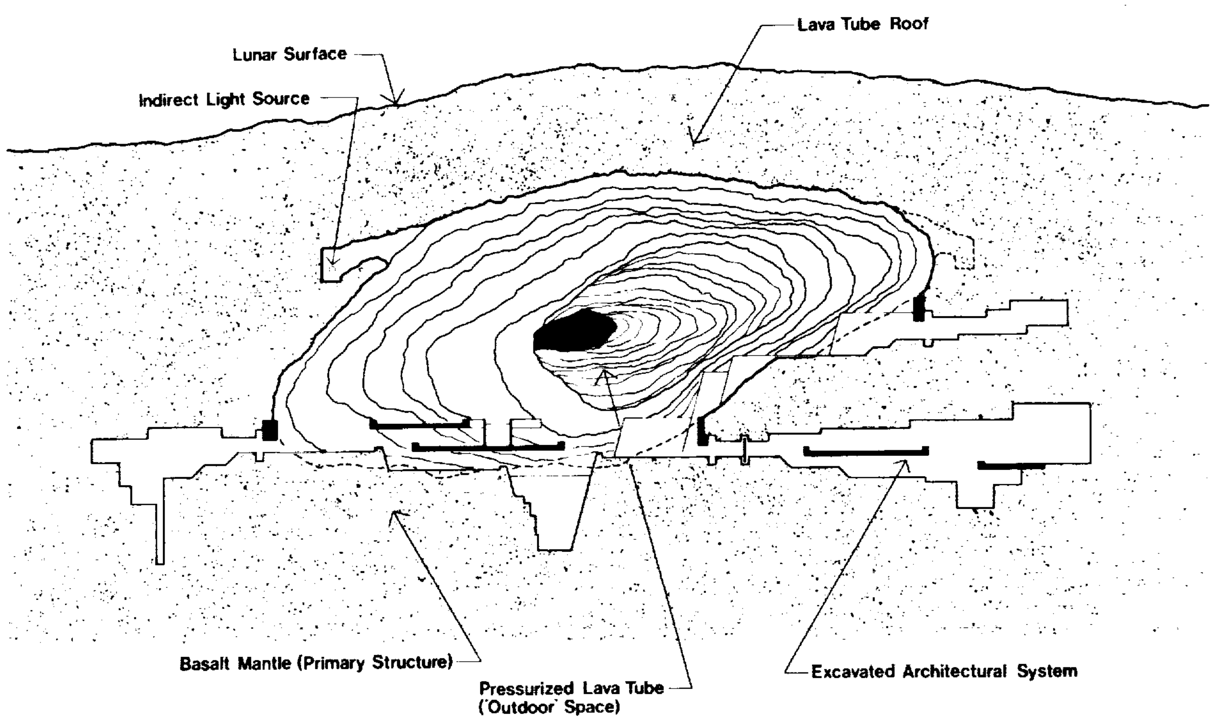

Fig. 2. While lava tubes may be exploited in the initial establishment of a lunar base simply as shelter for other structures, it is also conceivable that, eventually, entire tube segments could be sealed off and pressurized. In this role, the surrounding basalt mantle would provide the primary lunar base envelope. The architecture of the base could be created not only by placing structures within the tube, but also by excavating the tube walls, cutting away stone and creating usable spaces as required. The vast interior of the tube, measuring perhaps several hundred meters in cross section, could provide the spatial volumes and hierarchy necessary for permanent habitation.

reconfigured by the simple removal of material; new cavities could be created and appended to the tube by excavating through tube walls and floor; two or more proximally situated tubes could be connected by tunneling; penetrations through lava tube roofs could also be made, providing direct communication with surface constructions. Significantly, the option to revise, reconfigure, even to abandon particular spaces, would always remain available. It is conceivable that, from a primary lava tube, a virtual labyrinth of spatial successions and hierarchies could eventually be carved out, creating a very interesting place indeed. Traditional apprehensions concerning the high cost of mining and earth-moving put aside, the subselene milieu may well prove far preferable than any open field on the surface.

BUILDING IN THE SUBSELENE MILIEU

Lava tube interiors are far more conducive to a far wider range of construction operations and materials than the surface. We have already alluded to the fact that there are considerable advantages that relate to the performance and range of available construction materials. These advantages relate to the superior thermal and electromagnetic protection provided by the profound situation of tube environment. We can expand on these advantages by considering the possibilities for construction within tube environments, particularly in the case where entire tube segments are pressurized and transformed.

Construction Conditions

Within such a setting, the first great advantage for construction would be the substantially reduced danger to construction workers. Traditional notions of extravehicular activity (EVA) practice and precaution could, with care, give way to far more productive operations, quite possibly even within shirtsleeve conditions. With less need to rely on robots and teleoperation, more time devoted to actual construction, and fears allayed, we could expect dramatic improvements in construction capabilities, as well as related base activities such as mining and manufacturing. In the case of lava tubes used as shelters for habitat modules, EVA construction operations could be practiced with a greater level of safety than could be achieved at the surface.

Masonry Construction

Fully exploited tube segments allow architectural constructions within the enclosure that are adjunctive, and which are not necessarily prescribed by the need to contain atmospheric pressure. Various scales of habitational adaptation and spatial definition within pressurized tubes could indeed be achieved with forms and materials that would otherwise be inappropriate to pressure-differentiated structural skins. Within a pressurized lava tube, it is quite possible that simple masonry construction methods could find wide application. Here is a potential use of largely unprocessed indigenous material (stone) that could go a long way toward the goal of creating a very large and sophisticated environment without competing with other base operations and resources. The use of stone, the Moon's most abundant natural resource, seems to us a rather elegant proposition.

Concrete

The intriguing potential of lunar-sourced calcium cements for base construction has been pointed out by several authors. *Young* (1985), *Cullingford and Keller* (1991), *Lin* (1985), *Lin et al.* (1988), *Nanba et al.* (1988), and *Ishikawa et al.* (1988) are all notable in their discussion of lunar concrete from both experimental and practical views. If cementitious products prove to be viable on the Moon, we feel that there will be no better site for their application than within lava tubes, where environmental moderation during processing, application, and curing is a clear advantage.

Cementitious products may find a very wide range of applications within subselene environments, most notably in the form of concrete. Cementitious pargings may be a practical means of sealing lava tube interior surfaces and cracks. Simply poured concrete mass structures and floor slabs may provide a means of defining areas and reshaping spaces. Reinforced concrete may find great application as a highly adaptable structural system, for use in spanning large areas, and also as a means of partitioning lava tube segments. Given the unpredictable and highly irregular interior of a lava tube, the highly plastic and conforming nature of concrete will undoubtedly prove to be a great advantage.

Fused Structures and Surfaces

Khalili (1985, 1988) discusses the adaptability of masonry-type structures to the lunar scene as he asks us to recall the ways in which vernacular builders have come to rely on these methods throughout history. He also recalls for us a similar methodology whereby stone-masonry constructions can be thermally fused *in situ*, creating mass constructions and even spanning structures of exceptional strength. Such thermally fused mass constructions may find their best application where there is no need for atmosphere containment, and where the availability of cement constituents, principally water, is insufficient. This thermal-fusing technology may also be quite useful as a means of sealing the interior surfaces of lava tubes and excavated spaces, and of giving strength to any masonry construction used within the tube.

Inflatable Structures

Inflatable structures have been proposed for use as lunar habitats by many authors. While this class of structure may offer some advantages as a means of establishing a surface base (particularly in the early phases of development), we would like to mention their possible application in lava tubes. Because access to a lava tube is likely to be difficult, inflatable structures would seem to offer the advantage of improved mobility. If an early capability for subsurface lunar basing is sought, the use of packaged inflatable habitats within lava tubes would seem almost mandatory. The advantages of placing inflatable or nonrigid structures within the protection afforded by a lava tube are substantial, and the combination of these two elements may indeed evolve into a plausible outpost-phase strategy for lunar basing. Figure 3 illustrates the placement of an inflatable structure (as well as space-station-derived habitat modules) within a small lava tube.

Spaceframes

Modular three-dimensional trusses, or spaceframes, are another form of construction that we feel would be particularly well-suited for subselene situations. Spaceframe systems are in widespread terrestrial use, and they are finding growing application in space, where their performance is being studied. (The space station will eventually be structured around a spaceframe truss system.) It is conceivable that lessons learned with spaceframes in low Earth

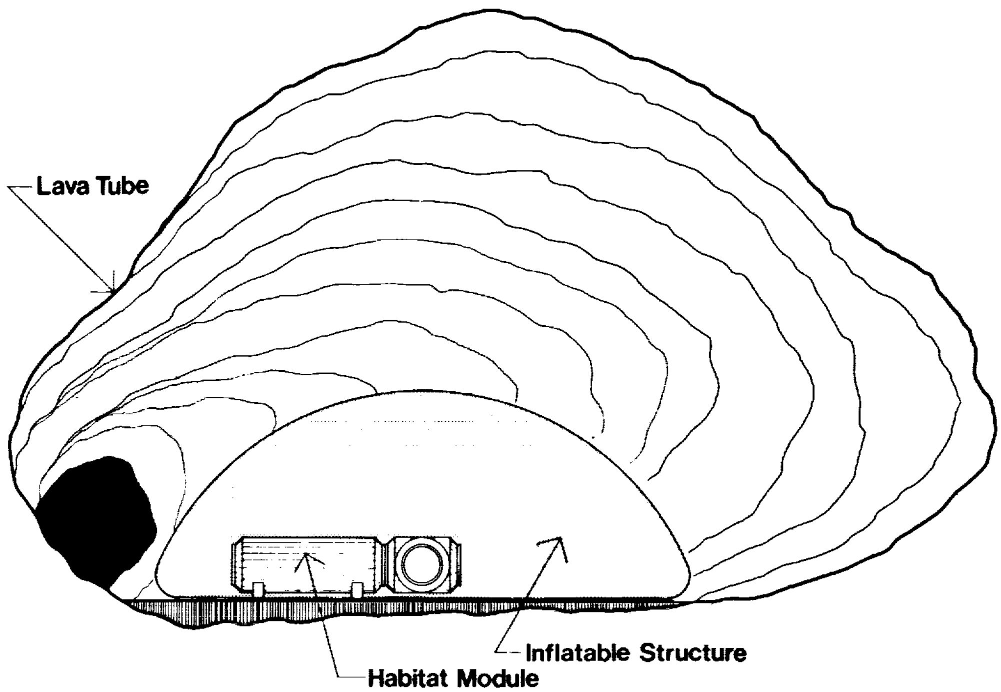

Fig. 3. The placement of habitat modules and inflatable structures within a lava tube may offer significant advantages as a means of base expansion following an initial surface deployment. Structures placed within the tube would not require any radiation shielding, and would not be subject to the thermal extremes normal at the surface. EVA operations and other activities could proceed with considerably less risk. The placement of "packaged" inflatable structures within an open tube may provide the best means of establishing an advanced lunar habitat.

orbit (LEO) may favor their application on the Moon. We are intrigued by this technology for several reasons.

Principally, spaceframes offer an extremely versatile technology for spanning large and irregularly shaped areas. While not moldable in the sense of concrete, spaceframes readily conform to a limitless range of two- and three-dimensional geometries, thereby allowing them to easily adapt to the variable shape of any lava tube or excavation. Spaceframes are versatile enough to be used for both surface and subsurface modes of development, and they represent one of the few practical modes of development that are well-suited to operate in both environments.

The two primary elements that combine to create the three-dimensional truss, the hubs and struts, are easily produced, and may be manufactured from a variety of materials. The source of these materials may be simply transitioned from the Earth to the Moon, without great disruption of construction practice. Spaceframes may be assembled and disassembled repeatedly, and while teleoperated and robot assembly are possible, construction by humans has been simplified to the point where assembly without tools is practical.

CONCLUSION

The purpose of this paper has been to present the authors' belief that subselene lunar basing may provide the most satisfactory and comprehensive solution to the extreme problems posed by lunar architecture. We have elucidated a number of key issues in an attempt to underscore the difficulty that we foresee, and to persuade the reader that a radical architectural solution is essential.

We believe that the development of a time-scaled architectural program is required for any serious future study of lunar base habitation. Using this as the basis for continuing study, various disciplines may begin to compare notes and work toward the eventual resolution of the architecture. Progress toward the definition of the architecture may in turn lead to revised expectations of lunar base potential.

What becomes clear as one begins to view even the most rudimentary version of this program is that the time-honored methods that have yielded our heritage of building structures on Earth (or, for that matter, in LEO) should not be allowed to prejudice our approach to building on the Moon. Certainly the materials and technologies in use in modern construction practice on Earth cannot be easily transferred to the Moon. But more profoundly, the very notion of constructing a "building" on the Moon must be questioned. Subselene development offers the real prospect that our most tenuous early foothold on the Moon may be allowed to evolve into an enduring settlement.

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REFERENCES

Bova B. (1987) *Welcome To Moonbase.* Ballantine, New York. 254 pp.

Chow Y. P. and Lin T. Y. (1988) Structures for the Moon. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 362-374. American Society of Civil Engineers, New York.

Coombs C. R. and Hawke B. R. (1988) A preliminary survey of lunar rilles: The search for intact lava tubes (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 58. Lunar and Planetary Institute, Houston.

Cullingford H. S. and Keller M. D. (1992) Lunar concrete for construction. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Diamandis P. H. (1988) The need for artificial gravity on the Moon (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 71. Lunar and Planetary Institute, Houston.

Duke M. B., Mendell W. W., and Roberts B. B. (1985) Strategies for a permanent lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 57-66. Lunar and Planetary Institute, Houston.

Dula A. M. (1988) Lunar law—Regulation of human activities on the new frontier (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 75. Lunar and Planetary Institute, Houston.

Eijadi D. A. and Williams K. D. (1988) Extraterrestrial applications of solar optics for interior illumination. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 298-309. American Society of Civil Engineers, New York.

Finney B. (1985) Lunar base: Learning to live in space. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 751-756. Lunar and Planetary Institute, Houston.

Grabynowicz J. I. (1992) The "Province" and "Heritage" of mankind reconsidered: A new beginning. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Grün E., Zook H. A., Fechtig H., and Giese R. H. (1984) Collisional balance of the meteoritic complex. *Icarus, 62,* 244-272.

Hörz F. (1985) Lava tubes: Potential shelters for habitats. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 405-412. Lunar and Planetary Institute, Houston.

Ishikawa N., Okada T., and Kanamori H. (1988) Making concrete for lunar structures (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 122. Lunar and Planetary Institute, Houston.

Johnson S. W. and Leonard R. S. (1985) Evolution of concepts for lunar bases. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 47-56. Lunar and Planetary Institute, Houston.

Joyner C. C. and Schmitt H. H. (1985) Extraterrestrial law and lunar bases: general legal principles and a particular regime (INTERLUNE). In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 741-750. Lunar and Planetary Institute, Houston.

Kaden R. A. and Pense L. D. (1988) Planetary base inflatable form construction. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 540-551. American Society of Civil Engineers, New York.

Kaplicky J. and Nixon D. (1985) A surface-assembled superstructure envelope system to support regolith mass-shielding for an initial-operational-capability lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 375-380. Lunar and Planetary Institute, Houston.

Kelso H. M., Hopkins J., and Thomas M. (1988) Design of a second generation lunar base. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 389-399. American Society of Civil Engineers, New York.

Khalili E. N. (1985) Magma, ceramic, and fused adobe structures generated *in situ.* In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 399-403. Lunar and Planetary Institute, Houston.

Khalili E. N. (1988) Regolith and rocks for lunar structures and shielding (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 136. Lunar and Planetary Institute, Houston.

Klassi J. (1988) Lunar sub-surface architecture enhanced by artificial biosphere concepts (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 137. Lunar and Planetary Institute, Houston.

Land P. (1985) Lunar base design. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 363-373. Lunar and Planetary Institute, Houston.

Lin T. D. (1985) Concrete for lunar base construction. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 381-390. Lunar and Planetary Institute, Houston.

Lin T. D., Love H., and Stark D. (1988) Physical properties of concrete made with Apollo 16 lunar soil sample (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 159. Lunar and Planetary Institute, Houston.

Lowman P. D. Jr. (1985) Lunar bases: A post-Apollo evaluation. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 35-46. Lunar and Planetary Institute, Houston.

Mangan J. J. (1988) The expandable platform as a structure on the Moon. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 375-388. American Society of Civil Engineers, New York.

McCauley L. A. and Watson P. M. (1988) Lunar mining and construction: A status report (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 172. Lunar and Planetary Institute, Houston.

Nanba H., Kai Y., Sugihara S., Matsumoto S., Yoshida T., and Osawa T. (1988) Construction of concrete lunar structures (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 180. Lunar and Planetary Institute, Houston.

Polette T. and Toups L. (1988) A phased approach to lunar-based agriculture. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 287-296. American Society of Civil Engineers, New York.

Richter P. J., Anderson T. L., Drake R. M., and Kumar R. P. (1988) Strategy for design and construction of lunar facilities. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 904-915. American Society of Civil Engineers, New York.

Robinson G. S. and White H. M. (1986) *Envoys of Mankind: A Declaration of the First Principles for the Governance of Space Societies.* Smithsonian Institution, Washington. 292 pp.

Rocha C. J. and Carr C. (1988) Thermal tunneled first stage lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 205. Lunar and Planetary Institute, Houston.

Santy P. A. (1988) Biobehavioral considerations for manning a lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 213. Lunar and Planetary Institute, Houston.

Sherwood B. (1988) Lunar architecture and urbanism (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 217. Lunar and Planetary Institute, Houston.

Silberberg R., Tsao C. H., Adams J. H. Jr., and Letaw J. R. (1985) Radiation transport of cosmic ray nuclei in lunar material and radiation doses. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 663-669. Lunar and Planetary Institute, Houston.

Soeder D. J. (1988) Gas-sealing abilities of bedrock in an underground lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 227. Lunar and Planetary Institute, Houston.

Solar System Exploration Committee (1983) *Planetary Exploration Through the Year 2000.* U.S. Government Printing Office, Washington, DC. 167 pp.

Szilard R. (1959) Structures for the Moon. *Civ. Eng., 29,* 46-49.

Vaniman D. T., Heiken G., and Taylor G. J. (1985) A closer look at lunar vulcanism from a base on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 211-222. Lunar and Planetary Institute, Houston.

Watson P. M., Lundquist R. G., and Blacic J. D. (1988) Construction of an underground lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 252. Lunar and Planetary Institute, Houston.

Ximenes S. W., Winisdoerffer F. J., and Brown J. D. (1988) Lunar Ecosystem and Architectural Prototype (Project LEAP) (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 259. Lunar and Planetary Institute, Houston.

Young J. F. (1985) Concrete and other cement-based composites for lunar base construction. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 391-403. Lunar and Planetary Institute, Houston.

LUNAR SUBSURFACE ARCHITECTURE ENHANCED BY ARTIFICIAL BIOSPHERE CONCEPTS

N93-17448

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The integration of artificial biosphere technology with subselene architecture can create a life-enhancing, productive habitat that is safe from solar radiation and extreme temperature fluctuations while maximizing resources brought from Earth and derived from lunar regolith. In the short term, the resulting biotectural (biospheric and architectural) designs will not only make the structures more habitable, productive, and manageable, but will ultimately provide the self-sufficiency factors necessary for the mature lunar settlement. From a long-term perspective, this biotecture approach to astronautics and extraterrestrial development (1) helps reduce mass lift requirements, (2) contributes to habitat self-sufficiency, and (3) actualizes at least one philosophy of solar system exploration, which is to exploit nonterrestrial resources in an effort to conserve our natural resources on this planet.

INTRODUCTION

This study does not propose the viability of a completely self-sufficient lunar habitat any time in the near future, but it does recommend that the initial design stages take artificial biosphere concepts into consideration. The implementation of these regenerative technologies into the proposed subselene habitat scheme may be more costly initially, but the costs of long-term operation will be reduced as the area of the facility expands to accommodate the regenerative artificial biosphere systems (*Hypes*, 1988).

BIOSPHERICS, THE BIOSPHERE, AND ARTIFICIAL BIOSPHERES

Biospherics has been defined as an "integrative science of the life sciences just as astronautics is an integrative science of the physical sciences. Together with astronautics, biospherics opens up the ecotechnical possibilities to expand life on Earth into other parts of our solar system" (*Allen and Nelson*, 1986).

Earth's biosphere, which is the layer of life on the surface of the planet inclusive of the atmosphere, is the only presently known biosphere. This planetary-wide system has been continually present on Earth for over 3.5 b.y. and is the primary geologic force maintaining the transitory balance between the gaseous, solid, and liquid matter of the planet. The biosphere supplies most of the free energy that fuels humans (with the air they breathe and the food they eat) and their machinery. Whenever humans venture beyond Earth's biosphere into space, they must always take some form of the biosphere with them in order to survive, be it containerized food, oxygen, or fuel. There is an intrinsic connection between humans and the Earth's biosphere that becomes more and more apparent the further "disconnected" or away from Earth we get. Space travelers need food, water, an adequate atmosphere, and a protective, psychologically suitable shelter. Limited by such things as rocket payload lift capability, economics, and an incomplete understanding of the biosphere's mechanisms, the challenge is to design an evolving habitable environment that will provide these necessities for continually longer periods of time with fewer and fewer resupply missions.

Useful human habitation of the Moon will require environmental conditions similar to those on Earth where humans evolved. Like the biosphere of planet Earth, lunar habitats as artificial biospheres should strive to be stable, complex, evolving systems containing life, composed of various "scaled-down" ecosystems operating in synergetic equilibrium, essentially closed to material input or output, and open to energy and informational exchanges. As we have yet to create a livable artificial biosphere, achieving this goal will not come in the first phase of lunar base development, or perhaps not even in the second or third phase. However, applying what we know in the initial stages will only expedite the growth of our first lunar "hut" into a life-promoting artifical biosphere habitat just as the Earth's biosphere evolved from a once barren planet.

A TUNNELING SCHEME FOR A LUNAR BASE

A first-stage lunar base can be created using a tunneling device that produces an underground network of habitats, work spaces, and passageways. The resulting interior walls of the tunneled chambers are hardened silica to which inflatable membranes can be attached and deployed creating the desired environment. These habitats will be essentially isolated from the lunar surface environment by a closed structure composed of components derived from the lunar surface itself and an inflatable interior bladder. The scheme also provides a practical way to create the large amounts of pressurized volume needs for the regenerative life support of a large lunar habitat. The basis for building subselene facilities are (1) protection from radiation and meteorites, and (2) a relatively constant temperature of a -20°C, which exists at depths of 3 m below the lunar surface.

Background

Current thinking on the initial lunar base configuration is exemplified by *Kaplicky and Nixon* (1985) in which a prefabricated module is built on Earth, launched into low Earth orbit,

then transported to the lunar surface. At this point the module will be soft landed on the surface and then covered with regolith to protect the inhabitants from harmful solar and cosmic radiation. The modules could be existing space station designs and outfitted with much the same hardware.

The application of this logical scenario for an initial lunar base configuration would advance the knowledge of human productivity and technology interfaces in a foreign environment. In the meantime, testbed habitats conducted in Antarctica, underwater, and in other harsh and isolated environments are furthering this understanding. Based upon this current thinking, we can examine an alternative scenario facilitating mankind's expansion to the Moon.

A proposed cost-effective method for an alternative next phase lunar base using indigenous materials could be achieved by using a thermal tunneling device (*Rowley and Neudecker,* 1985) and an inflatable membrane configured for specific uses. As seen in previous studies of lunar habitats, most involve heavy manufacturing of components and modules. These systems need to be launched from Earth, placed in lunar orbit, and then deployed on the surface. It would seem reasonable to use existing lunar materials to create the shell of these habitats, thus reducing the cost of the heaviest component by using material that is indigenous to the lunar environment. This system also holds potential as a way to enclose the large amounts of pressurized volume required for Controlled Ecological Life Support System (CELSS)/biosphere applications.

As we have seen throughout architectural/construction history, humans have always used materials and processes that were indigenous to the local region. From the use of ice for igloos in Alaska, bamboo and grass for habitats in jungles, and sod and adobe during the expansion into the American West, people have adapted their resources, knowledge, economics, and creativity to serve their needs. Practical designs for space facilities are derived from that same context of creativity and resourcefulness. The use of indigenous lunar materials could eliminate a major cost while providing a shielded habitat.

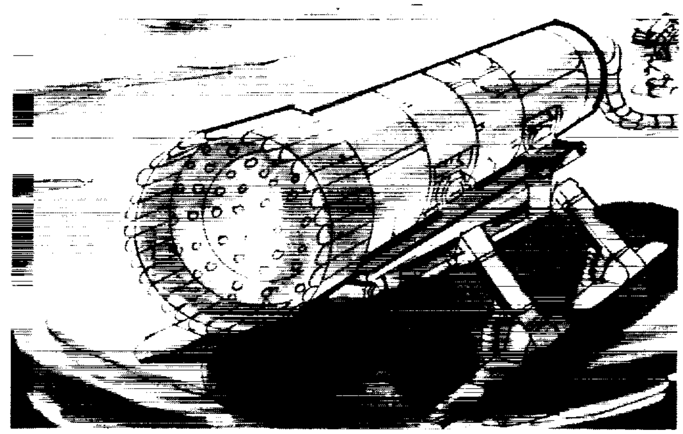

Fig. 1. The melt-tunneler separates from the systems tractor.

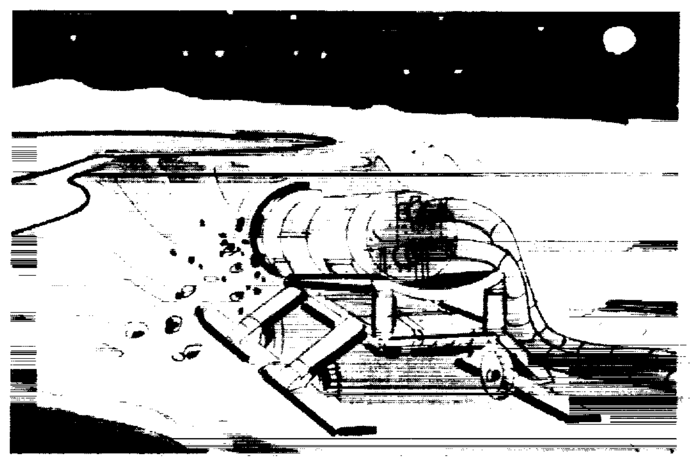

Fig. 2. The device is activated and begins to core.

Subsurface Scenario

A device called a subselene nuclear powered melt tunneler was presented in 1985 by J. W. Neudecker and J. C. Rowley (*Rowley and Neudecker,* 1985). This concept used heat from a nuclear reactor to melt rock and form a self-supporting, glass-lined tunnel. They favored subselene tunneling for the following reasons: (1) the process uses highly energy efficient nuclear power supply, (2) it does not require water or other rare volatiles for open system residue handling or cooling, (3) the mechanism can penetrate through a varied sequence of rock types without complicated configurational changes, (4) the process forms its own support structure as it goes along, and (5) the system is highly adaptable to automated operation.

This type of mechanism can be launched into orbit with an expendable rocket system. The unit would proceed to a low lunar orbit, deploy the protective transportation shields, and begin its descent to a predesignated site on the Moon's surface. After soft landing, the tunneling device detaches from the support tractor containing the main guidance computer, Earth communication system, and regolith extraction pumps (Fig. 1). Then the tunneling device manuevers into position, attains the required angle, and ignites (Fig. 2). Once there is a cavity, the device attaches its gripping pads to the sides of the newly formed tunnel and continues digging in a preprogrammed configuration, be it linear, circular, or octagonal (Figs. 3 and 4).

After the inner walls of the tunnel have cooled down sufficiently, astronaut construction crews would arrive and install the inflatable interior membranes, which are transportable in a compacted form (Figs. 5 and 6). These membranes would provide a secondary pressure skin should the fused glass walls of the tunnel crack or lose pressure integrity. Membranes previously researched by *Goodyear Aerospace Corporation* (1982) were expandable, flexible structures with enough rigidity to infill the interiors of the tunneled sections. They could be easily attached to the hard silica walls and expanded.

These inflatable units would be predesigned and outfitted according to their desired uses. Each module could be 40 ft in length and 12 ft high at the center point of the barrel vault featuring ready-made interior partitions and circulation corridors. Ducts and tubes would already be in place within the framework composition of the inflatable material.

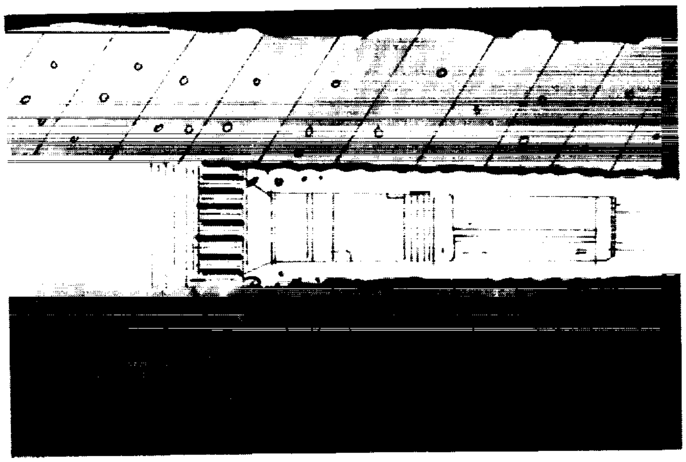

Fig. 3. Subselene chambers are created.

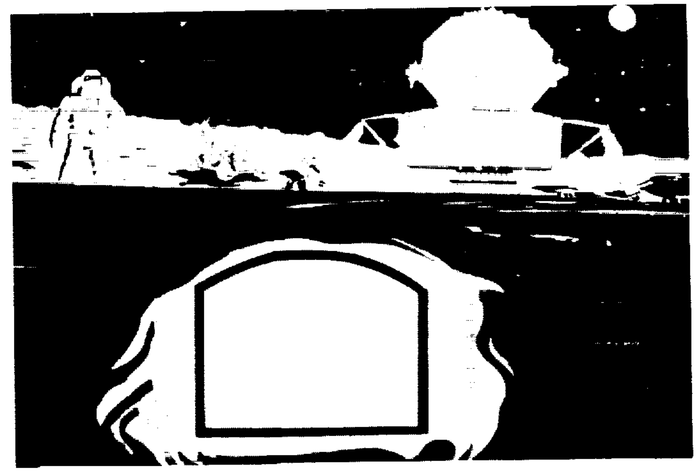

Fig. 5. Astronauts land with interiors.

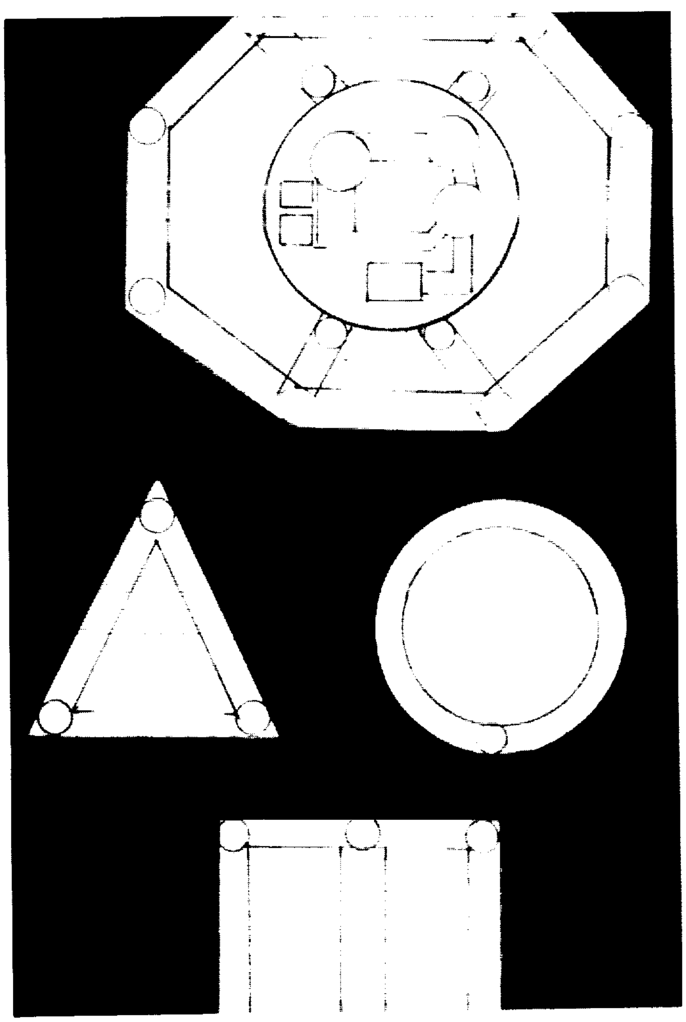

Fig. 4. Preprogrammed configurations of tunneler paths.

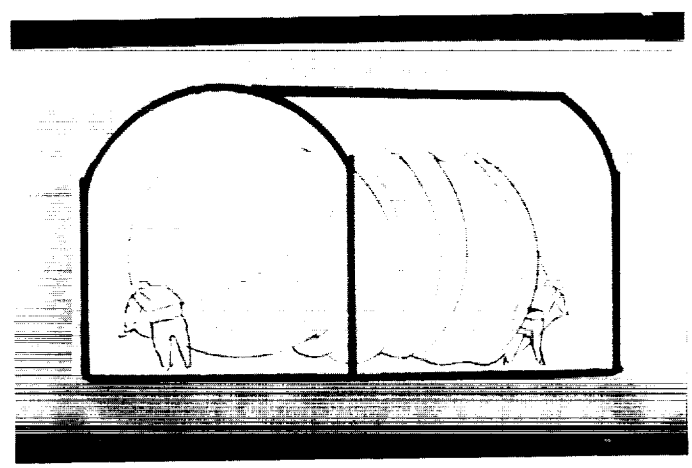

Fig. 6. Install inflatable interiors safe from surface radiation.

Each module can be connected to another module with 12-ft-diameter interlink node providing an entry/exit airlock to the exterior environment. The interlink nodes are also suitable as emergency rescue pods when providing access to the surface. Similiar nodes are being considered for use in the space station. Modules and nodes could also be pressurized for use as artificial biosphere chambers (Fig. 7).

Large volumes designated for artifical biosphere applications, perhaps on the order of thousands of square feet, may not require the extra security the inflatable interior offers if the volmes were not inhabitated on a day-to-day basis. The glass-walled tunnels themselves could be sealed off into large pressurized growing chambers. Interiors could then be attached to the mechanical equipment or the life support system.

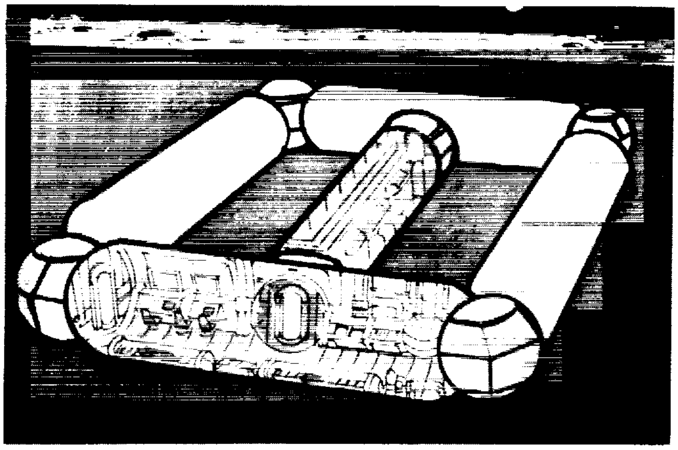

Fig. 7. Modules can be previously outfitted for labs, growing areas, or habitats.

This subselene scenario presents alternative architectural and construction processes that are compatible with plans to integrate CELSS or artificial biosphere technology into lunar habitats. The proposed scenario features the following:

Minimum surface time. The hazardous radiation astronauts are exposed to during extravehicular activity (EVA) construction can be reduced.

A balance of manned and automated technology. Surface construction tasks that subject humans to hazardous solar radiation could be accomplished by the use of automated subselene systems.

The following construction and operational tasks requiring the presence of humans can be conducted for the most part in the resulting, protected subselene conditions.

Expendable launching system. Expendable launching systems are viable options for deployment of the tunneling device. Transportation and delivery of lunar excavating equipment and other automated systems can economically occur before manned flights are launched.

Earth-based analogs. Tunneling and construction in harsh and isolated environments such as Antarctica and underwater continue to provide valuable information relevant to this subselene scenario.

Economical creation of pressurized volumes. Elimination of heavy material support from Earth could allow a mature lunar settlement to achieve self-sufficiency by the creation of adequate habitation area and the required large volume necessary to implement a biospheric life support system.

A minimum facility volume required to merely feed a hypothetical crew of six lunar astronauts can be calculated. Data from Martin Marietta indicate that one hypothetical lunar base crew member requires 1.36 lb of food per day (*Hypes,* 1988). A Phytofarm hydroponic facility in Illinois is currently producing 1 lb of food per day for every 25 sq ft of the facility (*Field,* 1988). Calculations from this data indicate that every crew member requires 34 sq ft of growing space to meet the 1.36 lb of food per day. A six-member crew would therefore require approximately a one-story, 204-sq-ft hydroponic facility. As the Phytofarm facility is a large-scale operation, a contingency factor for the minimum volume of support systems required to produce a certain amount of biomass comes into play in this calculation. Since the melt tunneling technology can evacuate subterranean sections up to 15 ft wide, an intensive hydroponic facility only 12 ft long would provide the necessary volume needed for supplying food.

INTEGRATION OF ARTIFICIAL BIOSPHERE CONCEPTS

The Earth's biosphere did not happen overnight but grew from a sterile world to an evolving, complex living entity. Over its approximately 3.5-b.y. lifetime, the Earth's biosphere seems to have maintained an optimal, evolving state of health. Health could be defined as the state of dynamic equilibrium between the organism and its environment, which is a definition that could be appropriately applied to a successful biosphere-oriented lunar habitat and its inhabitants. Designing, constructing, and maintaining a healthy habitat that can supply itself with life essentials such as food, water, and air is a challenging task akin to raising a child that will someday walk out the door and live its own life. The process must grow and evolve just as the young child or the young lunar base. The habitat should be designed to accommodate growth from the very beginning.

There are many artificial biosphere concepts that could be integrated in even the earliest stages of a lunar base. These include everything from applicable wilderness survival techniques (*Dowling et al.,* 1988) to the current space station reference configuration Environmental Control and Life Support System (ECLSS) featuring water reclamation and oxygen recovery subsystems (*Hypes,* 1988). For example, the Phytofarm hydroponic process makes efficient use of water by recirculating the nutrient solution—nitrogen, phosphorus, potassium—and using just enough for the plants to absorb their fill (*Field,* 1988). By controlling the environmental conditions of temperature, humidity, and light by computer as well as the input of carbon dioxide stored in tanks, an optimum rate of photosynthesis could be achieved in a similar hydroponic system on the Moon.

The Role of Proper Plant Combinations

Simple algae and bacterial colonies could be included for immediate operation in a first-stage lunar base to supply certain functions such as waste-recycling, gas exchange, food production, and eventually fertilizer. These algae and bacterial elements could be employed in the agricultural system and their by-products stored for later use. Empty descent propellant tanks are suitable for storage or cultivation tanks providing they can be completely cleaned of harmful residue. Small greenhouses and water tanks could also be made from simple, low-pressure inflatable structures of plastics or thin foil material.

One resourceful aspect of maximizing biospheric processes in space development concerns the notion of transportability. Lifting heavy materials into orbit is a costly and energy intensive process. However, many useful materials and functions can be generated by carrying just a mere seed or a single cell into orbit. One example is bamboo, which has been used for centuries as a fundamental building component on Earth for both housing and furniture. Bamboo could also be grown and used on the Moon, eliminating the high cost of transporting certain basic building materials. One can cheaply transport bamboo seeds to the Moon,

cultivate them in lunar soil, and then use the mature plants to construct everything from furniture to equipment racks.

Other key ingredients to this synergetic combination of plant materials beside algae and bamboo would be the foodstuff plants such as spinach, lettuce, corn, and beans. Under artificial conditions, spinach, for example, can be germinated in just one day as compared to the standard eight days. Under the same conditions, lettuce can grow from seed to full head in 26 days (the best lettuce can do outdoors in soil is 42 to 60 days). At normal atmospheric CO_2 levels (340 mg L-1), plants like corn most efficiently fix CO_2, while at elevated CO_2 levels (1200 mg L-1), plants like beans would have the advantage in CO_2 fixation due to reduced carbon loss via photorespiration (*Beer,* 1986). Soybeans also offer an optimal protein source since the biomass and volume required to raise beef or foul is unquestionable in the near-term lunar base.

Just as a plant is a system that processes earth and atmosphere, so can plants under the right artifical conditions process properly conditioned lunar regolith or nutrient-enhanced water, generating many useful products and by-products with a high degree of recyclability. It is foreseeable that the obvious improvements in technology and cultivation techniques can make the use of bioregenerative systems competitive with other life support systems, especially with increased numbers of crew and durations of stay at a lunar base.

Architectural Components

As the design for the proposed subselene facility matures, so do certain concepts currently in development. The Biosphere II Project in Arizona is a private project under development by Space Biospheres Ventures exploring the issues of artificial biospheres. This 98,000-sq-ft structure is designed to be a materially closed and energetically and informationally open system capable of supporting six to eight people (*Hawes et al.,* 1988). The research being done at Biosphere II includes many applications to lunar base design and to the proposed subselene scenario.

One of these applications is the "lung" or pressure/volume compensating chamber. Since the subsurface lunar facility will be completely sealed, an expandable lung device accommodates changes in the facility's internal air volume and eliminates pressure fluctuations that could break the seals and leak the valuable atmospheric elements. The variable volume chamber expands and contracts with shifts in the internal atmospheric volume-caused changes in temperature and pressure.

Achieving a total seal between the subselene structure's internal bladder and the access nodes is critical. The sealing techniques currently being researched must withstand temperature variations, and terrestrial ones are not as extreme as the temperature variations structures on the lunar surface will be subject to.

Advanced sensor technology and artificial intelligence systems will control such environmental parameters as light intensity, temperature, soil moisture, relative humidity, CO_2 and O_2 concentrations. Certain indicator plants can also assist in monitoring such key health vectors as pH and trace contaminants.

The use of a tiered facility structure could also assist natural convection currents to move moisture and temperature to and from designated interior spaces. A multimodule facility can be designed with each module at a different level so that warm, moist air flows toward the agricultural module and cool, dry air flows toward the module containing heat-producing equipment. The chambers could optionally be sealed off from each other and operate as temporary independent systems in emergencies.

The beneficial use of solar radiation is very important from the standpoint of artificial biosphere technology and subselene architecture. If the lunar facility is located near the lunar pole, solar radiation may prove to be near constant and harnessable with one or two strategically placed collector towers (*Dowling et al.,* 1988). If not, the 14 day/14 night cycles on the Moon make solar radiation impractical as an energy source unless an alternative system such as double lunetta system is used (*Ehricke,* 1980). These large, orbiting reflectors may be positioned in orbit so that sunlight is continuously reflected onto the area surrounding the lunar base (Fig. 8). The sunlight can then be harnessed in a variety of means, including solar collectors, and brought down inside the subselene facility with fiber optics.

Fiber optics reduce the need for transporting lamp systems and decrease the power requirement. Proper adjustment of a fresnel lens onto a solar optic collector can filter out harmful ultraviolet and photosynthetically ineffective infrared rays. Such solar optic systems with a lunetta tracking system could provide the lunar habitat with the life-giving properties of light minus the harmful ultraviolet elements. The sunlight could be brought from above ground and down into the subsurface facility. The application of solar optics to CELSS and subsurface lunar habitats are many. They include (1) cultivating algae-like chlorella in sealed tanks as a promising source of food and gas exchange, (2) intensive horticulture and aquaculture processes, (3) purifying and recycling human waste, and (4) satisfying the need humans have for sunlight both physically and psychologically (*Mori,* 1988).

SUMMARY REMARKS

If humans are to someday become self-sufficient on the lunar surface, the planning, design, and implementation of the life-supporting habitat must begin in the first stages. As the viability of a completely regenerative system has yet to be proven, the design should implement the currently practical aspects of artificial biospheres with a plan for expansion.

The proposed subselene tunneling system is able to create areas safe from solar radiation and extreme temperature variations.

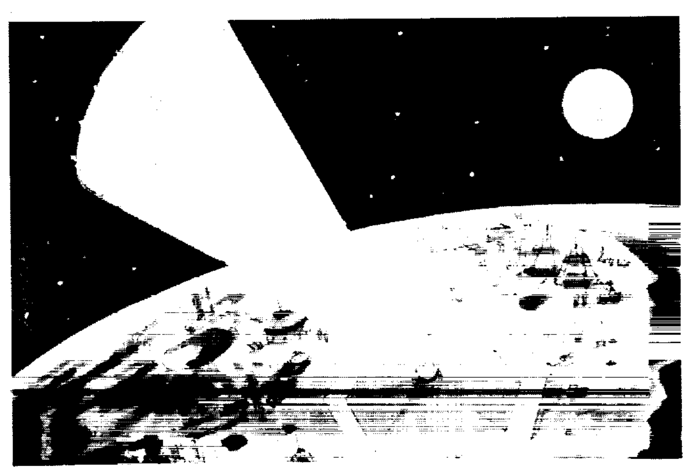

Fig. 8. Lunetta reflecting sunlight into subselene fiber optics system.

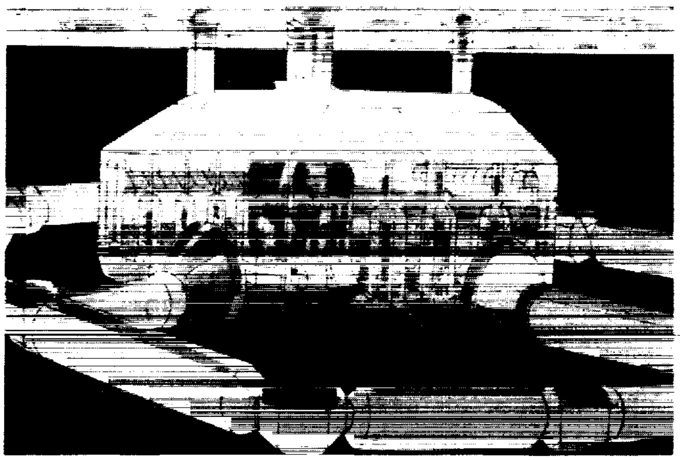

Fig. 9. A subselene growing chamber of an advanced lunar facility.

Augmented with artificial biosphere concepts, these subselene modules can become more habitable, productive, and continually less dependent on resupply as progress is made. Many of these subselene module cavities can work together to create large pressurized volumes, which are ultimately necessary for a more self-sufficient, artificial biosphere (Fig. 9). The objective should be to design a system that will be functional in the short term while maintaining the flexibility to evolve into a mature, independent lunar facility.

REFERENCES

Allen J. and Nelson M. (1986) *Space Biospheres*. Orbit Book Company, Malabar. 89 pp.

Beer S. (1986) The fixation of inorganic carbon in plant cells. In *Carbon Dioxide Enrichment of Greenhouse Crops, Vol. 2*, pp. 21-23. CRC Press, Boca Raton.

Dowling R., Staehle R. L., and Svitek T. (1988) A lunar polar expedition (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 73. Lunar and Planetary Institute, Houston.

Ehricke K. A. (1980) Lunetta system analysis. *XXXI Int. Astronaut. Congr.*, Paper No. 80-A-11.

Field R. (1988) Old MacDonald Has a Factory. *Discover*, December, p. 46.

Goodyear Aerospace Corporation (1982) *Innovative Structures for Space Application*, pp. 20, 22, 28. Goodyear Aerospace Corporation, Akron.

Hawes P. B. and Nelson M. (1988) Life systems for the lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 109. Lunar and Planetary Institute, Houston.

Hypes W. D. and Hall J. B. Jr. (1988) The ECLS system for a lunar base (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century*, p. 121. Lunar and Planetary Institute, Houston.

Kaplicky J. and Nixon D. (1985) A surface-assembled superstructure envelope system to support regolith mass-shielding for an initial-operational-capability lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 375-380. Lunar and Planetary Institute, Houston.

Mori K. (1988) *Himawari*, p. 246. La Foret Engineering & Information Service Co., Ltd., Minato-ku, Tokyo.

Rowley J. C. and Neudecker J. W. (1985) In-situ rock melting applied to lunar base construction. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 465-478. Lunar and Planetary Institute, Houston.

EXTRATERRESTRIAL APPLICATIONS OF SOLAR OPTICS FOR INTERIOR ILLUMINATION

N93-17449

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Solar optics is a terrestrial technology that has potential extraterrestrial applications. Active solar optics (ASO) and passive solar optics (PSO) are two approaches to the transmission of sunlight to remote interior spaces. Active solar optics is most appropriate for task illumination, while PSO is most appropriate for general illumination. Research into solar optics, motivated by energy conservation, has produced lightweight and low-cost materials, products that have applications to NASA's Controlled Ecological Life Support System (CELSS) program and its lunar base studies. Specifically, prism light guides have great potential in these contexts. Several applications of solar optics to lunar base concepts are illustrated.

INTRODUCTION

The purpose of solar optic (SO) system design is to enable use of the visible portion of the solar spectrum as a source of general or task illumination and thereby reduce dependence on electrically powered illumination. Passive solar optics (PSO) systems require less precision to design and build and will effectively deliver diffused light as well as beamed light. Active solar optics (ASO) systems effectively deliver beamed light only and need smaller collecting areas but require control systems and a power source.

A sustained effort to develop beamed sunlighting technologies began in 1978 during the "energy crisis." Solar optic technologies were initially developed for concentrating beamed sunlight into remote interiors of Earth-sheltered buildings. The impetus for this research and development, the need for energy conservation, and the applied criteria (lightweight and low-cost materials that minimize volumeric requirements) parallel those of NASA's Controlled Ecological Life Support System (CELSS) program and lunar base applications.

Critical design considerations throughout the development of PSO and ASO systems have included energy conservation, the use of lens designs that do not produce life or safety hazards, ease of construction, weatherability, and optical coatings for selective transmissivity. Wherever possible, nonimaging optics are used in PSO designs. When imaging optics are used, the possibility of conflagration is eliminated through the use of selective filters or containment of focal areas within protective enclosures. Throughout the development of ASO systems, particular attention has been paid to alternatives for heliostat control systems, artificial light sources with integrated controls, and strategies that reduce the volume of optical material required to transport light. Terrestrial and extraterrestrial criteria will no doubt be different; however, many of the lessons learned in developing terrestrial applications are directly transferable.

PASSIVE SOLAR OPTICS

The PSO system is a form of fenestration control that reduces the ratio of aperture size to lighted area in comparison to conventional natural lighting strategies (*Eijadi,* 1983). The system enables sizing and designing an aperture based on the desired quantity and directionality of the available sunlight. There are two basic types of PSO systems: refractive systems (Fig. 1) and

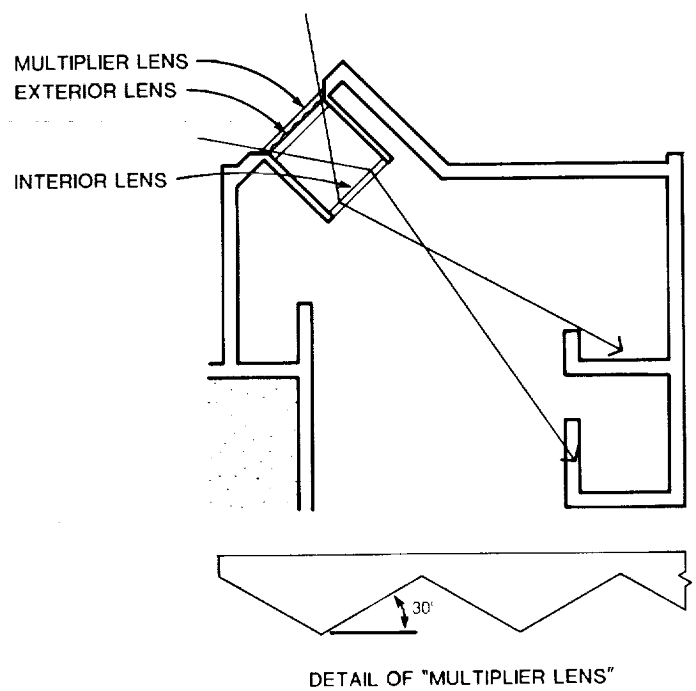

Fig. 1. Refractive PSO system.

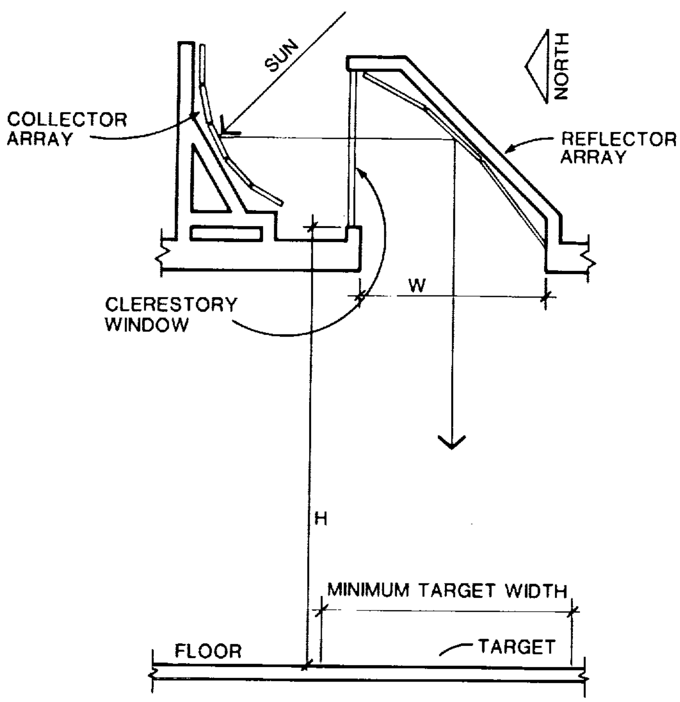

Fig. 2. Reflective PSO system.

reflective systems (Fig. 2). Each system is described in terms of terrestrial applications; however, the same design process applies to extraterrestrial applications. Consideration of filtering undesired radiation and of physically protecting the exterior components needs to be addressed. A meteorite shield with a selective radiation coating similar to that proposed for protecting the heliostat of the CELSS module is worth consideration (*Oleson and Olson,* 1986).

REFRACTIVE PASSIVE SOLAR OPTIC SYSTEM

A refractive PSO system consists of three major elements: (1) interior Fresnel lens; (2) exterior Fresnel lens; and (3) prismatic multiplier lens. The interior and exterior lenses are spaced approximately one focal length apart in an axial arrangement. The multiplier lens is placed ahead of the exterior lens in order to increase the cone of vision in one or both axes altitude and azimuth. Without the multiplier, the limitations of focal length and f-stop generally result in a solid angle cone of view of 32°. The multiplier can double the cone of vision but at the expense of total system efficiency, approximately 12.5% and 6.25%, depending on increasing the cone of view on one or both axes, respectively. The refractive PSO system exhibits some chromatic aberrations. It should be installed at an angle to the ground equal to the latitude of the site for best performance.

REFLECTIVE PASSIVE SOLAR OPTIC SYSTEM

Current installations of the reflective PSO system consist of three major elements: (1) collector array; (2) clerestory window; and (3) reflector array. The collector array faces the sun and reflects sunlight through the clerestory window to the reflector array. The reflector opposes the collector array and redirects the sunlight to the desired target area. Sunlight is diffused approximately 10° when reflected from either array. The pattern of the Fresnel lenses of the reflector array is turned 90° to that of the collector array so that light is diffused in each direction, resulting in a uniform distribution of light at the target area. An approximate system efficiency, the ratio of illumination reaching the target to the total available on a horizontal surface, can be assumed to be 10% for preliminary design purposes. Empirical testing and a mathematical model were used to estimate overall efficiency (*Eijadi,* 1983). The effects of dirt, light scattering, diffusion, and absorption in the acrylic contribute to the depreciation of illumination as light is reflected through the arrays. Passive solar optics systems require a separate electrical lighting backup system.

The determination of the angles of repose for each section of the arrays is a function of the latitude the system is designed for and the distance from the system to the target. The topmost panel is aimed at a low horizon. For Earth-based systems, each successively abutting collector panel is aimed higher until all the useful or desired annual solar horizons are within one or more regions of the array. The angles of the reflector array are designed to redirect the light from the collector to the target area.

ACTIVE SOLAR OPTICS

Active solar optics systems are distinguished from PSO systems in that they have a component, a heliostat, that mechanically tracks the solar disk. It is also possible to physically integrate the electrical backup illumination system within the sunlight distribution network. The four components of an ASO system are (1) the heliostat; (2) intermediate transport networks; (3) artificial light sources with controls; and (4) a distribution system (Fig. 3).

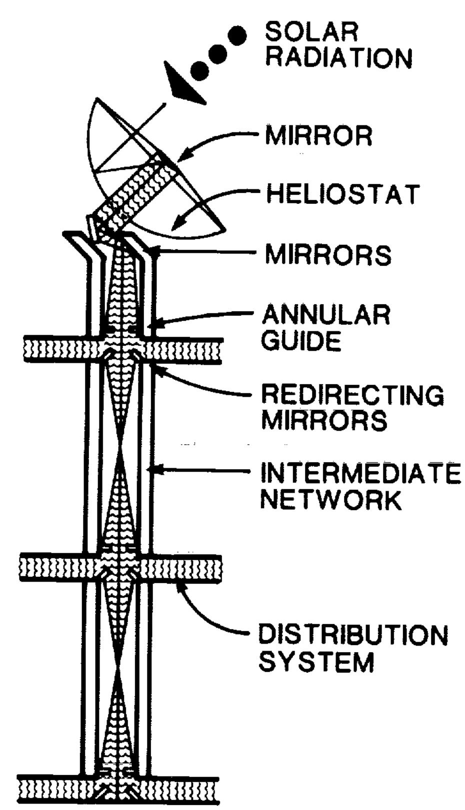

Fig. 3. Active solar optic system.

Simply put, the heliostat tracks the sun and positions a series of reflectors that beam the sunlight into the intermediate transport network. The intermediate network transports the light horizontally or vertically to the delivery device, which illuminates the occupied space.

Various materials were investigated for use in the system (*Eijadi et al.,* 1987). Intermediate network materials included fiber optic cables, reflective pipes, holographic pipes, solid-angle lens guides, and prism light guides (PLG). Distribution devices evaluated included fiber optic cables, diffusing reflectors, specular reflectors, and PLG. Each component was evaluated on the basis of performance, cost, constructibility, and ease of integration with conventional construction practices.

It was concluded that intermediate networks were decidedly best with the least amount of physical material incorporated into their design. They should be dedicated, airtight passageways that are as short as possible. Depending on precise distances, PLG and reflective pipes worked nearly as well. Prism light guides were the preferred choice for the distribution device. Artificial light sources should be high-intensity discharge (HID) sources located as close to the distribution device as possible because the energy for that resource is paid for and should not be subjected to any unnecessary losses prior to utilization. The distribution device (light fixture) should be linear and oriented downward to maximize distribution and minimize room losses associated with absorption and maintenance.

A proof-of-principle model was constructed using the heliostat at the University of Minnesota's Civil and Mineral Engineering building. Sunlight and light from a metal halide source were alternately introduced into the same PLG and produced identical distributions with nearly identical efficiencies. A complete system designed, fabricated, and installed by Whitehead is presently in operation in Toronto, Canada.

Several components of ASO systems have been reduced to common practice. Among those are heliostats, vertical and horizontal distribution networks, and fixtures capable of delivering beamed sunlight and/or electric light. A variety of selective coatings and control systems for heliostats are offered in the marketplace.

PROPERTIES OF MATERIALS USED FOR SOLAR OPTICS

The unique aspect of the work on ASO systems presented in this paper relates to the use of the prism light guides for the transportation and distribution of both sunlight and artificial light. The light guides are hollow tubes made with either an optical grade polycarbonate or acrylic polymer film. The films are called "Scotchlamp Film" by their manufacturer, 3M. The acrylic film is more stable than the polycarbonate, but the polycarbonate can resist higher temperatures, 248°F vs. 190°F. Each film can be made at varying thicknesses but typically is 0.022" (0.56 mm) and weighs about 0.13 lb/ft (0.064 kg/m). The surface of the film is formed into nearly microscopic prismatic facets that transmit light using the principle of total internal reflection (*Saxe et al.,* 1986).

The film used in the PSO system is an optical-grade acrylic similar to that used in the ASO system. It is approximately 0.022" thick (0.56 mm) and comes in widths up to 24" (61 cm). The walls of the film are formed into grooves so that sunlight is reflected with a diffusion of approximately 10%. An aluminum backing is added to the film.

Testing has been performed on similar films exposed to the low-Earth-orbit (LEO) environment (A. Zderad, personal communication, 1988). With the current design, the films degrade in the presence of or impact with monatomic oxygen in LEO. No testing has been performed in deep space. A further investigation of the thermal and ionic space environment in relation to these films is needed in conjunction with a rethinking of the manufacturing process to determine if direct exposure is feasible.

If the films are used in a controlled environment such as the interior of the space station or a lunar base, degradation of the films should not be a problem. Hazards associated with outgassing or fire should be no greater than with any other synthetic materials used.

APPLICATIONS OF SOLAR OPTICS TO SPACE STATION *FREEDOM*

A conceptual design study clearly identified the need to supplement artificial lighting with solar illumination to grow plants in the CELSS module (*Oleson and Olson,* 1986). In fact, an all-solar illumination system was preferred, based on a parametric study of electric power, volume, cost, and mass. Unknown plant growth behavior with short illumination cycles prompted the investigation of two hybrid systems, one using a combination of fiber optic cables and fluorescent lamps and the other using fiber optic cables integrated with a remote HID light source.

The hybrid system using fluorescent lamps was identified as the best choice. In this system, a heliostat with an array of 2712 Fresnel lenses concentrates light on 2712 glass fibers that transmit light to the plant-growth units. Solar illumination is utilized during the 60 minutes of available sunlight, and the fluorescent lamps are used during the 30 minutes of darkness each orbit. The fluorescent lamps, adjacent to the plant trays, would provide 750 fc (8070 lux) to the plants and the solar lighting would provide 7500 fc (80,700 lux).

Concerns in using the fluorescent/fiber optic system were identified: (1) mercury in the fluorescent lamps is a health hazard; (2) the fluorescent lamps will have a shorter life span than HID lamps; (3) fluorescent lamp replacement will be more difficult than with a remote HID lamp; and (4) fluorescent lamps cannot be closely spaced because of mutual interference. Each of these concerns was deemed manageable.

The other hybrid system considered in the study utilized HID lights and fiber optic cables. Solar illumination would be transmitted to the plants as before, but a remote HID light source would transmit light via fiber optic cables that would, in turn, be integrated with the cables coming from the heliostat.

Concerns about the HID/fiber optic system include (1) the integration of an HID light source and fiber optic cables is an unknown technology, and significant losses of efficiency were assumed at the interface of the two; (2) higher costs were associated with the HID/fiber optic system, and these costs are directly attributable to the development and design work required by an unknown technology; (3) HID lamps require preheating; and (4) this configuration represented the greatest mass of all the systems evaluated because of additional fiber optic cable required.

The HID/fiber optic system was identified as having the advantages of better maintenance, safety, accessibility, centralized cooling, and lamp efficiency when compared to the fluorescent system.

While two artificial lighting systems were evaluated, only one solar transmission system was evaluated. The use of prism light guides rather than fiber optic cables may offer some opportunities to improve the cost and mass characteristics of the hybrid HID system and utilize the inherent advantages of a remote HID light source. The reasons to consider using prism light guides for this application are (1) The integration of HID lamps and PLG is a known technology, unlike the integration of HID lamps and fiber optics, so a reduction of development costs can be anticipated; (2) PLG can transmit the same amount of light with less optic material, so that assuming the same volumes as the preliminary design, order-of-magnitude calculations reveal that substituting PLG for the fiber optic cables will reduce the mass for the light transfer device by a factor of 50; (3) There exists the opportunity to eliminate the physical connection between the heliostat and CELSS module if the heliostat is mounted on the space station structure for better solar access, and a lens with the proper focal length could concentrate the sun's rays to a porthole on the module; and (4) The system incorporating PLG can filter harmful UV and IR radiation in a similar fashion as fiber optics, which is important when plant growth is concerned (*Saxe et al.,* 1986).

THE APPLICATION OF SOLAR OPTICS TO THE LUNAR BASE

The initial advantage of using ASO for the outpost is the conservation of available space. Longer-range advantages of using SO on the Moon are threefold. First, many concepts for lunar outposts encapsulate habitable areas with regolith or natural formations for protection from harmful radiation. Solar optics provides a way to transmit sunlight to these shielded environments. Second, SO systems can filter out undesired radiation wavelengths, thereby transmitting the desired visible spectrum for interior illumination. Third, the technology is very simple: The heliostat is the only mechanical device used.

The disadvantages of using SO are, first, the need for solar access and, second, the potential loss in efficiency if dust proves to be excessive from lunar operations. Lack of solar access during the lunar night may be mitigated if available luminance from the Earth can be utilized (*Ebricke,* 1985).

The location of the first lunar site has not been determined. Given this fact we made the assumption that the use of solar energy is desirable wherever the lunar base is located, either as a primary source of power and interior illumination at the lunar poles, or as a secondary source at the lower latitudes of the Moon.

The most promising use of SO would be if a polar location were selected as the initial site for lunar habitation. Because the Moon's equatorial plane is only inclined 1½° to the ecliptic, there is a possibility for continuous sunlight availability (*Burke,* 1985). At the lower latitudes, there will be two-week days and two-week nights. In either location, the design of the PSO and ASO systems on the Moon will be modified from Earth-based systems. The small tilt of the lunar axis enables the PSO design emphasis to be placed on tracking the horizontal movement of the sun during the lunar day rather than the vertical, seasonal movement on Earth. Similarly, the ASO design can be simplified by tracking the sun on one axis rather than two.

We have taken several lunar base schemes and speculated on how PSO and ASO might be applied. The schematic diagrams illustrated here represent the stages of lunar development identified in previous studies (*Duke et al.,* 1985). The diagrams assume a lunar pole location but can be easily modified for any latitude.

Current lighting strategies for the space station habitat and laboratory modules favor the use of all-artificial sources (C. Wheelwright, personal communication, 1988). An opportunity to utilize ASO exists, however, if the habitat modules are used for the first lunar outposts. Figure 4 shows such an outpost (*Kaplicky and Nixon,* 1985). A preliminary lighting evaluation for the space station identified several interesting concepts (*Walter, Dorwin, Teague Assoc.,* 1987). One concept is to integrate the air plenum, fluorescent lamp, and light diffuser. Removing the fluorescent lamp, the plenum space could be lined with PLG. A heliostat, supplemented with an HID lamp, could then pipe light down the plenum to provide interior illumination. This concept is similar to the one proposed for the CELSS module. As the base incorporates more modules, the ASO system can grow with it.

The scheme shown in Fig. 5 shows housing an operational or advanced base inside a lava tube (*Hörz,* 1985). This scheme is similar to current terrestrial applications in office buildings. Passive solar optics is used to provide general illumination within the lava tube and ASO is used to provide habitat lighting.

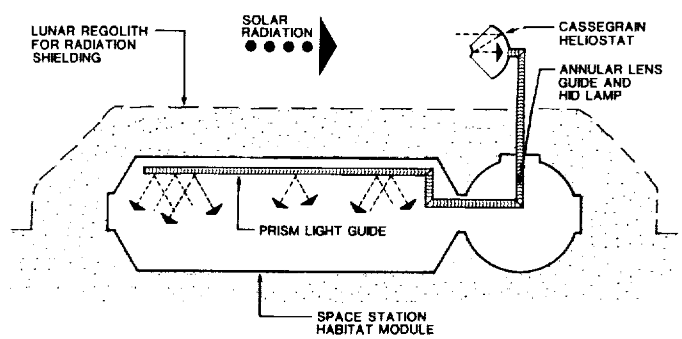

Fig. 4. Lunar outpost using ASO.

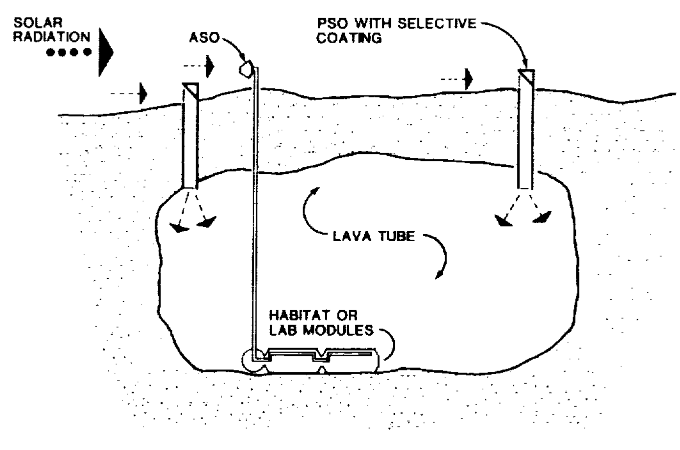

Fig. 5. Operational base using ASO and PSO.

The scheme shown in Fig. 6 is for a self-sufficient colony. The scale of the SO systems is limited only by the available volume. As the lunar community grows, the illumination scheme can grow with it. The diagram shows PSO being used as general illumination and gives a sense of orientation to the colony. Active solar optics is used as building-specific illumination, with light being transmitted both horizontally and vertically.

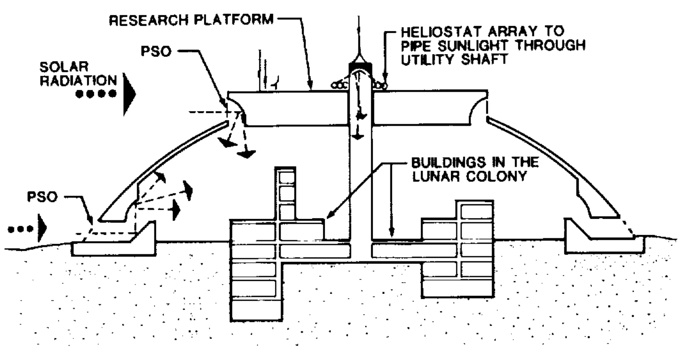

Fig. 6. Self-sufficient colony using ASO and PSO.

CONCLUSION

The SO systems described in this paper are known technologies that have been proven to deliver the quantity and quality of light needed for various human activities on Earth. Active solar optics is most appropriate for general and task illumination, and PSO is most appropriate for general illumination only. Both systems can filter harmful radiation. It is concluded that these same technologies should be considered for use with the space station CELSS module and for the various development phases of the lunar base.

To determine the feasibility of applying SO to extraterrestrial applications, and in particular lunar bases, further investigation as to the effect of the thermal and ionic environment and of lunar dust on the SO system must be undertaken.

REFERENCES

Burke J. (1985) Merits of a lunar polar base location. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 77-84. Lunar and Planetary Institute, Houston.

Duke M., Mendell W., and Roberts B. (1985) Strategies for a permanent lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 57-68. Lunar and Planetary Institute, Houston.

Ehricke K. (1985) Lunar industrialization and settlement—birth of a polyglobal civilization. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 827-855. Lunar and Planetary Institute, Houston.

Eijadi D. (1983) The Civil Mineral Engineering building solar optic systems. In *Papers Presented at the Sixth Miami International Conference on Alternate Energy* (T. N. Beziroglu, ed.), pp. 600-603. Clean Energy Research Institute, Miami.

Eijadi D., Probst D., Shadid W., De Los Rios M., and Bennett D. (1987) *Sunlight Distribution in Remote Interior Locations Using Multi-Axis Suntracker as Light Source.* Contract De-AC02-83ER-80047, U.S. Department of Energy. 125 pp.

Hörz F. (1985) Lava tubes: Potential shelters for habitats. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 405-412. Lunar and Planetary Institute, Houston.

Kaplicky J. and Nixon D. (1985) A surface-assembled superstructure envelope system to support regolith mass-shielding for an initial-operational-capability lunar base. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 375-380. Lunar and Planetary Institute, Houston.

Oleson M. and Olson R. L. (1986) *Controlled Ecological Life Support Systems (CELSS): Conceptual Design Option Study.* NASA Contractor Report 177421, NASA Ames Research Center, Moffett Field. 144 pp.

Saxe S., Whitehead L., and Cobb S. (1986) Progress in the development of prism light guides. *Proceedings of the Society of Photo-optical Illuminating Engineers.*

Walter, Dorwin, Teague Assoc. (1987) *Space Station Lighting Evaluation.* Study prepared for Boeing Aerospace Co. 33 pp.

4 / *Scientific Investigations at a Lunar Base*

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THE ROLES OF HUMANS AND ROBOTS AS FIELD GEOLOGISTS ON THE MOON

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The geologic exploration of the Moon will be one of the primary scientific functions of any lunar base program. Geologic reconnaissance, the broad-scale characterization of processes and regions, is an ongoing effort that has already started and will continue after base establishment. Such reconnaissance is best done by remote sensing from lunar orbit and simple, automated, sample return missions of the Soviet Luna class. Field study, in contrast, requires intensive work capabilities and the guiding influence of human intelligence. We suggest that the most effective way to accomplish the goals of geologic field study on the Moon is through the use of teleoperated robots, under the direct control of a human geologist who remains at the lunar base, or possibly on Earth. These robots would have a global traverse range, could possess sensory abilities optimized for geologic field work, and would accomplish surface exploration goals without the safety and life support concerns attendant with the use of human geologists on the Moon. By developing the capability to explore any point on the Moon immediately after base establishment, the use of such teleoperated, robotic field geologists makes the single-site lunar base into a "global" base from the viewpoint of geologic exploration.

INTRODUCTION

Geoscience will be one of the prime scientific activities associated with a permanently staffed lunar base. The geologic exploration of the Moon is an ongoing task occurring before, during, and after base establishment. Various methods and techniques of geologic investigation exist that serve a variety of purposes; these different methods involve differing hardware, operational, and interpretive approaches. In this paper, we first distinguish between the two different types of geological investigation and the philosophies and operational methods behind them. We then consider how the goals of advanced, detailed geologic study conducted from the lunar base may be best accomplished, specifically by examining the relative roles of humans and robots as lunar field geologists. Our purpose is not to provide a detailed plan for the exploration of the Moon, but to examine the relative merits of two different approaches to lunar field geology.

TYPES OF GEOLOGIC FIELD WORK

Geology is the science concerned with the origin, history, and evolution of terrestrial planetary bodies. To decipher and understand the record of planetary evolution retained in its rocks, it is necessary to examine and study rocks in their natural environment (for a detailed discussion of the methodology and philosophy of geology, see *Albritton,* 1963); in geology, this technique is termed *field work.* Geologic field work on Earth has a long and venerable history, and the techniques for lunar geologic field work were adapted from terrestrial experience for the Apollo lunar missions with only minor modifications (*Hess,* 1967; for a summary of the current status of lunar geological problems, see *Lunar Geoscience Working Group,* 1986).

For the purposes of this discussion, we subdivide geologic field work into two broad categories: *reconnaissance* and *field study.* The goals of geologic reconnaissance are to provide an admittedly incomplete, but broad characterization of the geologic features and processes on a planetary body. The questions asked during the reconnaissance phase are of first-order and fundamental importance. For example, one may identify the most sparsely cratered, dark flow on the Moon from orbital photographs; the geologic interpretation of such a feature would be that it represents the youngest lunar lava flow (an important datum for understanding lunar thermal history). An example of geologic reconnaissance would be a simple sample return mission (e.g., Soviet Luna class; see *Johnson,* 1979) to provide bits of the lava flow that could then be dated by radiometric techniques. Such a mission has relatively simple, focused objectives: Sample the flow to determine its age and composition. More detailed questions, such as the petrogenesis of the basaltic magma and the flow's relation to overall lunar volcanic history, can be tentatively addressed, but such a mission is not designed to answer these questions. This type of preliminary exploration paves the way for the more detailed type of study to follow.

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Geologic field study, as here defined, has more ambitious goals. The objective of field study is nothing less than to understand planetary geologic processes and units at all levels of detail. Such a goal makes it a virtual certainty that field study is a protracted and complex operation; moreover, field study is an iterative process involving repeated visits to the same field site interspersed with analytical laboratory work and revision of the working hypotheses. The operational methods developed for reconnaissance are inadequate at this level of study. Not only must a field study site be sampled at increasing levels of detail, but one does not know in advance which recognizable subunits may hold the answers to a given series of questions. Autonomous, automated machines are incapable of the decision-making necessary at this level of study; human intelligence and interaction during the field work is an absolute necessity.

These two methods of geologic study are both necessary; we do not begin a detailed field study of a given region unless we know what questions are appropriate to ask. Conversely, no single set of reconnaissance results gives us a really complete understanding of the history and evolution of a region or process. Thus, both types of investigation proceed simultaneously and both will be essential in conjunction with lunar base establishment and operations.

GEOLOGIC RECONNAISSANCE AND THE LUNAR BASE

A cornerstone in the geologic reconnaissance of any planetary object is the acquisition of global remote-sensing data; this includes determining the morphology, the chemistry and mineralogy, and the physical characteristics of surface and subsurface units. Prior to the establishment of a lunar base, such a global database should be provided for the Moon by a polar-orbiting spacecraft; the proposed Lunar Observer (LO) mission goes a long way toward providing this information (*LGO Science Workshop Members,* 1986). The data produced by this mission should be used to plan a systematic sampling program using the automated sample return spacecraft described earlier. Such a series of sample returns can be planned for both scientific exploration and specific operational reconnaissance designed to support lunar base operations (*Ryder et al.,* 1989). Examples of the former include compositionally distinct mare basalt units, the impact-melt sheets of large complex craters (both to provide an estimate of the gross target composition and to give absolute ages of the impact events to calibrate the lunar geologic timescale), and regions of the highlands that appear from the orbital data to be geologically interesting. Examples of operational missions include the return of samples from potential ore deposits identified from orbital data and the examination of possible volatile-rich areas for base life support and propellant extraction.

Another class of reconnaissance mission involves the use of semiautonomous rovers. Such a spacecraft could traverse long distances on the Moon, performing chemical analyses of soils and mapping the mineralogy of rock exposures through multispectral mapping techniques. It could also provide detailed engineering data on lunar surface and subsurface conditions, including the identification of optimum mining prospects and the surface and subsurface characteristics of potential base outpost sites. Experience with the Soviet Lunakhod series (*Vinogradov,* 1971) suggests that the potential of such vehicles for the collection of both scientific and engineering data has yet to be fully realized. The use of rovers as base precursors could provide a very cost-effective means of gathering hard data for the planning of more complex surface operations in the future.

Geologic reconnaissance both precedes and follows base establishment. In the first case, it is by no means obvious that we will want to emplace the lunar base at a previous (Apollo) landing site; basic information about the geologic setting, resource potential, and physical nature of possible base sites must be reasonably well understood before base establishment. Geologic reconnaissance provides some of these basic data. In the second case, the ongoing geologic exploration of the Moon as a planetary body requires increasingly longer, more complex, and more detailed field work; such work cannot be planned and accomplished without precursor reconnaissance of geologically interesting regions. Expanding human presence on the Moon also requires that we eventually identify and characterize all available lunar resources for ultimate, if not immediate use. Thus, we believe that the capability to perform geologic reconnaissance before, during, and after base establishment is a required element of any lunar base infrastructure.

GEOLOGIC FIELD STUDY AND THE LUNAR BASE

To completely understand lunar evolution and history, geologists must conduct intensive field studies of promising areas on the Moon. In this phase of work, large- to small-scale processes and units are studied and the questions under investigation are likely to be layered with increasing levels of specificity and complexity. Examples of sites studied during this phase include the central peaks of large craters where complex outcrops occur, megablocks of brecciated highland crust that may occur both as ejecta and as exposures within crater walls, crater and basin ejecta deposits, and the genesis of lunar landforms such as sinuous rilles and wrinkle ridges. The methods of investigation for such targets differ greatly from those described above; a Luna-type sample return from any of these kinds of targets would probably create more confusion than enlightenment.

The key element necessary in these types of study is the guiding influence of human intelligence and experience. Moreover, the presence of the human intelligence must be of such a nature as to proceed interactively and simultaneously with the field work being performed. Given such a requirement, what techniques are best suited to accomplish scientific goals? For such complex surface operations, we envision two basic approaches: human field geologists and *teleoperated* (not automated) robots. The principles and techniques of human field work are well understood after 200 years of geologic investigation on the Earth; they may be applied to the Moon with only slight modification (*Schmitt,* 1973; *Spudis,* 1984).

The use of teleoperated robots as field geologists heretofore has not received detailed consideration, but robots have many potential advantages over humans. They can be made with sensory capabilities at any wavelength in the electromagnetic spectrum, which gives them a particular advantage over humans in the area of mineral and chemical identification while in the field. Robots can be made to possess great physical strength and endurance (useful in a field geology context to move boulders for sampling and to work for extended time periods). Possibly their most important advantage over human workers is their unique ability to work in the harsh lunar environment unencumbered by

complex and massive life support systems; moreover, serious safety issues arise with the consideration of extended human presence on the lunar surface, particularly in regard to radiation exposure and, to a lesser extent, micrometeorite impacts. Robotic field geologists can be designed so that these concerns are greatly alleviated.

As we envision their use, these cybernetic field geologists would perform tasks identical to their human counterparts. In terms of field geology, this involves recognizing distinct lithologies in the field, collecting both representative and unusual samples, and returning them to the lunar base for detailed analysis. During periods of intensive field study, the robots would be under the direct and complete control of a human geologist. The goal of this mode of operation is *telepresence*; i.e., to simulate reality for the human operator through the use of robotic teleoperations (*Wilson and MacDonald,* 1986; *Sheridan,* 1989). But where should these human operators be, on the Moon or on Earth? The round-trip radio time for lunar operations controlled by an operator on Earth is 2.6 sec, and this lag time between command and observation of command response might seriously degrade the telepresence effect. Do the geologist-operators really need to believe that they are at the field site? Is a near-instantaneous response necessary for sound field work? Or is telepresence a luxury?

The question seems to focus on the maximum time delay that can be tolerated without degrading the quality of the field study. Time delay might be a more tolerant criterion for geologic field work than it is for complex mechanical tasks such as construction. More research is needed to determine the allowable limits of time delay. Experiments can assess the possibility of operating robots on the Moon from Earth (2.6 sec) and of operating them on Mars from Earth (5 min to 40 min).

The most important factor in doing field work properly, besides the training, talent, and experience of the geologist, is the presence of human powers of thought and observation at the field site. It is not clear that this requires full telepresence. It sounds enticing to think of yourself as the operator, actually sensing that you are in the field. Nevertheless, *Wilson and MacDonald* (1986) point out that the most important factor from the standpoint of the operator is the intellectual challenge, in this case the challenge of unraveling some of the Moon's geologic history. However, we feel that the sense of discovery and the excitement that goes with it are also important. Telepresence may not be required for stimulating the operator's intellect or for generating the sense of excitement that goes with exploration. On the other hand, if remote operation becomes too cumbersome (for example, because the time delay is extreme) the operator will concentrate more on mechanical aspects of the work and less on the intellectual ones. After all, when doing field work on Earth, geologists do not need to think about focusing their eyes or moving along an outcrop. When they do, as when the outcrop is a cliff with a narrow ledge, geologists spend more time watching their steps than examining the outcrop.

If experiments show that high-quality field work can be done on the Moon (and perhaps Mars) by operators located on Earth, some interesting possibilities result. Most important is the active involvement of many more geologists than will be on the Moon during the first few decades of base operations. More areas could be studied, more samples could be returned, and more intellectual energy could be expended on solving problems in lunar and planetary science. Graduate students, some of whom might someday do field work in person on the Moon or Mars, could be trained in extraterrestrial field work. A major advantage of this is that many important geological discoveries have been made by students doing field work for their master's or doctoral theses. We could expect the same on the Moon and Mars.

CYBERNETIC LUNAR FIELD GEOLOGIST: A DESIGN CONCEPT

Attempting to predict the state of the art in robotics technology in the next century is futile. Nevertheless, we can identify the likely requirements and capabilities of a teleoperated robot designed for geologic field work. We offer the following design concept for a machine to geologically explore the Moon (Table 1, Fig. 1).

One of the prime requirements for such a robot is mobility. The Apollo Lunar Roving Vehicle (LRV) performed splendidly and reliably on three separate Apollo missions (*Morea,* 1988); it was a wheeled vehicle powered by four independently operated electric motors that outperformed its design specifications on the Moon. Although we have no particular prejudices regarding the type of motive system used, we have chosen to base our concept on a wheeled, roving vehicle. It is possible that some type of walking vehicle (e.g., *Brazell et al.,* 1988) or tracked vehicle may be ultimately preferable to a wheeled one.

The instrumentation advocated for this robot (Table 1) not only meets our criteria for telepresence, but it is optimized for additional sensory capabilities appropriate for geologic field work. In this regard, we are interested in the near- and far-infrared portions of the spectrum, where characteristic absorption bands of the common rock-forming minerals occur, and in the X-ray and gamma-ray bands, which contain lines related to elemental abundance. Real-time identification of rock types in the field will be greatly aided by such instrumentation. We envision that during teleoperations, a selected subset of this mineralogical and chemical data would be image-superimposed on the high-resolution, real-time television display; this mode of operation would be selected by the operating geologist. When lithologic differences are recognized, a reversion to normal vision may be desirable for the next steps.

Visual recognition of rock types in the field is followed by systematic and representative sampling of the desired units. We envision at least two robotic arms will be necessary; these arms should possess some type of tactile feedback, as the touch sense is one that is commonly used in terrestrial field work (e.g., the friability of a breccia is an important piece of geologic information). The robotic arms could be fitted with a variety of end articulators designed to perform various functions. It is desirable for one arm to have an anthropomorphic hand for normal manual operations; the other arm could be used as a combination percussion hammer (the traditional tool of the field geologist) and a small drill core capable of boring and extracting specific portions of a complex rock. Polymict breccias on the Moon frequently contain numerous clasts, but usually a limited series of rocks of a given type; the most effective way to sample such a rock is to obtain a few of those clasts recognized as representative (determined from the sensory data described above), sample any clasts recognized as unusual, and return them all for detailed analysis. Collected samples would be documented and placed in sample return containers carried on the bed of the rover.

Additional articulators for the robot's arms could also serve useful functions. Studies of Apollo samples show that rake sample

TABLE 1. Specifications for a teleoperated robotic field geologist.

System	Instrument or device	Comments
Mobility	Roving vehicle	Range thousands of kilometers
Vision	Stereo, high-definition color television	Minimum resolving power 30″ of arc; telescope mode, 1″ of arc
Manipulation	Anthropomorphic arm(s) and hand(s) with tactile feedback	
	Percussion hammer and drill core arm	Capable of extraction of 2-cm-diameter rock core
Sample identification	Visual-infrared mapping spectrometer	0.3-20 μm; 1200 spectral channels
	X-ray fluorescence spectrometer	Real-time chemical analysis
Sample stowage	Four to five sample return containers	Each container with over 200 documented subcompartments

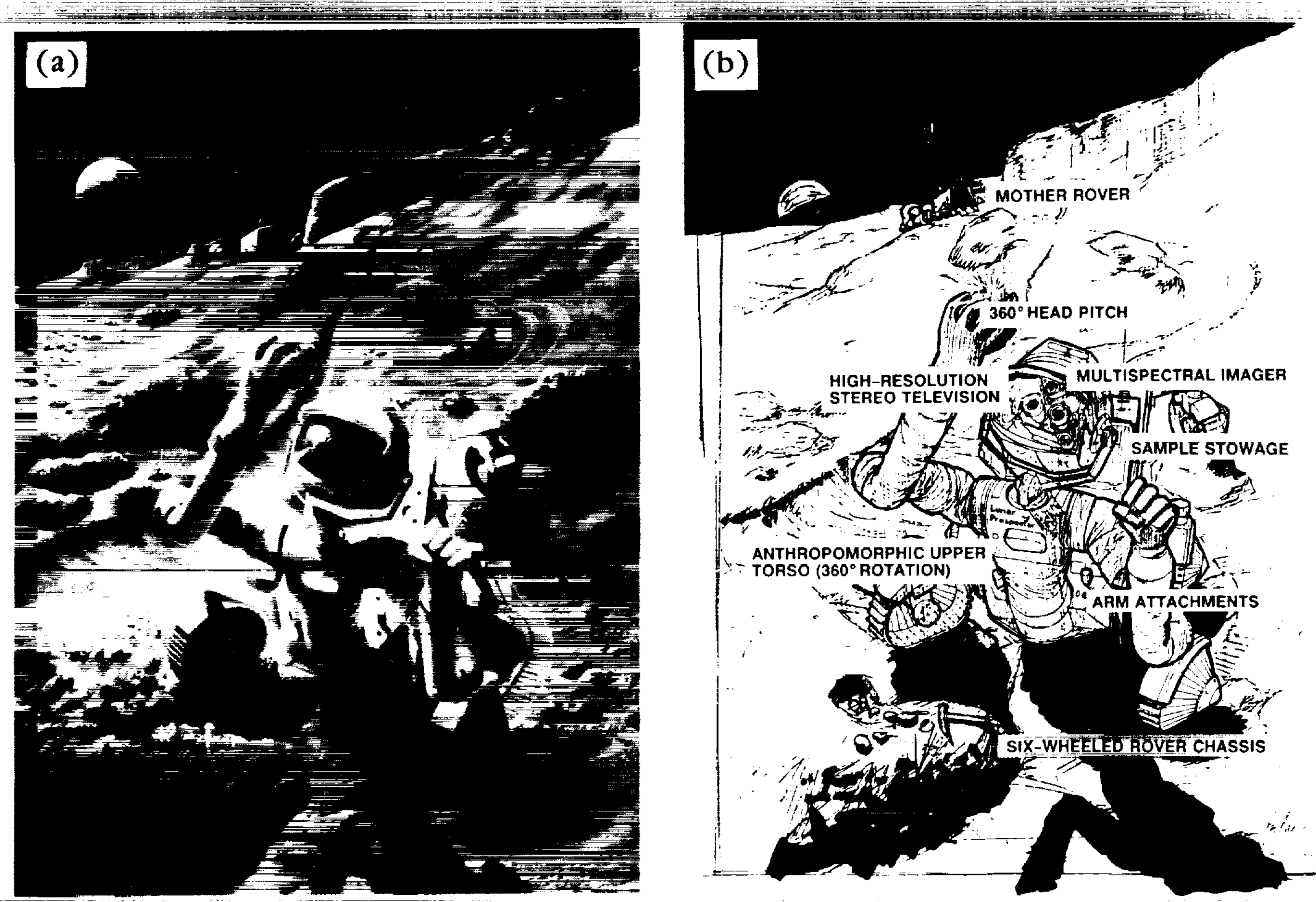

Fig. 1. **(a)** Artist's concept of a teleoperated robotic field geologist discovering a xenolith in a lunar mare basalt flow. Painting by Pat Rawlings. **(b)** Sketch of the robotic field geologist showing configuration of equipment. See Table 1 for instrument description and text for operational details.

collection, the gathering of a statistically representative sample of small, walnut-sized rocks, and regolith drill cores, down to depths of 2-3 m, are useful ways to sample the Moon. These sample collection functions require little active input from the teleoperator and could be automated.

Constant communication of the robot with the teleoperator is required. For operations on the lunar nearside controlled from Earth, direct and constant radio contact will be possible. However, for operations on the farside and for robot control by operators on the Moon, a series of comsats, either in halo orbits at the Lagrangian points or in lunar orbit, will be needed. In addition to communications, these comsats could also be the most effective way to perform lunar surface navigation for long-distance (hundreds of kilometer) traverses through radio positioning and orbital tracking. An alternative method of communication between the robot and lunar base operators might be to deploy line-of-sight relay stations along the traverse route. Although we have not considered this technique in detail, the abrupt curvature of the Moon (the horizon for a 2-m-tall viewer on a flat mare plain is about 2.6 km away) suggests that this might severely restrict the effective operational range of the robot. The use of lunar topography to site relay stations may partially alleviate the problem; however, for an extended geological traverse such as the one described by *Cintala et al.* (1985), the use of available topography in the Imbrium Basin region (average elevations between 3 km and 4 km) suggests that at least 10 relay stations would have to be employed (range between stations about 240 km) between the rover at maximum traverse range (about 2400 km) and the base control site. Moreover, this deployed relay net would then not necessarily be available for future use, as new traverses would probably strike out in different directions, requiring the deployment of yet another relay net. We feel that the use of a lunar comsat system would probably be the most efficient way to communicate with a long-range roving vehicle.

In addition to its field geologist role, our robotic bus could be easily adapted to perform other surface operational tasks. For example, the deployment of network equipment, such as geophysical stations, could be done efficiently by teleoperations. Moreover, it is also possible to combine two functions on a single traverse, with the robot deploying geophysical instrumentation on its outbound traverse and performing field geology during its return to base. Thus, this proposed robotic vehicle could be easily adapted to perform multiple functions during lunar base surface operations.

A SCENARIO FOR GEOLOGIC OPERATIONS ASSOCIATED WITH THE LUNAR BASE: THE "GLOBAL" LUNAR BASE

It is not our intention to develop here a detailed plan for the geologic exploration of the Moon associated with a lunar base program. However, we can envision a series of operations that may be undertaken with such a program (Fig. 2) that will both support the establishment and operation of a permanent lunar base and provide a wealth of knowledge for lunar geoscience.

The most important step prior to base establishment is global geologic reconnaissance; this is most effectively accomplished by a polar-orbiting, remote-sensing mission (or series of missions) followed by a succession of simple, sample return missions. The landing sites for these sample return missions should be selected on the basis of the global data provided by LO or its equivalent. We envision a series of such missions aimed at gathering scientific, engineering, and resource-utilization data. Such information will be crucial to the intelligent selection of the ultimate lunar base site. The use of semiautonomous rovers to survey prospective sites in detail may also occur in this phase, depending upon the identified needs of the lunar base site-certification process. Because the need for geologic reconnaissance continues after the base is established, we envision this series of reconnaissance missions as a key part of the total lunar base infrastructure and such missions will continue for the indefinite future.

A great deal of geologic field work after initial base establishment will be conducted in the vicinity of the base site. This phase offers an excellent opportunity to field test the techniques of robotic teleoperation by conducting field study simultaneously with human and robotic geologists. The work would not only calibrate the robotic operatives for future independent traverses, but would also give the human teleoperators valuable experience in the use of their robotic alter egos for actual lunar geologic field work.

Eventually a series of increasingly longer traverses away from the base site to targets of geologic interest would be conducted. Such traverses could be designed to spend as much or as little time as desired at given field stations; moreover, route planning may involve circular paths to visit a series of different stops, or linear/radial paths to revisit previously examined stations. At least three, and possibly as many as five, robotic geologists should be available, thus permitting simultaneous traverses to many different geologic targets, in addition to allowing concurrent operational, instrument-deployment, or field-service missions. This phase of detailed geologic exploration would take years, if not centuries to complete, and it constitutes the bulk of geologic exploration of the Moon conducted from the lunar base.

During this phase of the exploration, we will undoubtedly encounter sites of great mystery and beauty. It is inconceivable to us that, no matter how compelling the robotic telepresence at such sites is, the human inhabitants of the Moon would not want to visit some of these sites in person. The whole human drive to explore and colonize the Moon defies rational analysis; therefore, we strongly advocate that the capability to transport humans to any point on the lunar globe be a required element of the infrastructure supporting a lunar base. Such human visits may not be common, but past experience with the human exploration drive suggests that they will be inevitable.

Although the ultimate goal of a lunar base program is the settlement of the Moon on a global scale, this goal will take many years to accomplish. It takes a great deal of energy to transport humans and their bulky life support systems great distances around the Moon from a single-site base. In some base-development scenarios, the ability to send human field workers to points on the Moon distant from the base occurs only in the advanced stages of base development. Possibly the most exciting aspect of our proposal to explore the Moon with teleoperated robots is that we can have scientific access to any point on the Moon very early in the base development program. In this sense, the use of teleoperated robots makes the single-site base into a "global" base. Such a strategy of exploration by robots under human control from a central base site is applicable to initial base operations on any terrestrial planetary body.

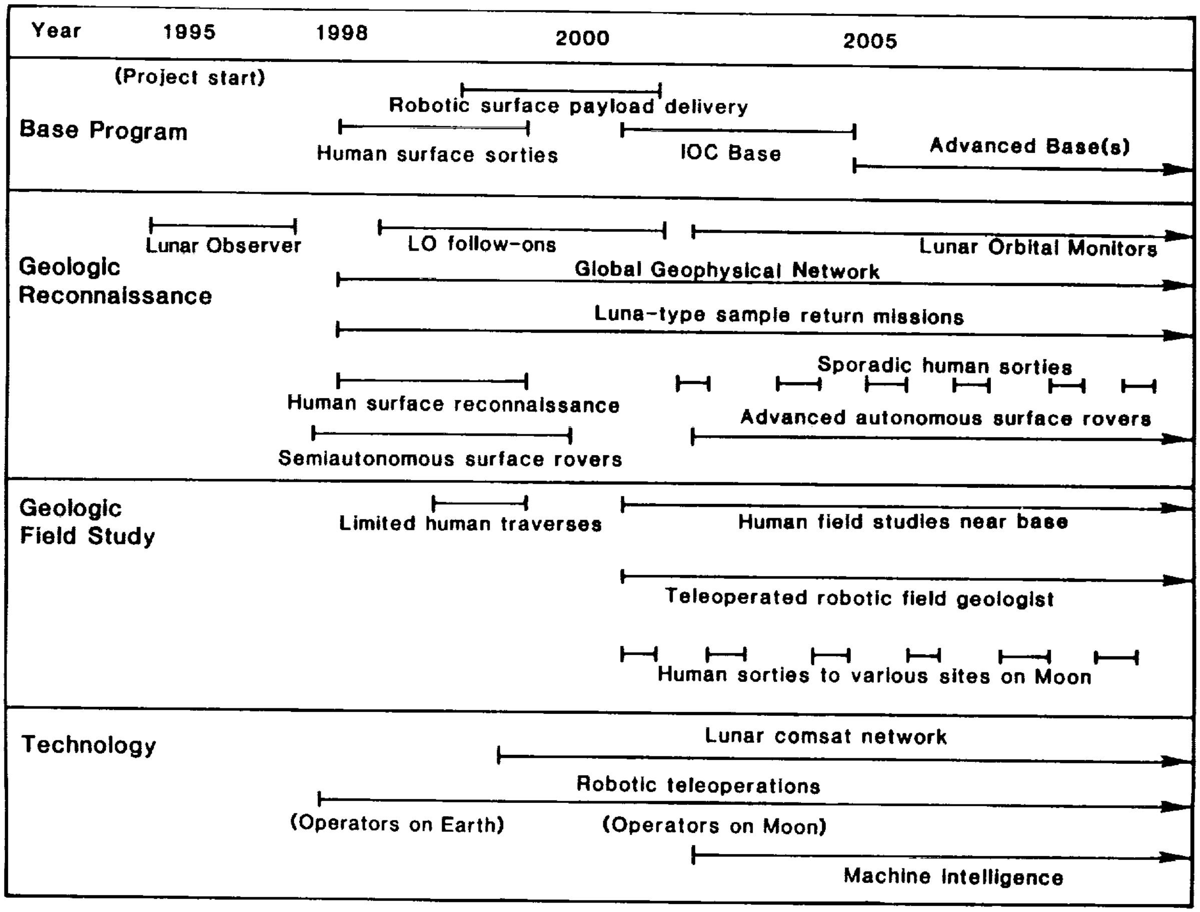

Fig. 2. Hypothetical timeline for geological requirements associated with a lunar base program. Milestones in lunar geological exploration are shown in relation to key events in the lunar base program and required technological developments. Scale of dates is arbitrary.

CONCLUSIONS

On the basis of the foregoing discussion, we conclude the following regarding the roles of humans and robots in the geologic exploration of the Moon:

1. Geologic reconnaissance is an ongoing effort prior to and concurrent with the establishment of the lunar base. Such reconnaissance may be best accomplished by remote sensing from lunar orbit and by relatively simple, automated sample return missions.

2. Geologic field study, by contrast, requires long stay times, intensive work capabilities, and human "presence."

3. The bulk of geologic field study conducted from the lunar base should be performed by teleoperated, robotic field geologists.

4. Humans in the field undoubtedly will be required in some instances. This capability should be a required element of the advanced lunar base infrastructure.

5. From the viewpoint of geologic exploration, teleoperated robots make the single-site base into a "global" base by providing a capability to explore any part of the Moon (or any planet) from the moment of base start-up operations.

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REFERENCES

Albritton C. C., ed. (1963) *The Fabric of Geology.* Addison-Wesley, Reading, Massachusetts. 372 pp.

Brazell J. W., MacLaren B. K., Williams W. M., and McMurray G. V. (1988) A three-legged walker (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 32. Lunar and Planetary Institute, Houston.

Cintala M. J., Spudis P. D., and Hawke B. R. (1985) Advanced geologic exploration supported by a lunar base: A traverse across the Imbrium-Procellarum region of the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 223-237. Lunar and Planetary Institute, Houston.

Hess W. N., ed. (1967) *Summer Study of Lunar Science and Exploration.* Univ. of California-Santa Cruz, NASA SP-157. NASA, Washington, DC. 357 pp.

Johnson N. L. (1979) *Handbook of Soviet Lunar and Planetary Exploration.* Univelt, San Diego. 262 pp.

LGO Science Workshop Members (1986) *Contributions of a Lunar Geoscience Observer (LGO) to Fundamental Questions in Lunar Science.* Southern Methodist Univ., Dallas. 86 pp.

Lunar Geoscience Working Group (1986) *Status and Future of Lunar Geoscience.* NASA SP-484. NASA, Washington, DC. 54 pp.

Morea S. F. (1988) The lunar roving vehicle—A historical perspective (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 176. Lunar and Planetary Institute, Houston.

Ryder G., Spudis P. D., and Taylor G. J. (1989) The case for planetary sample return missions: Origin and evolution of the Moon and its environment. *Eos Trans. AGU, 70,* 1495, 1505-1509.

Schmitt H. H. (1973) Apollo 17 report on the valley of Taurus-Littrow. *Science, 182,* 681-690.

Sheridan T. B. (1989) Merging mind and machine. *Technol. Rev., 92,* 32-40.

Spudis P. D. (1984) Lunar field geology and the lunar base (abstract). In *Papers Presented to the 1984 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 6. NASA Johnson Space Center, Houston.

Vinogradov A. P., ed. (1971) *Lunakhod 1—Mobile Lunar Laboratory.* Translation JPRS-54525. Joint Publications Research Service, Washington, DC. 189 pp.

Wilson J. and MacDonald D. (1986) Telepresence: Goal or by-product of remote systems. *Proceedings of the 10th Conference on Robotics,* pp. 5.1-5.8. Robotics International, Soc. Manufact. Eng., Chicago.

SOME ASTRONOMICAL CHALLENGES FOR THE TWENTY-FIRST CENTURY

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This paper addresses some of the scientific puzzles that astronomers may face in the next century. Four areas in astronomy are discussed in detail. These include cosmology and galaxy formation, active galaxies and quasars, supernovae and stellar remnants, and the formation of stars and planets. A variety of observatories on the Moon are proposed to attack these astronomical challenges.

INTRODUCTION

It is now fairly well established that the Moon is an excellent location from which to perform astronomical observations (e.g., *Burns and Mendell,* 1988; *Burns et al.,* 1990). The sky is dark and quiet, especially on the lunar farside. There is very little atmosphere, both in terms of neutral (10^5 molecules/cm^3; *Taylor et al.,* 1988) and ionized (<100 ions/cm^3; *Douglas and Smith,* 1985) gases. The ground is stable since the Moon is geologically inactive (seismic energy is 10^{-8} that of the Earth). The backgrounds of light, radio waves, radioactive-induced radiation, and even 1-Gev to 10-Tev neutrinos (*Cherry and Lande,* 1985) are much less than that on the Earth.

These characteristics have stimulated proposals for a wide variety of astronomical observatories on the Moon (e.g., *Mendell,* 1985; *Burns and Mendell,* 1988). Interferometric arrays at radio, optical, and infrared wavelengths will take advantage of the phase coherency that is possible without an atmosphere and on a stable surface. The lower gravity of the Moon may prompt the construction of very large radio dish antennas at high frequencies and large mirror arrays for use in the optical and infrared. The low radiation backgrounds, high vacuum, and negligible magnetic fields may permit the deployment of large-area, sensitive detectors of X-rays, gamma-rays, and cosmic rays, as well as detectors for neutrinos at moderate energies.

Clearly, a discussion of astronomical observatories on the Moon in the 21st century must be motivated by exciting and challenging scientific goals. In this paper, I attempt to examine what some of the astronomical challenges for the 21st century may be. My list is by no means exhaustive, but represents some of my own scientific biases and some of the astronomical programs that I find particularly attractive. My colleagues will surely extend this list. Crystal ball gazing is always dangerous in a field that is as rapidly advancing as astronomy. Such predictions are particularly dangerous now before the Hubble Space Telescope is in full operation and before the launch of the Gamma-Ray observatory, since these remarkable facilities will probably uncover new and interesting problems that could be appropriate for study with telescopes on the Moon. Nonetheless, it is likely that within the four areas that I discuss below a wide variety of interesting problems will remain well into the twenty-first century.

My approach in this paper will be to examine possible astronomical programs for the next century, independent of the location from which the observations will be conducted. In the final section, I will note what types of lunar observatories may contribute to the proposed astronomical programs.

COSMOLOGY AND GALAXY FORMATION

The Hubble Parameter

Hubble discovered in 1929 that the universe expands at a rate that is directly proportional to the distance. The constant of proportionality is now known as the Hubble parameter, H_o. After nearly 60 years of effort, we still only know the value of this most important cosmological parameter to within a factor of 2 (50-100 km/sec/Mpc). High-resolution, high-sensitivity observations in both the optical and infrared (IR) of "standard candles" such as Cepheid variables and supernovae in distant galaxies are needed to determine accurate distances. These data, combined with spectroscopically measured recession velocities, are used to calibrate the rate of universal expansion. Although the Hubble Space Telescope will make a giant leap forward toward determining the value of H_o, its aperture is relatively small (2.4 m) and the light-gathering power is limited. A larger optical telescope, possibly on the Moon (e.g., *Bely et al.,* 1989), would be able to resolve Cepheids and supernovae in more distant galaxies, and would possess greater sensitivity because of the larger aperture and lower light background (both galactic and extragalactic). This would also permit the use of fainter variables such as RR Lyrae stars to calibrate the Hubble flow.

Another intriguing possibility for constraining H_o is the use of water vapor masers in other galaxies. Recently, VLBI observations of trigonometric parallaxes of galactic H_2O masers were used to make the best estimate of the distance to the Galactic Center to date. An ultralong baseline radio interferometer with space-based antennas would be able to resolve masers in other galaxies and thus use similar techniques to determine accurate distances.

We should also conduct searches for other standard candles. An interesting possibility involves X-ray binaries whose periodicities are proportional to their luminosities. An array of X-ray variability monitoring telescopes with modest resolution would be adequate for this task.

Photometry of extragalactic star clusters with a large optical/IR telescope could provide another important distance indicator. Using the classic technique of main sequence fitting to the H-R diagram of a star cluster would allow accurate distance determinations.

Expansion of extragalactic supernovae and supernova remnants offers intriguing possibilities as measured by high-resolution optical and radio interferometers. The proper motion of compact components in the expanding nebulae combined with spectroscopic measurements of the radial velocities can yield a parallax or distance estimate.

More effort should be devoted to combining data from several wavelength bands to determine H_0. For example, sensitive microwave observations (away from sources of interference and high terrestrial background levels from the sky and ground) of clusters of galaxies coupled with X-ray measurements can, in principle, yield a measure of the Hubble parameter using the Sunyaev-Zeldovich effect (i.e., the cosmic microwave background is diminished by Compton scattering off thermal electrons in the intracluster medium).

Dark Matter and Closure of the Universe

One of the most important realizations of the past few decades is that the majority of matter in the universe (possibly 90%) is nonluminous. The nature of this dark matter is highly uncertain. Candidates include Jupiter-like planets, brown dwarfs, small black holes, as well as weakly interacting particles such as massive neutrinos (hot dark matter) and gravitinos (cold dark matter). The density of dark matter is of critical importance in determining the rate of deceleration of the expansion of the universe and the geometry of spacetime (an open or closed universe).

Galaxy rotation curves have been important in constraining the mass of galaxy halos. The discovery of flattened rotation curves (i.e., nearly constant velocities at large distances from the galaxy core) implies the existence of massive dark halos. Because of the sensitivity requirements, only a relatively few nearby galaxies have been measured. There is a great need to determine rotation curves for more distant galaxies, younger galaxies, and a broader class of galaxies. In addition to the usual techniques involving H-alpha and 21-cm spectral line observations, millimeter and submillimeter telescopes, free of absorption and scattering in the terrestrial atmosphere, could use molecular line transitions for the galaxy dynamics studies.

On larger scales, it has been known since the 1930s that clusters of galaxies must possess significant quantities of dark matter to counteract the large observed velocity dispersions. The magnitude of the motions of galaxies in clusters depends upon the total cluster mass. However, a direct determination of the mass is complicated by uncertainties in galaxy orbits and the state of cluster relaxation. Only relatively nearby clusters have measured dispersions so we presently do not know how cluster dynamics (and thus relaxation) evolves with time. Larger, more sensitive telescopes at optical and radio (HI line) frequencies are needed to make these measurements. However, better constraints on dark matter will come from combining the galaxy velocity data with high spectral and spatial X-ray observations of clusters. Hot gas (10^7 to 10^8 K) between cluster galaxies emits thermal bremsstralung radiation and serves as an excellent tracer of the gravitational potential well of the cluster. X-ray line emission can also be used to constrain mass loss from galaxies and motions of the intracluster medium. This combination of multiwavelength observations could be a useful tool in constraining the nature and extent of dark matter in clusters.

Even larger structures, superclusters and cosmic voids, are powerful probes of dark matter when combined with theoretical models. We need to measure the three-dimensional structure of the universe on scales of hundreds of Mpc at large look-back times. The galaxy covariance function, the distribution of galaxies, and the recently discovered large-scale streaming motions are presently the best tools that we have for distinguishing between various models of galaxy formation (pancake collapse on large scales vs. a hybrid hierarchical clustering). In turn, these models depend upon the particular form of dark matter that forms the seed density perturbation.

Finally, gravitational lenses offer an interesting method to probe dark matter. Multiply imaged quasars at radio and optical wavelengths and time variability of the images can constrain the total mass of the foreground lensing galaxy. However, ground-based studies have been limited by both resolution and sensitivity. Our present inability to detect and distinguish all images formed by a gravitational lens means that the resulting lensing mass estimate will be highly model-dependent.

Evolution of Galaxies and Clusters

An understanding of the formation and evolution of galaxies requires observations of galaxies at large distances. As in the above, ground-based observations have been limited by wavelength coverage (optical through near infrared), resolution, and sensitivity.

In particular, exciting new results using recently developed two-dimensional infrared imaging detectors suggest that primordial galaxies can be observed. Since these galaxies tend to be very red in color, one would like to search for such objects at even longer infrared wavelengths (10 to 100 μm). Large-area, infrared detectors are needed for very deep surveys of galaxies in formation.

High-sensitivity, high-resolution telescopes in the UV, optical, and IR offer the possibility of studying the initial mass function (IMF) of stars in galaxies and extragalactic star clusters. These observations, coupled with submillimeter studies of star-forming regions in other galaxies, will provide a broader perspective on the physics of protostellar collapse and early stages of star formation than is possible using the Milky Way alone.

The detection and imaging of distant galaxy clusters ($z > 2$-3) at both X-ray and optical wavelengths would add greatly to our understanding of the evolution of galaxy environments. In particular, line observations at X-ray energies of cosmologically distant clusters would help us to determine the origin of the elements in the intracluster medium and the evolution of the gas.

Another crucial database that is needed concerns the morphology of younger galaxies. When and how do galaxy disks form? How common is starburst activity and how might this be related to quasar activity? How does the interstellar medium in galaxies evolve with time? These questions could potentially be addressed with telescopes in space or on the Moon at UV, optical, IR, and millimeter/submillimeter wavelengths.

Background Radiations

The 3-K microwave background holds one of the important keys to understanding how structure emerged from the high-energy soup of the early universe. In particular, anisotropies in

this background are directly related to the spectrum of initial density fluctuations (unless an epoch of reionization occurred). Ground-based and balloon observations are difficult because of the Earth's atmosphere. The COBE satellite is the first U.S. spacecraft to be devoted to measurements of the microwave background. However, it too is limited in sensitivity and resolution. One needs to employ a wide range of sensitive detectors from millimeter to the IR at a variety of angular resolutions to detect and measure the full spectrum of anisotropies.

A diffuse X-ray background has been known to exist for nearly two decades. The origin remains unclear. *Einstein* Observatory deep field observations suggested that some of this background is composed of quasars, galaxy clusters, active galaxies, and stars. However, the results are still controversial. A deeper, higher resolution survey over a wide energy range (1 to 100 keV) is needed to resolve this question.

ACTIVE GALAXIES AND QUASARS

Nature of the Engine

Active galaxies and quasars often emit more energy as nonthermal radiation than the superposition of thermal emission from all the stars in the parent object. What kind of "engine" is capable of such energy production? Theoretically, collapsed objects—particularly black holes—seem to have the best promise. Energy can be efficiently extracted from the deep gravitational potential well within an accretion disk. However, there is at present no observational proof that massive black holes exist at galaxy cores.

Several galaxies such as M87 and M32 have suggestions that black holes may be present in their cores based upon studies of the distribution of light and stellar velocities. However, all such arguments are limited by the inability of current instruments to resolve the cores of generally distant active galaxies. An optical/IR interferometer could directly attack this problem with angular resolutions of a microarcsecond.

Similarly high resolutions could be achieved by a space-based radio interferometer. One could study how energy is extracted from the cores, and thus infer the properties of the engine, by observing radio jets at very high resolutions. However, such studies may be limited in linear resolution by Compton scattering effects.

One would also like to observe the gas in the direct environs of the engine. To accomplish this, high-spatial and-spectral resolution images are needed of the narrow-line region surrounding the active galactic nucleus.

Another clue to the nature of the engine can come from variability of the X-ray emissions from AGNs. Such observations set the firmest limits on the size of the emitting region. They also can produce estimates of accretion rates and information on the physics of the radiation processes. Time-serial, ungapped data is needed to detect multiple-periodic structures.

The recent detection of neutrinos from the supernova in the Large Magellanic Cloud ushered in a new observational branch of astronomy. The Moon is a particularly good location for observing neutrinos in the energy range from 1 Gev to 10 TeV since the background is much lower than on Earth. A large area neutrino detector might then be used for observations of active galactic nuclei. The neutrinos are created via high-energy interactions within the accretion disk and travel outward nearly unimpeded by the intervening galaxies and intergalactic gas. Thus, one may have an opportunity to study particle emissions in the direct environs of the engine and therefore constrain the nature of the engine.

Acceleration of Particles

We still have a very poor understanding of how electrons (and presumably protons) are accelerated to relativistic energies, thus radiating synchrotron emission in both galaxy cores and in larger-scale radio lobes. This is a general question that also applies to the formation of cosmic rays in galaxies, energetic particles in supernovae, and flares on stars and the sun. A very low frequency array (0.5 to 30 MHz) operating on the lunar farside or in lunar orbit offers the best hope for studying such processes. Such observations are not possible from the ground because of man-made interference and the ionosphere, and are difficult from Earth orbit because of the leaky ionosphere and auroral kilometric radiations from the magnetosphere. Low-frequency emission comes from low-energy particles. Broad-band, low-frequency spectra can provide important constraints on particle lifetimes and electron reacceleration. One can potentially differentiate between competing models by examining the shape of the broad-band continuum spectrum at low frequencies.

Higher-frequency observations at IR, optical, and X-ray can also constrain *in situ* particle acceleration in extended jets and lobes by examining the high-frequency turnover in the synchrotron spectrum. However, higher resolutions and higher sensitivities are needed to map these weak emissions.

Quasar-Galaxy Relationship

The nature of quasars remains controversial nearly 30 years after their discovery. There is good evidence for halos around the point-like cores of some nearby quasars; however, one can still argue that these nearby (generally lower-power) objects are not true quasars but are closer to Seyfert galaxies. To resolve this issue, two key observations are needed. First, detection, imaging, and spectroscopy of optical/IR "fuzz" around distant quasars ($z > 2$) are needed. This will require large-aperture telescopes and/or an optical interferometer. Second, one needs to search for galaxies around quasars. A convincing case has been made that nearby quasars tend to occur within groups of galaxies, but similar observations are needed for the more distant quasars.

Quasar Absorption Lines

Lunar or space-based observatories offer major advantages over ground-based facilities in studying quasar absorption lines. Many of the strong resonance lines for low atomic weight gases occur at UV wavelengths. Such strong lines can be used to study the nature of the foreground intergalactic medium, gas in cosmic voids, and spatial correlations of absorption clouds. All these have important cosmological significance. Similarly, quasars that lie in projection near the halos of galaxies or behind clusters of galaxies can be used as tracers of the foreground gas. In particular, the absorption lines could be used to study chemical abundances in galaxy halos and clusters of galaxies. Finally, the strong resonance lines of He and D that occur in the UV will allow us to make direct measurements of the He/H and D/H ratios, thus constraining models of cosmic nucleosynthesis following the Big Bang.

SUPERNOVAE AND STELLAR REMNANTS

Detection of Supernovae in Other Galaxies

Major questions concerning the origin and evolution of supernovae can be studied most successfully by examining supernovae and their remnants in other galaxies. This is an important issue for understanding the evolution of massive stars in galaxies, heat sources in the interstellar medium, and galaxy winds, and has additional possible cosmological significance (re-ionization of the microwave background?). An automated search for such supernovae using large-aperture optical and UV telescopes is required.

As mentioned earlier, supernovae (particularly type I) can be used as important standard candles for cosmological distance determinations and measurements of the deceleration of the Hubble expansion. The dispersion in magnitudes of SN I are small and thus serve this purpose very well.

Nearby Supernovae and Their Remnants

The recent supernova in the Large Magellanic Cloud has demonstrated how successful current models are for describing the evolution of the explosive event. However, one among many surprises that emerged from the data had to do with the progenitor star. Its color and spectral type were quite unlike that expected to produce a type I supernova. Clearly, there is much to be learned about which stars become SN I and SN II. Surveys in both our galaxy and in our neighboring galaxies are needed to fill in missing pieces of the puzzle regarding the final explosive stages of stellar evolution. Since massive stars that produce supernovae radiate most effectively in the UV, space-based studies of such stars would be productive.

Now that the neutrino window is open, neutrino observatories could play a useful role in astrophysics. Neutrino emissions from supernovae are critical tests of explosive nucleosynthesis. High time resolutions over a broad range of energies are needed to constrain models. For more distant galaxies, sensitivity, and thus large-area detectors, will be required.

Much progress has been made in studying the remnants of past supernova explosions. Particularly important advances have come in combining data from radio through X-ray wavelengths. Such a multifrequency approach would be very productive in studying the physics of expanding remnants. High-spectral and -spatial resolution mapping at X-ray, optical, and UV wavelengths are crucial in studying the interaction of the supernova shock wave and the interstellar medium.

Searches for more remnants need to be conducted within our galaxy and within neighboring galaxies. With the several dozen reliable remnant identifications that are presently known, it is difficult to classify supernovae and determine the physics of the progenitor event.

Stellar Remnants

What are the progenitors of various endpoints in the stellar evolution cycle: white dwarfs, brown dwarfs, neutron stars, and black holes? Theoretically, the ideas regarding the evolution of giant stars into compact objects are fairly well established. However, observational verifications are still lacking and there are uncertainties in the details of the evolution. Searches for more of these stellar endpoints are needed. For example, the cooling of a white dwarf is an important check on the equation of state and thermal conduction models. As the star cools, more sensitive observations at progressively redder wavelengths are needed. Similarly, X-ray pulsations can be used to identify more candidate neutron stars.

In general, X-ray observations are useful probes of the direct environs of stellar remnants. Higher-sensitivity X-ray observations for point sources will push the detectability for remnants radiating with a blackbody spectrum (i.e., accretion disks). Variability studies, as noted above, can set limits on source sizes and accretion rates. X-ray binaries are particularly interesting to study in this regard.

Finally, the cores of globular clusters may house a variety of stellar remnants. High-resolution observations at optical and IR wavelengths can be used to determine the core stellar dynamics and set limits on the masses of compact objects. X-ray observations can provide good hunting grounds for binaries and central compact objects in globular clusters.

FORMATION OF STARS AND PLANETS

Gas Clouds into Stars

One of the shining successes of modern astrophysics has been the theoretical understanding of the evolution of stars on the H-R diagram. Although the "middle-age" portion of a star's evolution is relatively well understood, there remain many unsolved problems regarding the initial collapse and formation of stars from gas clouds.

We have, at present, a poor understanding of the initial mass function (IMF) for stars, particularly for low-mass stars. How does this function depend upon the star's metallicity and location within the galaxy? What is the time dependence of the IMF? How does this function vary with local physical conditions? The answers to these questions, especially for lower-mass stars, will come with detailed imaging and spectroscopic observations in the red and infrared.

The contraction phase of stars from gas clouds is not well understood. For single stars, how is angular momentum from an initially rotating cloud dissipated? What is the role of magnetic fields in the collapse process? The study of these crucial questions will require high-resolution total intensity and polarization observations in the IR. The IR is important to see through the outer cloud and accurately locate the protostellar core. Large-area detectors would provide such sensitive, high-resolution observations. One would like to conduct time-dependent imagery of the cloud cores to examine proper motions and radial velocities of the gas motions. This would require both high-spatial and -spectral resolutions.

Young stellar objects have recently been observed to possess outflows presumably from the vicinity of an accretion disk. The energization and collimation of such flows are uncertain. Observations at submillimeter wavelengths would be most useful here since molecular transitions become optically thin at these wavelengths. One could attempt to study the chemistry of the outflow and direct environs of the protostar as a function of distance from the core. Near-IR and millimeter imaging could be used to determine the location of the core and the properties of the initially colimated outflow.

Little is known about the very earliest stages of collapse of cold clouds. Because of the low temperatures (about 10 K), observations in the far IR would be needed to probe these clouds. The IRAS demonstrated the value of far-IR observations in studying star-formation regions. Two-dimensional polarimetry would be important to examine how magnetic fields support the clouds,

generate turbulence, dissipate angular momentum, and aid in the collimation of outflows. Very little work has been done to date to study these quiescent clouds because of the inaccessibility of the far IR from the ground.

One must also study star formation in other galaxies as a link to understanding the physics of star formation in the Milky Way. Protostellar clouds in our galaxy provide us with an opportunity to study details of the star formation process because of their proximity. Star forming regions in other galaxies provide us with an opportunity to study the broader dependences of star formation on galaxy environment—a more global perspective.

Other Planetary Systems

The formation of planets is even less well understood than stars. Presently, we know of only one planetary system, namely our own. Clearly, searches for other such systems are needed to guide our understanding of our solar system.

One technique that has been suggested for such searches is coronographic imaging. The Hubble Space Telescope will conduct this type of imaging for nearby stars. One would like to extend this to more distant and less luminous stars at optical and IR wavelengths using larger lunar telescopes.

The IRAS operating in the far IR revealed several interesting candidates for protoplanetary disks. These observations demonstrate the feasibility of such studies at both IR and optical (using scattered light) wavelengths.

Astrometry of stars is the most straightforward method for detecting gravitational perturbations by planets. Ground-based observations have been conducted for several decades, but with no clear detections because of atmospheric limitations. Optical/IR and radio interferometers with microarcsecond resolution would clearly advance these searches.

POSSIBLE LUNAR OBSERVATORIES

In my opinion, the natural location for the next generation of space-based astronomical facilities, following the Great Observatory series, is the Moon. The many advantages of the Moon described in the Introduction, coupled with the regular transportation, mining, and self-sufficient habitats that are anticipated for the early twenty-first century make lunar observatories look very attractive. In addition, the cost (in fuel mass calculated for shuttle-class engines) of transportation of materials from the Earth's surface to the Moon's surface is only about 50% more than from the Earth to geosynchronous orbit (e.g., *Keaton,* 1985).

The Moon as a site for astronomy can contribute to an understanding of the astronomical problems described in the previous sections. Among the possible lunar observatories are (1) a farside low-frequency array, operating from about 0.5 to 30 MHz, that would open an entirely new window to the electromagnetic spectrum and contribute to our understanding of particle acceleration from the sun to extragalactic radio sources; (2) an optical/IR interferometer with a resolution of a microarcsecond that could be used to further constrain the Hubble parameter, locate planets around other stars, and resolve the cores of active galaxies; (3) a far-infrared telescope, possibly located within a polar crater, that could take advantage of the natural cryogenic environment of the Moon (with large-area, "naked" detectors) to map stars and galaxies in the process of formation; (4) a submillimeter array, free of the absorption of the Earth's atmosphere and on the stable lunar surface, that could probe deeply into the cool interiors of molecular clouds and measure the dynamics of molecular gas in other galaxies; and (5) a large area X-ray detector that could accurately measure the X-ray background and monitor the variability of compact galactic and extragalactic sources. For more details on these instruments, see *Burns and Mendell* (1988).

There are many challenges awaiting the astronomical community in the next century. Access to electromagnetic windows in the IR, the UV, submillimeter, X-ray, and very low radio frequencies will allow us to approach solutions to problems from star to galaxy formation. My scope in this paper has been limited. I have not even touched upon exciting problems in solar and planetary astrophysics. However, it is clear that there is an abundance of interesting problems that might be addressed with astronomical observatories on the Moon.

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REFERENCES

Bely P. Y., Burrows C. J., and Illingworth G. D., eds. (1989) *The Next Generation Space Telescope.* Space Telescope Science Institute.

Burns J. O. and Mendell W. W., eds. (1988) *Future Astronomical Observatories on the Moon.* NASA CP-2489.

Burns J. O., Duric N., Taylor G. J., and Johnson S. W. (1990) Observatories on the Moon. *Sci. Am., 262,* 42.

Cherry M. and Lande K. (1985). A lunar neutrine detector. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 335-343. Lunar and Planetary Institute, Houston.

Douglas J. N. and Smith H. J. (1985) A very low frequency radio astronomy observatory on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 301-306. Lunar and Planetary Institute, Houston.

Keaton P. W. (1985) A Moon base/Mars base transportation depot. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 141-154. Lunar and Planetary Institute, Houston.

Mendell W. W., ed. (1985) *Lunar Bases and Space Activities of the 21st Century.* Lunar and Planetary Institute, Houston. 865 pp.

Taylor G. J., Burns J. O., Femini L., and Johnson S. (1988) Site selection criteria for astronomical observatories on the Moon (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 238. Lunar and Planetary Institute, Houston.

RADIO ASTROMETRY FROM THE MOON

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An array of three radio telescopes on the Moon, separated by 100-1000 km, could measure the positions of compact radio sources 50-100 times more accurately than can be done on Earth. These measurements would form an all-sky reference frame of extreme precision (5-10 μarcsec) and stability, with applications to the dynamics of the solar system, our galaxy, and nearby galaxies.

RADIO ASTROMETRY: EARTH-BASED LIMITATIONS

The angular resolution, θ, of a telescope of diameter d receiving electromagnetic radiation at a wavelength λ is limited by diffraction to $\theta \approx \lambda/d$. At optical wavelengths, the diffraction limit is <1″ for telescopes larger than about 10 cm. At radio wavelengths of 1-6 cm, where telescopes of diameter 50-100 m operate, the diffraction limit is 20″-10′. This would seem to favor optical wavelengths for high angular resolution. However, the use of superheterodyne detectors at radio wavelengths allows data from widely separated telescopes to be coherently combined, thus synthesizing very large apertures. The angular resolution of this technique, radio interferometry, is limited only by the physical separation of the antennas.

The precision of radio astrometry, the measurement of angular positions on the sky, can be superior to the angular resolution. However, a number of systematic error sources limit the accuracy. The two most serious such error sources for Earth-based measurements arise from the troposphere and from deformations and irregular rotation of the Earth.

The troposphere introduces a delay of about 7 nsec to incoming wavefronts. The component of this delay due to water vapor is difficult to calibrate precisely because the distribution of water vapor is very inhomogenous, both spatially and temporally. Water vapor radiometers measure the integrated line-of-sight emission from the 22-GHz water line, an imperfect measurement of the column density of water vapor along that line of sight. Even with parameter estimation from the astrometric measurements, this limits the accuracy of the measurements. Future water vapor radiometers will improve the accuracy by an unknown amount. Based on current experience, tropospheric water vapor may well limit the accuracy of radio astrometry to 300-500 μarcsec.

The rotation of the Earth is uniform to first order, both in rate and in the direction of the rotation axis. However, there are second-order perturbations in both quantities due to a variety of effects with large stochastic (i.e., unpredictable) components. These stochastic terms arise from the evolving fluid sheath of the Earth, as well as the large fluid core. At the submilliarcsecond level of accuracy, the rotation phase will need to be measured several times per day, or perhaps even much more often, precluding the possibility of estimating it from astrometric measurements. The largest variation in the shape of the Earth is due to lunar tides, and can be modeled. However, more minor deformations due to the variable distribution of the atmosphere and of liquid and solid water greatly complicate the time variation of the vector between the two radio telescopes (the baseline vector). One or several of these rotational and tidal effects may impose a fundamental limit of 300-500 μarcsec on the accuracy of ground-based radio astrometry.

Differential Earth-based observations can determine small angular separations (i.e., between two closely spaced radio sources) to an accuracy of better than 500 μarcsec. However, the accuracy of an all-sky astrometric catalog is limited by the effects discussed above.

ADVANTAGES OF THE MOON

The Moon allows an opportunity to escape from the two largest error sources of Earth-based radio astrometry. The Moon has a negligible neutral-gas atmosphere, and because it has no fluid sheath or sizable liquid core, its rotation and deformations can be modeled to very high accuracy. This latter facet gives it a large advantage over Earth orbit as a site for high-precision astrometry.

PROPOSED LUNAR ASTROMETRIC INSTRUMENT

I propose an array of three radio telescopes on the Moon. They would be arranged very roughly in an equilateral triangle located near the lunar equator (thus allowing most of the sky to be accessible). The distance between telescopes would be in the 100-1000 km range (it is not clear what separation will give the best accuracy). Lightweight antennas of about 15-m diameter with surface tolerance in the range 0.2-0.5 mm would be used.

Microwave links or coaxial cables would connect the three antennas to a control center (this could be located near one of the antennas). Local oscillator phase would be distributed to the antennas, with intermediate frequency (IF) data transmitted back for correlation. The correlation process would reduce the data volume to a few kilobytes per hour, which would be sent to Earth for subsequent analysis.

SOURCES OF ERROR AND ESTIMATED ACCURACY

This instrument would concentrate on several hundred compact radio sources, distributed more or less uniformly over the sky. It would observe the entire set of sources 3-20 times during a month (only half of them are visible at any one time).

This observing strategy reveals systematic effects in the data much better than long integrations on individual sources. A one-year observing program on 500 sources would give a total integration time of about 15 hours per source. This yields a precision from S/N limitations of

$$\Delta\theta = \frac{8\mu\text{arcsec } T_{50}}{D_{100} S_{100} (BW_{GHz})^{0.5} d_{15}^2 \nu_{30}}$$

T_{50} is the system temperature in units of 50 K, D_{100} is the antenna separation in units of 100 km, S_{100} is the source correlated flux in units of 100 mJy, BW_{GHz} is the IF bandwidth in GHz, d_{15} is the antenna diameter in units of 15 m, and ν_{30} is the sky frequency in units of 30 GHz.

The absence of a neutral-gas atmosphere eliminates one large error source. The delay introduced by charged particles in the solar wind and the Earth's magnetosphere can be measured and removed by dual frequency observations (e.g., 40 GHz and 2 GHz). The bandwidth at the lower frequency can be much smaller than at the higher frequency.

The rotation of the Moon must be either modeled or determined via parameter estimation. Because the Moon is geologically quiet, with no atmosphere or oceans, its motions are very deterministic. Libration (nonlinear rotation) terms due to gravitational harmonics up to degree and order 4 (18 terms), with perhaps a few fifth degree terms, will be important for accuracies in the microarcsecond regime. In addition, Love numbers through degree 3 or 4 (12-18 parameters) will be required to model tidal deformations.

The proposed astrometric instrument will therefore be required to solve for 30-40 time-invariant lunar motion parameters. These parameters are of great interest in their own right, as they provide information on the physical properties of the lunar interior. In addition, there may be a small stochastic component to lunar rotation. It is at least 100 times smaller than its terrestrial equivalent, and varies on much longer timescales, allowing it to be easily determined from the data. The fundamental limitation on astrometric accuracy from lunar motions appears to be below 10 μarcsec.

Source structure effects may cause the largest errors. Compact extragalactic radio sources have nonzero sizes and time-variable structure. The relatively short baselines (<1000 km) of the proposed instrument will result in many sources being only slightly resolved. The measured positions for these sources will be that of the emission centroid. In general, the smallest apparent position shifts will occur for the most compact sources. There are many moderately weak (100 mJy) sources with estimated sizes under 50μarcsec at observing frequencies of 30-50 GHz. The variation in emission centroid will generally be <20 μarcsec. Measurements of many sources will determine a global frame to an accuracy at least a factor of 2 better, or <10 μarcsec. The desire to minimize source structure effects on the data forces the sky frequency to be at least 30 GHz, with 40-50 GHz desirable. Yet larger frequencies are disadvantageous due to higher system temperatures and degraded antenna performance.

The control of instrumental errors will be a major problem at this level. Thermal and gravitational deflections in the antennas will have to be known to better than 0.1 mm. Systematic instrumental phase errors must be smaller than 0.5° for 10 μarcsec accuracy at 40-GHz sky frequency on a 200-km baseline. Very careful design and construction will be required, but this should not be the limiting source of error.

CONCLUSIONS AND APPLICATIONS

The overall limiting astrometric accuracy of this instrument is estimated at 5-10 μarcsec, a factor of 50-100 improvement over Earth-based capability. A global reference frame with this accuracy would yield significant scientific results. The dynamics of our own galaxy could be determined to high accuracy through the study of the proper motions and parallaxes of compact galactic radio sources. The galactic rotation curve and deviations from rotational motion would be revealed in detail. Proper motions of nearby (distance less than 5 Mpc) galaxies would be measurable.

This reference frame would have other applications. The positions of planets could be determined in the frame through VLBI observations of spacecraft in orbit about those planets. This would allow very precise studies of solar system dynamics.

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REQUIRED TECHNOLOGIES FOR LUNAR ASTRONOMICAL OBSERVATORIES

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Each of the major new observatories proposed to take advantage of the characteristics of the lunar environment requires appropriate advances in technology. These technologies are in the areas of contamination/interference control, test and evaluation, manufacturing, construction, autonomous operations and maintenance, power and heating/cooling, stable precision structures, optics, parabolic antennas, and communications/control. Telescopes for the lunar surface need to be engineered to operate for long periods with minimal intervention by humans or robots. What is essential for lunar observatory operation is enforcement of a systems engineering approach that makes compatible all lunar operations associated with habitation, resource development, and science.

INTRODUCTION

Several types of astronomical observatories have been proposed to take advantage of the unique nature of the lunar environment. These observatories include the Very Low Frequency Array (VLFA) for radio astronomy (*Douglas and Smith,* 1985), the Moon-Earth Radio Interferometer (MERI) (*Burns,* 1985, 1988), and the Optical Interferometer (*Burke,* 1985, 1990). Examples of some of the technology development considerations to be addressed for each of these observatory types are shown in Table 1. With each proposed telescope, there are myriad engineering issues to be resolved (*Burns et al.,* 1990).

ADVANCED TECHNOLOGIES AND CRITICAL ENGINEERING ISSUES

A major difficulty in determining what the critical engineering issues are for these three astronomical observatories on the Moon is that these systems are in their early planning stages and point designs are in the future. The identification of critical engineering issues is somewhat arbitrary predicated on judgment as to observatory design and types of materials and technologies to be used. There will be many significant components such as foundations and supporting structures (which will have stringent requirements for stiffness and thermal stability), thermal control systems, power, communications and control, and data processing and transmission (*Johnson,* 1988).

Each of these significant components suggests a set of critical engineering issues that can be addressed from the point of view of required technologies to make the lunar telescopes perform in an acceptable way. Table 2 lists the significant new technologies discussed in this paper that will be required for these three example telescopes.

CONTAMINATION/INTERFERENCE ISSUE

One of the challenges facing telescope designers and operators is coping with natural and operations-induced sources of contamination/interference (Table 3) on the Moon. Sources of

TABLE 1. Examples of technology development considerations for observatory options.

MERI - Parabolic Dish Radio Antenna
- System definition and specifications
- Site selection and characterization
- Thermal strain rates at sunrise and sunset
- Sun shield
- Foundation excavation and placement
- Foundation dynamics
- Breakdown into transportable packages with semiautomated erectability
- Shielding for electronics and other vital operations

VLF Radio Telescope - Dipoles on Surface over a Large Area
- System definition and specifications
- Site selection and characterization
- Capability to traverse large area and place dipoles
- Erection and shielding of a control facility

Optical Interferometer
- System definition and specifications
- Site selection and characterization
- Control capability (stringent requirements limiting differential settlements, tide compensation)
- Rails several kilometers long laid out on lunar surface require locating and modifying a suitable site
- Dynamic response of lunar soil to movement of telescopes
- Preservation, cleaning, and renewal of optical surfaces and coatings

General Technology Needs for all Three Options
- Automation, telepresence, and robotics for construction, operations, and maintenance
- Human factors considerations (man-in-the-loop) and realistic artificial intelligence interaction
- Stiff, stable, light-weight structures and materials either transported from Earth or made on the Moon
- Data gathering, storage, processing, and transmission (e.g., with a communications satellite orbiting the Moon)
- Thermal control, cryocoolers, heat dissipation, and heaters as appropriate
- Power sources to serve lunar outpost requirements
- Potential applications of superconductivity
- Mobility on the surface (robots/human)
- Earth-to-Moon and return transportation
- Self-organizing failure characteristic prediction

TABLE 2. Technologies for lunar observatories.

Contamination/Interference Control	Manufacturing:
Test and Evaluation	Terrestrial
Construction	In space
Power and Cooling/Heating	Lunar
Stable Precision Structures	
Optical Systems	Autonomous/Semiautonomous:
Parabolic Antennas	Deployment
Shielding	Operations
Communications and Control	Maintenance

TABLE 3. Some contamination/interference sources and implications.

Fine-grained particulates from the lunar surface—stick to surfaces

Meteoroid impacts—loft debris; cause surface pitting

Gases—stick to surfaces
- Natural
- Induced by operations
  - rocket plumes
  - outgassing from excavations/fill in soil and mining/manufacturing
  - outgassing from suited workers

Radio frequency—interference problem for radio astronomy/communication

Ground shock/vibrations, both natural and operations-induced—problem for optical interferometers/other instruments

Other:
- Reactor radiation
- Waste heat from power sources

contaminants and interference will have implications for all aspects of the lunar astronomical observatory performance (Tables 4 and 5).

Particulates and gases deposited on surfaces can significantly alter optical and thermal properties of surfaces and degrade performance. They can defeat the important attributes of delicate coatings and scatter light, create assembly and erection problems (particulates), and lead to problems in electronics. This paper first looks at some contamination/interference control technologies needed and then deals with selected other technologies for lunar observatories.

Telescopes on the Moon may tend to be surrounded by transient atmospheres resulting from manned and unmanned operations in the vicinity (*Burns et al.,* 1991). That there will be a transient gas cloud is evident from the work of investigators (T. H. Morgan, personal communication, 1988) interpreting measurements from atmospheric detection instruments on the Apollo Lunar Surface Experiments Package (ALSEP). Under a worst-case scenario, the "cloud" of transient atmosphere could degrade astronomical observations. The "cloud" density will be dependent on relative rates of contaminant generation and removal. Removal is by collisions with solar wind protons, diverging orbits of particles, expansion into space, decomposition and evaporation, and entrapment or sticking in the lunar soil or regolith.

Particulate and gaseous deposits on critical surfaces of astronomical instruments on the lunar surface may occur as a result of both natural and man-made environments. Deployment and emplacement will involve vehicles and perhaps suited construction workers outgassing water and other by-products of metabolism and suit functions.

TABLE 4. Instrument contamination/interference and possible countermeasures.

Instrument	Possible Contamination/Interference	Possible Countermeasures
VLFA and MERI	Radio frequency interference VLFA—1 MHz to 50 MHz MERI—GHz regime	Preservation/allocation of radio frequencies
	Gases from rocket plumes, manufacturing, construction, excavation, etc. creating denser transient atmosphere and ionosphere	Control and use of "clean" technologies
	Fine-grained particulates from lunar soil clinging to surfaces. Particular concern with MERI relative to erection/operation of steerable dish	Control of gas and dust mitigationtechnology
	Ground shock/vibrations related to pointing and tracking of steerable dish for MERI	Use of "quiet" operations technologies nearby; define keep-out zones
Optical Interferometer	Gases "sticking" to optical surfaces and changing optical properties	Reduction of effluent at source; technology to purge and renew surfaces
	Fine-grained particulates from lunar regolith adhering to optical surfaces and other surfaces	Dust mitigation technologies(reduce dust-disturbing operations); clean-up technologies
	Radio frequency interference with broad-band data transmission/reception	Frequency allocation and transmitter standards
	Ground shock/vibrations interfering with nanometer precision alignments	Alignment sensing/adjustment in real time; shock/vibration isolation at telescope; shock suppression at origin; keep-out zones

TABLE 5. Some recommended contamination technology programs for lunar surface astronomy.

Contamination effects research
- Determination of effects
- Development of acceptable standards

Modeling of the mechanisms of contamination

Critical diagnostics/measurements program for lunar surface contamination
- Material/structural samples deployed to lunar surface and data collected
- Verification/comparison of model to results of data collection

Development of contamination prevention and cleaning techniques

Required power and communication units may be sources of unwanted heat, radiation, and radio frequency interference. Surface operations for emplacement of observatories may involve excavation, compaction, trenching, and fill operations that will accelerate and disperse particulates and liberate gases.

CONTAMINATION/INTERFERENCE CONTROL TECHNOLOGIES

Contamination control is a prime area of concern for virtually any telescope installation. Contamination control technologies required for telescopes to be based on the Moon include protection of precision surfaces and parts through the life cycle including manufacture, assembly, test and checkout, transportation, landing, erection/deployment, and lunar surface operations/maintenance. Safe techniques to remove contaminants at any stage in this life cycle are needed. Obviously, means to detect and establish the nature of contaminants are required so that the severity of the contamination problems can be monitored and appropriate countermeasures can be taken.

Particular attention is needed to ascertain the implications of long-term lunar surface operations for accumulation on surfaces of contaminants such as fine-grained particulates, products of outgassing of materials, and propulsion products.

There are needs for investigations to improve our understanding of optical and thermal control coatings, their behavior, and interactions with contaminants and radiation environments on the lunar surface. The processes of contamination and contamination removal can be modeled to assist in predictions of the severity of problems developing as a result of various operational scenarios. To develop useful models will require an improved understanding of the physics of surface deposition and better characterization of the lunar environments, both natural and operations induced. The longer-term goal will be to develop techniques for surface cleaning and coating restoration *in situ* on the lunar surface.

Johnson et al. (1991), in this volume, discuss environmental effects on astronomical observatories that relate to experience with recovered Surveyor 3, Solar Max, and other parts exposed to the lunar and orbital environments. The results they present are instructive in formulating future contamination control technologies.

TEST AND EVALUATION TECHNOLOGIES

A methodology, facilities, and resources are needed to assure that systems concepts for a lunar astronomical observatory can and will be modeled and tested adequately at various stages of concepualization, research, development, fabrication, and preparation for launch. The goal is to avoid unpleasant surprises after arrival on the lunar surface. Questions to be resolved by a test and evaluation process relate to the operational effectiveness and suitability of the observatory system. Effectiveness questions for test and evaluation are those tied in with performance such as pointing and tracking accuracy and precision, resolution, and image quality. Suitability questions relate to reliability, maintainability, and supportability of the telescope operational systems on the lunar surface. All the suitability questions are of enormous importance when the logistics line of support is from the Moon to the Earth (*Johnson and Leonard,* 1988).

Early involvement of test and evaluation methodologies will start at the telescope system concept level to make adaptation possible to assure testability. Ground-based simulators will be needed to verify interoperability and autonomy of telescopes. Systems for calibration of telescope systems are an important aspect for the prelaunch modeling, test, and evaluation process.

MANUFACTURING TECHNOLOGIES

Two types of manufacturing capabilities should be pursued to support lunar-based astronomical observatories. One set of capabilities will be on Earth and the other eventually on the Moon. Terrestrial manufacturing of telescopes will be aimed at producing very lightweight, reliable, and packageable components of observatories for shipment to and deployment on the Moon. One example will be composites manufacturing that requires technology development for coatings, joints, fabrication techniques, and complex fixtures for support of steerable dishes and mirrors for radio astronomy and optical astronomy. Parts should be produced so that they are interchangeable where possible (e.g., the struts supporting mirrors and dishes). Optics and electronics suitable for long-term use at a lunar observatory require special care in manufacturing to avoid faults and impurities that lead to subsequent degradation and failure.

In the area of manufacturing, the prime technology issue is producibility. Required for lunar optical and radio telescope dishes are capabilities to manufacture, assemble, inspect, test, and maintain high quality at reasonable cost. This technology issue becomes of greater importance as more components are required as in the case of interferometers. Ultimately, some components may be manufactured from lunar materials on the Moon—requiring a whole new set of manufacturing technologies.

CONSTRUCTION TECHNOLOGIES

Mobility and transportation with minimal environmental impact are key elements in the deployment of the observatory and its components on the lunar surface. Transportation of components to the lunar surface will, for example, require safe and secure packaging to preserve the integrity and cleanliness of delicate optical and other elements. Deployment and erection sequences must be carefully preplanned so that components match up in spite of temperature variations from component to component and with time. Technologies for deployment should minimize the needs for intervention by construction workers in spacesuits. Teleoperated cranes may serve as backup for automated off-loading of components from arriving payload packages. Ways will be needed to prevent the accumulation of fine-grained particulates from the lunar regolith on mating surfaces of contiguous elements of the observatory. Confidence in deployment and erection technologies will be critical in determining the future success of the observatory.

The emplacement of an observatory on the Moon will require the capability to maneuver vehicles over many-kilometer distances in remotely controlled (teleoperated) or preprogrammed operational modes. The VLFA observatory will depend for deployment on the capability to emplace dipoles over surface areas extending to 20 or more kilometers in diameter (*Burns et al.,* 1989). A variety of terrains will be encountered including small and large craters, boulder fields, hills, and valleys.

AUTONOMOUS OPERATIONS AND MAINTENANCE TECHNOLOGIES

Autonomous operation and maintenance of telescope systems on the farside of the Moon is a goal that will be difficult to achieve because of the unpredictability of the problems that will be encountered. Allowance should be made for teleoperation and maintenance workers in spacesuits if unanticipated difficulties arise. Prelaunch test and evaluation efforts on Earth will focus on various aspects of teleoperated operation and maintenance to predict and resolve difficulties before arrival at the Moon.

The vehicle associated with the VLFA should be able to operate in several different modes as needs dictate change from manual operation to local teleoperation or to remote teleoperation, or perhaps to autonomous operation and hybrid modes. Technical issues with the vehicle design relate to vehicle size and mass, load-carrying capacity and range, communications and control, number of wheels (or tracks), manipulator capabilities, power, and how the vehicle copes with the environment (e.g., the soil, rock, and terrain; vacuum, meteoroid impact; radiation; extremes of temperature; and diurnal cycles of solar radiation). The robotic vehicle system that supports the construction of the VLFA on the lunar surface will be required to support all phases of the effort including transport of large reels of cable, laying out the cable according to the predetermined plan, emplacing a central station, and performing maintenance and repair tasks. The vehicle must have flexibility to meet unanticipated needs such as coping with cable breakage, unusual terrain, soil variability, and layout adjustments.

The prime power source for the lunar astronomical observatory and associated facilities will be either solar or nuclear. Power requirements will probably be much less than 100 kW for VLFA. Solar arrays appear to be suitable for the VLFA if backed by sufficient energy storage capacity (batteries or regenerative fuel cells) to continue operations during the lunar night. There is a strong need for development of regenerative/rechargeable power storage devices both large and small for use with solar energy devices to furnish power during the 14-Earth-days lunar night. One option for the next-generation battery is a Na/S battery being developed at the Aero Propulsion Laboratory at Wright-Patterson Air Force Base, Ohio (*Sovie,* 1988). Radioisotope thermoelectric generators also are possible power sources although they are inefficient and generate relatively large amounts of heat. Focal plane arrays for optical telescopes on the Moon will need to be cooled. Much technology development is required for cryocoolers to fill this need. One option is the development of an integrated radioisotope-fueled dynamic power generator and cryocooler to cool the focal plane arrays.

STABLE PRECISION STRUCTURES TECHNOLOGIES

Technology is required for large, stable, precision structures to support observatory components on the Moon. Geometrically precise structures using advanced materials such as metal matrix composites are needed. These structures can be designed to have the required very low coefficients of thermal expansion.

The supporting structures for large optical telescopes and steerable dish radio telescopes on the Moon need attention to isolation from disturbance, structures and controls interaction, and testing issues as portrayed in Table 6.

Some types of observatories on the Moon will involve very large structures or sequences of structures that must be precisely aligned and that must be movable and must track to high precision (e.g., to milliarcseconds). Technologies will be required to measure surface accuracies of millimeter and submillimeter radio astronomy parabolic dishes to 5-10 μm and to make adjustments if needed (Table 7).

TABLE 6. Issues relating to large structures to support optics and steerable radio telescope dishes on the Moon.

Disturbance Issues
- What are the critical disturbances
  - Natural—seismic shock, thermal
  - Operations-induced—ground shock, vibrations
- What mitigation technologies are applicable?
- How can disturbances be characterized and mitigation approaches formulated?

Structure Issues
- What approaches can be taken to build light-weight, high-stiffness structures optimized for the lunar ⅙ *g* and extreme thermal environments?
  - Structural parameters—how ascertained?
  - Improved models (computational)
  - Test and instrumentation challenges
  - Optimization
  - Assembly/erection/inspection

Control Issues (for orienting mirrors and radio telescope dishes)
- Control—structure interactions
- Transients and damping in structures optimized for ⅙ *g*
- Experiments and tests of control mechanics

Testing Issues
- Ground testing on Earth vs. on Moon
- Scaling of terrestrial structures tests to larger structures at ⅙ *g*
- Measurements/instrumentation for terrestrial/lunar use

TABLE 7. Technology development for millimeter and submillimeter astronomy.

- Surface accuracies—10 μm rms or less
- Precise demountable panels
- Stable frameworks
- Easily transportable pieces
- Disassemble/reassemble without loss of accuracy
- Means for adjustments
- Mounts with pointing accuracies better than 6-10 arcsec; tracking of 1-2 arcsec or better
- Foundations in lunar regolith

OPTICAL SYSTEMS TECHNOLOGY DRIVERS

There are many technology drivers for these optics. They include optical coatings that resist delamination, optics that are stress free after manufacture, and refractive materials that do not

darken or develop color centers. Refractive materials should have low scatter. Adaptive optics will be important for lunar optical telescope applications. Actuator and controls development and power and thermal control for adaptive optics should be pursued.

For mirrors on the lunar surface, active cleaning and contamination control techniques will be needed. Polishing techniques need to be improved; renewable coatings may be required. Materials used for telescopes need to be thermally stable. The appropriate degree of coating hardness against the ultraviolet and X-ray environments of the lunar surface will be needed. As always, the telescope optics will require the necessary vibration isolation.

PARABOLIC ANTENNA TECHNOLOGIES

Conventional techniques of maintaining the shape of steerable parabolic antennas rely on the use of low expansivity materials and the maintenace of isothermal conditions. As antenna size increases, the conventional approach becomes more difficult, and new materials and designs (*Akgul et al.,* 1990) for assembling, testing, deploying, and stabilizing structues with low natural frequencies (0.01 Hz) will be needed. Over system lifetimes, acceptable performance may depend upon the ability to control antenna shape by means of adaptive mechnical or electronic compensation.

Material systems should be developed to function as protective shields for antenna structures (and mirrors) against the worst extremes of the lunar thermal environment and the micrometeoroid environment.

COMMUNICATION AND CONTROL TECHNOLOGIES

There are many requirements on the communication system for the lunar astronomical observatory. Communication satellites in lunar orbit are needed. At the observatory site on the farside of the Moon, communication antennas will be needed for uplink and downlink that are high-gain, lightweight, and have low power consumption. Frequency and bandwidth selection for communications must be compatible with radio astronomy operations. Figure 1 illustrates the growing interference problem noted at the Very Large Array (VLA) west of Socorro, New Mexico.

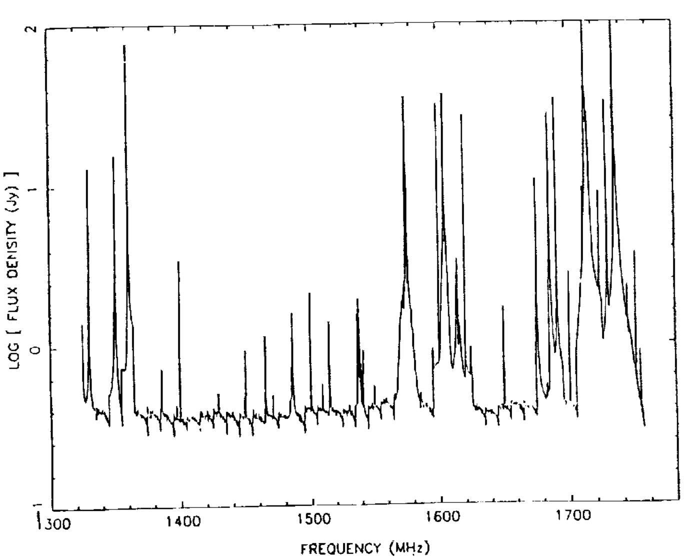

Fig. 1. Radio frequency interference is an increasing problem at the Very Large Array (VLA) west of Socorro, New Mexico, even though the frequency ranges of particular interest are protected by international agreement. The spikes of the man-made interference noted on the figure are of such magnitude to overwhelm signals from radio frequency sources in space. Technology advances should be focused on preventing such interference (from any man-made source) at radio astronomy sites on the farside of the Moon.

CONCLUSION

The need for all the observatories under consideration and for all extraterrestrial facilities is to engineer them with technologies that make it possible to perform well for long periods of time with minimal intervention by humans or robots. Better astronomy can be done if contamination and interference (gases, particulates, ground shock, and extraneous RF radiation) resulting from nearby operations can be kept to very low levels by limiting the need for nearby operations. An obvious need is to strive for facilities compatibility in lunar surface operations at various sites by controlling and reducing functions (e.g., proximity of mining operations or rocket launch pads to optical astronomy facilities) that lead to undesirable consequences. This need for compatibility implies the enforcement of a broad-based systems engineering discipline to all lunar engineering, construction, and operations.

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REFERENCES

Akgul F., Gerstle W., and Johnson S. (1990) Computer-aided structural design of a lunar radio telescope. In *Engineering, Construction and Operations in Space II* (S. W. Johnson and J. P. Wetzel, eds.), pp. 697-706. American Society of Civil Engineers, New York.

Burke B. F. (1985) Astronomical interferometry on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 281-291. Lunar and Planetary Institute, Houston.

Burke B. F. (1990) Astrophysics from the Moon. *Science, 250,* 1365-1370.

Burns J. O. (1985) A Moon-Earth radio interferometer. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 293-300. Lunar and Planetary Institute, Houston.

Burns J. O. (1988) 21st century astronomical observatories on the Moon. In *Engineering, Construction, and Operation in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 1083-1094. American Society of Civil Engineers, New York.

Burns J. O., Dunic N., Johnson S., and Taylor G. J., eds. (1989) *A Lunar Far-side Very Low Frequency Array.* NASA CP-3039. 78 pp.

Burns J. O., Duric N., Taylor G. J., and Johnson S. W. (1990) Astronomical observations on the Moon. *Sci. Am., 262,* 42-49.

Burns J. O., Fernini I., Sulkanen M., Duric N., Taylor, J. and Johnson S. (1992) Artificially generated atmosphere near a lunar base. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Douglas J. N. and Smith H. J. (1985) A very low frequency radio astronomy observatory on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 301-306. Lunar and Planetary Institute, Houston.

Johnson S. W. (1988) Engineering for a 21st century lunar observatory. *J. Aerosp. Eng., 1,* 35-51.

Johnson S. W. and Leonard R. S. (1988) Engineering test support for construction in space. *J. Aerosp. Eng., 1,* 206-218.

Johnson S. W., Taylor G. J., and Wetzel J. P. (1992) Environmental effects on lunar astronomical observatories. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Sovie R. J. (1988) Power systems for production, construction, life support and operations in space. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 928-939. American Society of Civil Engineers, New York.

ENVIRONMENTAL EFFECTS ON LUNAR ASTRONOMICAL OBSERVATORIES

N93-17454

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The Moon offers a stable platform with excellent seeing conditions for astronomical observations. Some troublesome aspects of the lunar environment will need to be overcome to realize the full potential of the Moon as an observatory site. Mitigation of negative effects of vacuum, thermal radiation, dust, and micrometeorite impact is feasible with careful engineering and operational planning. Shields against impact, dust, and solar radiation need to be developed. Means of restoring degraded surfaces are probably essential for optical and thermal control surfaces deployed in long-lifetime lunar facilities. Precursor missions should be planned to validate and enhance the understanding of the lunar environment (e.g., dust behavior without and with human presence) and to determine environmental effects on surfaces and components. Precursor missions should generate data useful in establishing keepout zones around observatory facilities where rocket launches and landings, mining, and vehicular traffic could be detrimental to observatory operation.

INTRODUCTION

The Moon's environment makes it an excellent place from which to make astronomical observations (*Burns and Mendell,* 1988; *Burns et al.,* 1990). Recent papers (*Johnson and Wetzel,* 1990) have considered the science, engineering, and construction associated with lunar astronomical observatories. Some of the environmental factors that make the Moon a useful platform for astronomy, however, are not benign and will require special efforts to mitigate their effects. This paper reviews the environmental factors likely to cause degradation of the components and systems of astronomical facilities on the Moon, summarizes results of studies of spacecraft exposed to the lunar environment, and presents a preliminary assessment of ways to diminish the damaging effects of the space environment.

SPACE ENVIRONMENTAL FACTORS

In this section, we summarize the features of the lunar environment that seem most troublesome to the longevity and operation of astronomical facilities on the Moon. Some environmental characteristics, such as a low magnetic field (10^{-2} to 10^{-4} Earth's field at the equator) and a seismically stable surface will not lead to degradation of equipment and will not be discussed. Details of these and other characteristics of the Moon's surface environment are given by *Taylor* (1988). Some environmental factors of the low Earth orbit (LEO) environment, which may provide additional insight into the lunar environment, are also discussed.

Atmosphere

The Moon has an extremely tenuous atmosphere. At night, it contains only 2×10^5 molecules/cm^3 (*Hoffman et al.,* (1973), giving a pressure of 10^{-12} torr. This hard vacuum will create problems with outgassing of materials and causes solar and cosmic radiation and micrometeorites to hit the lunar surface unimpeded, as discussed below. The nighttime atmosphere is composed chiefly of H and noble gases (*Hoffman et al.,* 1973). Measurements were not made during the lunar daytime by Apollo instruments, but slight enhancements of CO_2 and CH_4 just before sunrise (*Hoffman and Hodges,* 1975) suggest that these gases dominate the atmosphere during the daytime (*Hodges,* 1976).

The atmosphere in LEO is quite different from that of the Moon. The presence of atomic oxygen in LEO creates a difficult degradation problem, as was observed from the components of the Solar Maximum satellite (SMS) that were returned by the space shuttle (*Liang et al.,* 1985). Orbiting space debris (paint chips, etc.) also create problems for satellites in LEO (*Kessler,* 1985; *Barrett et al.,* 1988). Note that orbiting space debris and highly oxidizing gases, such as atomic oxygen, that are present in LEO are absent on the Moon.

Surface Temperatures

The Moon's surface undergoes a drastic thermal cycling from dawn to noon. The surface temperature is a function of the amount of incident solar radiation, the amount reflected off the lunar surface (only about 7%), and the amount radiated in the infrared. At the Apollo 17 site, for example, located about 20° north of the equator, the temperature ranged from 384 K to 102 K during the month-long lunar day (*Keihm and Langseth,* 1973). Furthermore, the temperature decreases rapidly at sunset, falling about 5 K/hr. In polar regions, the predawn temperature is about 80 K (*Mendell and Low,* 1970), and in permanently shadowed areas near the poles the temperature is even lower. The large range in temperature and rapid change at sunset could affect many structures and materials.

Radiation

Because of the lack of an absorbing atmosphere and, for charged particles, the small magnetic field, radiation from the sun and galaxy hit the lunar surface unimpeded. Sunlight provides one damaging type of radiation: ultraviolet light. The sun's spectrum peaks in the visible, at about 0.5 μm, but a significant amount of it, 7%, is between 0.28 and 0.40 μm (*Robinson,* 1966). Since the solar constant is 1393 W/m^2 at the Earth-Moon distance from the sun (*Coulson,* 1975), the total ultraviolet flux is about 95 W/m^2.

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There are three sources of charged-particle radiation with different energies and fluxes: (1) high-energy (1-10 GeV/nucleon) galactic cosmic rays, with fluxes of about 1/cm^2/sec and penetration depths up to a few meters; (2) solar flare particles with energies of 1-100 MeV/nucleon, fluxes up to 100/cm^2/sec, and penetration depths of about 1 cm; and (3) solar wind particles, which have much lower energies (1000 eV), small penetration depths, but high fluxes (10^8/cm^2/sec). These penetration depths refer to the primary particles only. Reactions between high-energy particles and lunar materials cause a cascade of radiation that penetrates deeper (*Silberberg et al.*, 1985), up to several meters for cosmic rays and solar flares. Although solar wind particles have low energies, their high flux might make them capable of damaging materials on the lunar surface. The more energetic radiations could damage electronic equipment.

Micrometeorites

The tenuous lunar atmosphere allows even the smallest micrometeorites to impact with their full cosmic velocity, which is 10 km/sec, though some arrive at >50 km/sec (*Berg and Grun*, 1973). This rain of minute projectiles poses a hazard to all surfaces exposed on the lunar surface, but it presents a serious threat to delicate materials such as telescope mirrors and coatings.

Almost all lunar rock surfaces that were exposed to space contain numerous microcraters. Studies of lunar rocks (e.g., *Fechtig et al.*, 1974) have revealed the average flux during the past several hundred million years. However, data from the Surveyor III TV camera shroud returned by the Apollo 12 mission and study of Apollo windows (*Cour-Palais*, 1974) indicate that the present flux of particles $<10^{-7}$ g, which are capable of making craters up to 10 μm across, is about 10 times greater than that measured on lunar rocks. Study of louver material from the SMS (*Barrett et al.*, 1988) confirmed that fluxes are greater now than the average of the past several hundred million years. Combining the fluxes of particles $<10^{-7}$ g measured on spacecraft with those $>10^{-7}$ measured on Apollo rocks, we arrive at the flux estimates in Table 1.

These fluxes are clearly high enough to damage telescope mirrors, but they apply to 2π geometry. A telescope shielded within a collimator would be exposed to a lower flux. For example, a telescope mirror 1 m across located at the base of a 1-m tube would be exposed to only 29% of the direct flux. A tube 3 m long would decrease the flux to 5% of the values listed in Table 1. Figure 1 demonstrates quantitatively how the direct flux is decreased by using a collimator tube for shielding. Even long tubes, however, still allow substantial numbers of micrometeorites to strike an unprotected surface, and there is an additional source of impact-derived debris due to secondary impact events caused by ejecta of primary events. These not only make craters, but also commonly cause deposition of accretionary spatter (*Zook*, 1978).

TABLE 1. Microcrater product rates on the Moon.

Crater diameter (μm)	Craters/m^2/yr
>0.1	300,000
>1	12,000
>10	3,000
>100	0.6
1000	0.001

Values are estimated from data given by *Fechtig et al.* (1974), *Cour-Palais* (1974), and *Barrett et al.* (1988).

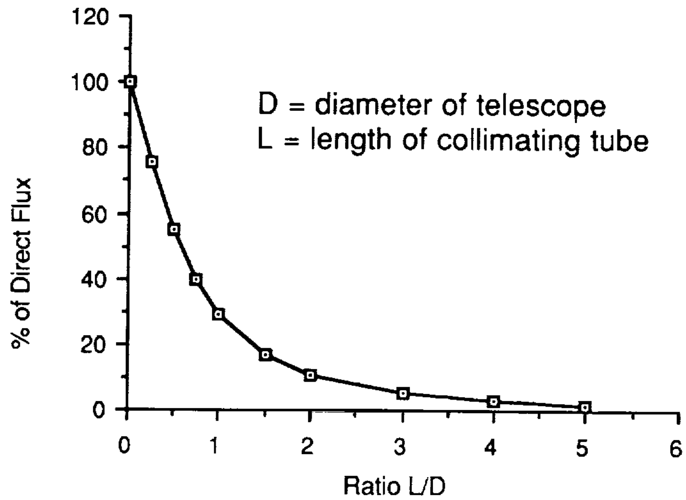

Fig. 1. Plot of the percent of direct flux of micrometeorites reaching a telescope surface of diameter D as a function of the length L of a collimating tube.

As alluded to above, micrometeorites create a degradation problem in LEO as well as on the lunar surface. The lack of atmosphere in Earth orbit allows micrometeorites to impact unrestrained, as in the case of the lunar surface environment. Measurements acquired from the study of returned components from the SMS (*Schramm et al.*, 1985; *Kessler*, 1985; *Barrett et al.*, 1988) indicate that unprotected surfaces are very susceptible to micrometeorite damage. Schramm indicates that the exterior insulation blankets returned from SMS inadvertently acted as micrometeorite capture devices. These results indicate the need for protective coatings or temporary covers for long durations in the space (orbital or lunar surface) environment.

Dust

The lunar surface is covered with a global veneer of debris generated from underlying bedrock by meteorite impacts. This material, called the lunar regolith, contains rock and mineral fragments and glasses formed by melting of soil, rock, and minerals. Its mean grain size ranges from 40 to 268 μm and varies chaotically with depths (*Heiken*, 1975). In most samples returned by Apollo and Luna missions, about 25 wt% of the rgolith is <20 μm in size and about 10 wt% is <10 μm. In short, the lunar surface is dusty, and optical equipment must be protected from contamination and subsequent damage by dust particles.

Dust could be thrown onto mirror surfaces by artificial means such as rocket launches, surface vehicles, or astronaut suits. This man-made degration problem is one that can (and should) be controlled with proper regulations and procedures, which are discussed in more detail later in this paper. An unknown amount of dust might be transported by charge differences built up by photoconductivity effects near the day-night terminator. *Criswell* (1972) described a bright glow photographed by Surveyor 7 and explained the phenomenon as levitation of dust grains about 5-10 μm in radius. The grains were lifted only 3-30 cm above the local horizon and had a column density of 5 grains/cm^2. How effective this mechanism is needs to be tested by measurements on the lunar surface.

DEGRADATION OF MATERIALS AND SYSTEMS

Investigations of Surveyor Components

Surveyor III components were studied on Earth after these parts had been exposed to the lunar environment for 31 months (roughly 32 lunar days) from April 20, 1967 until November 20, 1969. Parts studied were (1) the television camera, which included optics, electronics, cables, and support struts; (2) the scoop portion of the soil mechanics surface sampler device (which contained over 6 g of lunar soil); (3) a section of polished aluminum tube 19.7 cm long; and (4) a section of cabling and painted aluminum tube (*Nickle,* 1971; *Carroll et al.,* 1972).

These parts were analyzed for surface changes and characteristics (e.g., adherence of soil particles, sputtering, and UV-induced degradation of thermal control coatings), micrometeorite impacts, radiation damage, particle tracks, and naturally induced radioactivity.

Although the Surveyor III was on the lunar surface for 31 months, it was operated for only 2 weeks. It experienced 30½ months' exposure in a dormant or nonoperating state. Involved were 1500 resistors, capacitors, diodes, and transistors in the camera returned to Earth. Tests after recovery verified the integrity of most parts after 31 months on the Moon (*Carroll et al.,* 1972). A few components failed, apparently because of thermal cycling to very low temperatures (e.g., a tantalum capacitor) and as a result of thermal strain (e.g., glass envelopes). Some failures caused a cascade of failures. For example, a failure of the circuit that drove the shutter was caused by the failure of a transistor that had been degraded in a preflight test; this caused failure of a shutter solenoid, which in turn caused evaporation of a photoconductor in the vidicon as a result of the shutter being open (*Carroll and Blair,* 1972).

Solar radiation and effects. The maximum time of exposure to solar radiation during the time the retrieved parts were on the lunar surface is theoretically 10,686 hours. Shadowing effects limited actual exposure times to considerably less than the theoretical maximum. It was estimated, for example, that the clear optical filter on the camera had a total exposure of only 4180 hours, but that the scoop arm, which had been left fully extended at maximum elevation in 1967 at the Surveyor mission termination, had a total exposure of 9078 hours.

As the evaluation of Surveyor III parts was in progress, the tan color of the originally white paint faded due to photobleaching. Photobleaching of induced optical damage can also occur. Therefore, hardware must be sampled and returned carefully to avoid or account for subsequent alteration in the terrestrial laboratory environment (*Carroll and Blair,* 1972). Although some environment-induced failures occurred, it is clear from the superb results obtained by most Apollo Lunar Surface Experiments Packages (ALSEP) experiments that it will be possible to produce systems that will function through many lunations.

Degradation of thermal control coatings. Coatings exposed to the space environment exhibit radiation-induced darkening that increases with time. After 31 months on the Moon, inorganic coatings originally white were tan in appearance. This discoloration was observed to be in a pattern consistent with the amount of irradiation received (*Carroll and Blair,* 1972). Overall discoloration patterns were the result of several effects attributable to solar radiation (e.g., in the ultraviolet), lunar dust, and products of organic outgassing from spacecraft parts (*Carroll and Blair,* 1972). Dust and irradiation played the key roles in altering the appearance (and usefulness) of the surface coatings.

The blue color of the scoop faded to a whitish blue. The surfaces painted with inorganic white degraded from a solar absorptance of 0.2 to 0.38 up to 0.74, depending on orientation. Polished aluminum tubes rose in absorptance from 0.15 to 0.26 (on a "clean" or relatively dust-free surface) to 0.75 where dust was present (*Anderson et al.,* 1971).

The greatest changes in reflectance were for shorter (0.6 to 1.0 μm) as opposed to longer wavelengths (up through 2.0 or 2.4 μm). Both solar radiation and dust were instrumental in decreasing reflectance.

Dust presence. It was estimated that the upper portion of the clear filter, which was positioned over the Surveyor camera lens by remote command at the close of the Surveyor III mission, had 25% of its surface area covered by particulate material. This fine-grained lunar soil had a median grain size of 0.8 μm and ranged up to 15 μm in size (*Nickle,* 1971). Dust on the Surveyor mirror was thought to have caused a marked loss of contrast in relayed pictures during the performance of the Surveyor mission (*Carroll and Blair,* 1972). "Lunar material, even in small quantities, can have a significant effect on temperature control and optical performance of hardware on the lunar surface" (*Carroll and Blair,* 1972). Even 10^{-5} to 10^{-4} g of lunar fines per square centimeter can increase absorbed solar thermal energy for a reflective thermal-control surface by a factor as large as 2 or 3 (*Carroll and Blair,* 1972). On the other hand, there are no reports of degradation of the laser reflectors left by three Apollo missions.

Sources of dust. There was dust on the returned Surveyor III television camera attributable to one or more of five sources (*Carroll and Blair,* 1972): (1) The disturbance of the soil during the Surveyor III landing, accentuated by the vernier descent engines continued thrusting during two rebounds from the lunar surface; (2) disturbance mechanisms operating on the Moon (e.g., meteroid impact and electrostatic charging); (3) Apollo 12 lunar module approach and landing; (4) operation of the scoop on the Moon; and (5) retrieval and return to Earth by Apollo 12 astronauts.

The Surveyor III and lunar module (LM) landings were probably the most significant sources of the dust found on the camera. The descent engine, which disturbed the dusty surface over the last 1000 ft of its ground track before landing 155 m away, was probably the most significant dust source. Dust was accelerated by the LM rocket plume to velocities in excess of 100 m/sec. This accelerated dust literally sandblasted the Surveyor III and removed much discolored paint (*Cour-Palais et al.,* 1972).

Erosion surfaces in the lunar environment. Three processes may be considered in evaluating erosional effects on parts exposed to the lunar environment: (1) sputtering of individual atoms by the solar wind (mainly hydrogen); (2) damage from solar flare heavy nuclei (e.g., Fe); and (3) micrometeorite impact (*Baber et al.,* 1971).

Estimated erosion rates per year from these effects are very small (e.g., 0.4 Å for sputtering, 0.1 to 0.4 Å for heavy nuclei, and 1-2 Å for micrometeorite impacts). Micrometeorite impact is probably the most significant mechanism of the three for degradation of telescope optical surfaces, although the effects of sputtering on optical coatings over several years requires a restorative capability or replacement.

Results of examinations for micrometeroid impacts. The television camera shroud, the camera's optical filters, and a piece

of aluminum tube were scanned for possible craters resulting from micrometeorite impacts. Magnifications in the range of 25× to 40× and greater were used over substantial portions of the surfaces of these objects as the search for impact craters proceeded (*Cour-Palais et al.,* 1971; *Brownlee et al.,* 1971).

No hypervelocity impact craters were identified in the original studies on the 0.2 sq m of the shroud or on the optical filters. Five craters ranging in diameter from 130 to 300 μm were noted as having a possible hypervelocity impact origin. The many other craters found were thought to have originated as a result of impact of low-velocity debris accelerated by the LM descent engine plume. However, continued study of the Surveyor materials and of impact pits on lunar rocks led to a reevaluation of the original Surveyor data (*Cour-Palais,* 1974), which indicated that most of the craters on the returned material were hypervelocity impact pits. Nevertheless, damage from low-velocity impact was still substantial.

Buvinger (1971) performed an investigation by electron replication microscopy of two sections of the unpainted aluminum tubing. Erosion damage apparently resulted from impact of soil particles during landing maneuvers. Some pits in the approximately 1-mm range had some characteristics of hypervelocity impacts. Solar-wind sputtering apparently had little effect on the tube, and damage by particle impact was apparently by lower-velocity particles and limited to a depth no greater than 2 mm.

Investigations of LEO Satellites

Degradation studies of satellite components returned from LEO have been conducted. The space shuttle, or space transportation system (STS), with its reusable capability to be launched into orbit and return, has created the potential to go into space and repair satellites, and return components or even entire satellites. The STS has been used to perform a repair mission on the SMS (*SMRM,* 1985) and to retrieve two Hughes communication satellites, Palapa and Westar. The Long Duration Exposure Facility (LDEF) was placed into orbit by the shuttle for planned retrieval 12 months later. Many of the experiments on LDEF incorporated studies on space degradation. This section summarizes some of the degradation studies that have been conducted on LEO satellite components and relates their possible implications to the lunar environment.

Investigations of SMS components. The SMS was launched in February, 1980, into a 310 n.m. (674 km) circular orbit, with solar flare research as its primary objective. Between 6 and 10 months after launch, the satellite suffered a series of failures with the attitude-control system, rendering several of the instruments inoperable and some others at limited capability. The Solar Maximum Recovery Mission (SMRM) was performed in April, 1984. The Modular Attitude Control System (MACS) module, the Main Electronics Box (MEB), and their associated thermal blankets were replaced with new units, and the old units were returned to Earth for investigation following more than four years in LEO (*SMRM,* 1985). The flight electronics parts showed no adverse effects from the LEO radiation environment. In general, the components returned from the SMS were in good condition.

Analyses were performed on the materials retrieved from the SMS thermal control system. The presence of atomic oxygen caused most of the degradation of the materials. Fortunately, atomic oxygen, a major problem in LEO, is absent in the lunar environment, and will not be discussed in detail here.

Analysis of the multilayer insulation (MLI) blankets indicated that micrometeorite and debris impacts had caused hundreds of impact craters. Seventy-micrometer craters formed complete holes through the 50-μm-thick initial layer of thermal blanket. Roughly 160 of these craters penetrated the surfaces, which encompassed an area of 0.153 sq m (*Kessler,* 1985). This high micrometeorite flux demonstrates the importance of protecting components and systems exposed to the space environment. *Schramm et al.* (1985) indicated that the MLI blankets acted inadvertently as a micrometeorite capture device. This indicates the potential benefits gained by using protective coatings and covers.

In the lunar environment, any astronomical observatory, especially the delicate optical equipment and sensors, will need to be protected from the micrometeorite environment. Much can be gained from the study of the micrometeorite environment in LEO. Information gathered can be used to examine better ways to protect systems on the lunar surface.

Other LEO investigations. As we noted above, the STS retrieved two Hughes Communication satellites, Palapa B-2 and Westar VI, and returned them to Earth in 1984. The two spacecraft were only in orbit for eight months and there were no detailed degradation investigations conducted on the satellites (M. West, personal communication, 1987).

The LDEF was launched from the STS in 1984 with a planned retrieval 12 months later. This retrieval effort was delayed until 1990. LDEF was designed to accommodate a large number of science and technology experiments, many of which were designed to study space degradation (*Clark et al.,* 1984). There will be a vast amount to be learned about degradation in space from the study of the experiments.

Impact and debris studies have been conducted on the shuttle and Apollo/Skylab where impact craters have been found. However, these experiments and studies had either short exposure times or no conclusive technique to differentiate orbital debris from micrometeorites (*Kessler,* 1985). The detection of orbital debris is receiving an increasing amount of attention, and in the next few years both specially designed radars and experiments carried on the shuttle will produce new data on both orbital debris and the micrometeoroid flux.

MITIGATION OF DEGRADATION

As *Carroll et al.* (1972) note, "The need to protect optical elements from dust contamination was obvious during Surveyor III lunar operations in 1967 and was confirmed during analysis of returned hardware. All other optical performance information gained from post-mortem analysis is secondary to this conclusion."

Observatory design and operation can mitigate and compensate for the potentially detrimental effects of solar radiation, dust accumulation, surface erosion, changes in thermal control coatings, and micrometeorite impacts. We outline below some ideas for blunting the hazardous effects of the lunar environment.

Dust Mitigation

Rocket landing and ascent operations can be performed at locations sufficiently far removed from observatory sites to prevent dust erosion and accumulation on optics, antennae, and thermal control surfaces. Shielding against dust driven by rocket plumes may be useful. How great the required keep-out distances or shielding heights against accelerated dust must be depends on the rocket engine and plumes. Keep-out distances may be in excess of 1000 ft based on the extent of LM descent engine sand blasting effects, dust disturbance, and deposition on Surveyor III components.

H. Schmitt (personal communication, 1988) suggested that optics be provided with lens caps that could be remotely controlled to cover and protect optical surfaces before permitting construction and repair teams to approach observatories on the Moon. He notes that the lunar dust is difficult to avoid in astronaut and vehicular traffic on the Moon.

Preserving Thermal Control Surfaces

Some telescope components and other base facilities will be dependent for temperature control on the use of thermal control coatings designed to have appropriate values of absorptance and reflectance. If these coatings degrade—as was noted in the case of Surveyor III coatings—temperatures of critical components will deviate from specified values and diminish or negate observatory performance. Protecting coatings by use of layers that intercept UV radiation may help. More stable coatings applied under conditions avoiding contamination may also help.

Use of Shields

Shields against micrometeorite impact, dust particles, and solar radiation can be devised to reduce the probability of impact, contamination, or interference by stray light rays. Shields can reduce the probability of impact on optics by reducing the portion of the sky from which impacting particles can originate. Appropriate baffles can prevent the shield from directing stray or scattered light on mirrors or other optics.

Restoration

According to *Watson et al.* (1988), equipment for restoring coatings on telescope mirrors and thermal control surfaces has been developed and tested on orbit by the USSR. These metal coating operations were performed in space after extensive experimentation in ground-based laboratories to overcome technical difficulties associated with heating, vaporization, and deposition of aluminum. In 1975, cosmonauts Gubarev and Grecho were reported to have recoated the mirror of a solar telescope on the Salyut spacecraft. More coating restoration experiments on orbit were performed in subsequent spacecraft in 1979, 1980, and 1984. Details have not been made available, but results were reported as excellent. These coating-technology experiments suggest that the capability to restore optical and thermal control surfaces degraded by exposure to the space environment may be available for astronomical observatories on the Moon.

It has also been suggested that large mirrors for space use be composed of numerous replaceable segments so that if impact or abrasion causes damage, only the degraded portion need be replaced. Also, mirror surface coatings should be selected that are compatible with cleaning processes and reduce electric charge effects (*Bouquet et al.,* 1988).

Laboratory Investigations

Laboratory studies have played and continue to play an important role in estimating the degradation likely when components of space systems are exposed to the space environment. The thermal-vacuum test (*Flanagan,* 1986) will be an essential step in the development and preflight preparations for any observatory components to be deployed on the lunar surface. The systems will be subjected to vacuum and thermal cycling comparable to that found on the Moon to assure that they are capable of operating under very cold and very hot conditions and can accommodate large temperature gradients.

Vacuum chambers with thermal cycling can also include solar simulation that provides an approximation of the solar spectrum. Micrometeorite protection systems can be designed based on available laboratory data (e.g., from light gas guns and Van de Graff generators) and data gathered from recovered components (e.g., LDEF, SMS).

Precursor Missions

Plans to return to the Moon should include visits to at least one Apollo landing site to ascertain the degradation and changes in selected Apollo materials and components. Six Apollo landings were made between 1969 and 1972, and a wide range of equipment was left on the surface, including the descent stages of the LM, lunar roving vehicles (LRV), and the ALSEP. Items to be studied include thermal blankets, optics, retroreflectors (for laser ranging), batteries and motors (e.g., on the LRV), communications equipment such as parabolic dishes, various pieces of tankage, and test equipment.

These parts can be studied to ascertain the degradation caused by long-term exposure to micrometeorite bombardment, solar and cosmic radiation, thermal cycling, and vacuum. Areas for study are suggested by the previous experience with Surveyor hardware (*Scott and Zuckerman,* 1971; *Nickle and Carroll,* 1972). To be determined are dust and radiation darkening of surfaces, particle impact effects (both primary and secondary), and the effects of long-term thermal cycling in vacuum.

The goals of the visit and study will be to improve the technology for design, fabrication, and test of future lunar astronomical observatories (*Johnson,* 1988), enhance our understanding of processes that occur on the Moon and of the rates at which they operate, and to check the validity of accepted design approaches. Figure 2 demonstrates a generic representation of our need to better understand lunar environmental degradation (*Johnson and Wetzel,* 1988). As shown in this figure, we possess a very limited amount of experience with lunar surface degradation. We must gather additional information about degradation and its effects

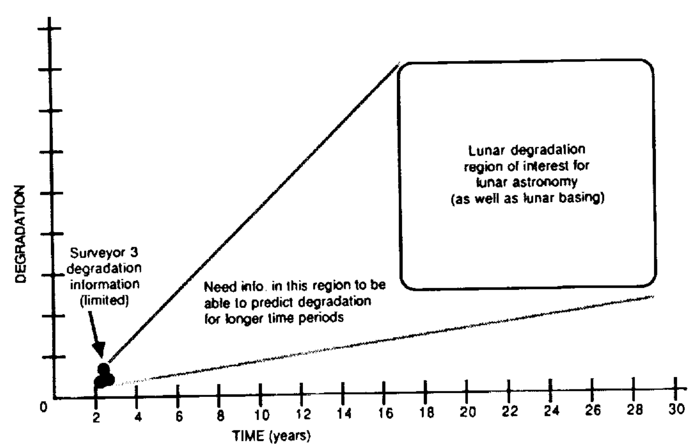

Fig. 2. Schematic representation of the information needed to investigate degradation on the lunar surface over a long period of time.

over a long period of time. For example, revisiting and studying the materials and equipment from the Apollo sites will allow us to acquire information about lunar degradation in the 30-year time range.

Examination of Apollo materials will be extremely valuable, but will leave many questions unanswered. Additional experiments will be required to fully understand micrometeorite impacts (both primary and secondary), dust levitation, and assorted operational disturbances.

Apollo materials will shed light on the present flux of micrometeorites and shrewd collection of surfaces shielded from direct impact will provide crucial information about the flux of and damage done by secondary projectiles. Nevertheless, an array of micrometeorite detectors, either passive or active, ought to be deployed on the lunar surface to obtain information on fluxes, masses, velocities, and directions of impacting particles. A device of this sort was emplaced during the Apollo 17 mission (*Berg et al.,* 1973). Furthermore, instruments like this will be developed for use on the space station. In addition to supplementing data that will be obtained from study of surfaces of the Apollo spacecraft and instruments, the new generation of lunar surface micrometeorite detectors will provide up-to-date data and a basis for comparison with detectors in LEO. This will help establish the natural flux in LEO, a critical parameter to know if we are to accurately monitor the growth of man-made debris in LEO.

As noted earlier, *Criswell* (1972) suggested that a brightening at the horizon in Surveyor photographs taken shortly after sunset was caused by electostatic effects. The idea is that electrons are removed by the photoelectric effect when sunlight strikes the surface. This results in a charge inbalance with the uncharged surroundings, causing small grains to be lifted off the ground. It seems prudent to determine the extent to which this process operates and assess whether it will interfere with lunar surface operations. It might, for example, cause micrometer-sized dust grains to be deposited on telescope mirrors, thereby degrading astronomical observations. An active detector designed to measure the flux and size distribution of low-velocity dust grains could provide the necessary information.

It will also be necessary to monitor disturbance caused by lunar base operations. This includes dust raised by rockets landing and taking off, vehicles moving, and astronauts walking. For example, if astronauts are needed to service telescopes, one must know how much dust could be transferred from their spacesuits onto a mirror. Perhaps this could be measured by having astronauts approach a low-velocity dust detector. If significant dust were measured, other means of servicing telescopes would have to be devised. Disturbance by the transportation system could also be monitored by an array of dust detectors. Effects of the lunar base operations on the present lunar atmosphere should also be monitered (*Fernini et al.*, 1990).

SUMMARY AND CONCLUSIONS

Although the Moon is an excellent place for astronomy, special efforts will be required to mitigate or compensate for detrimental effects of the lunar environment on observatory components. The most troublesome characteristics of the lunar environment are the vacuum (which leads to outgassing), solar and cosmic radiation, micrometeorite impacts, the surface temperature regime, and the ubiquitous dust particles.

Valuable information on degradation of parts and systems in the lunar environment was obtained by retrieval to Earth and careful analysis of Surveyor III components. These components had been on the Moon nearly 32 lunar days from April, 1967, to November, 1969. Most parts retained their integrity, but a few failed (e.g., because of thermal cycling). Degradation of coatings also occurred, primarily because of ultraviolet radiation and the static and dynamic effects of dust particles on optical and thermal-control surfaces. The dust can cause scattering of light and loss of contrast in optical trains.

Several approaches can be taken to mitigate the negative effects of the lunar environment on astronomical observatory components. First, an effort is needed to better understand and model the degradation mechanisms. This effort should be addressed early in precursor missions to the Moon. Second, operational rules will be necessary to confine activities that generate dust and rocket plumes to zones outside those where astronomical observatories are being used. When it is necessary to approach the observatory sites with vehicles and construction or maintenance teams, precautionary shielding should be activated to protect optics and reduce deposition on thermal-control surfaces. Processes will eventually be needed to clean and restore dusty and impact-damaged surfaces. Fortunately, the lunar environment, although dusty, lacks the hazards in LEO associated with atomic oxygen and orbiting debris, such as chips of paint, from previous missions.

Although the lunar thermal regime offers a severe test of observatory components, careful engineering can control degradation, and the number of cycles to be endured (about one per month) is much fewer than cycles encountered in LEO (about 480 per month). The environment on the lunar surface is conducive to the use of shields and baffles against micrometeorite impact, dust particles, and solar radiation. Experiments in terrestrial laboratories and precursor missions to the Moon are needed to assist in predicting degradation and in reducing its ravaging effects on future lunar astronomical observatories. Restoration processes should be developed to enhance the longevity of observatory components on the Moon. The technology of degradation mitigation that will be developed will apply not only to astronomical observatories, but also to a wide range of lunar base elements. It is prudent to initiate studies of lunar environmental effects early so that beneficial results can be implemented early in the planning of all lunar base facilities.

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REFERENCES

Anderson D. L., Cunningham B. E., Dahms R. G., and Morgan R. G. (1971) X-ray probe, SEM, and optical property analysis of the surface features of Surveyor III materials. *Proc. Lunar Sci. Conf. 2nd*, pp. 2753-2765.

Barber D. J., Cowsik R., Hutcheon I. D., Price P. B., and Rajan R. S. (1971) Solar flares, the lunar surface, and gas-rich meteorites. *Proc. Lunar Sci. Conf. 2nd*, pp. 2705-2714.

Barrett R. A., Bernhard R. P., and McKay D. S. (1988) Impact holes and impact flux on returned Solar Max louver material (abstract). In *Lunar and Planetary Science XIX*, pp. 39-40. Lunar and Planetary Institute, Houston.

Berg O. E. and Grün E. (1973) Evidence of hyperbolic cosmic dust particles. In *Space Research—XIII*, pp. 1047-1055.

Berg O. E., Richardson F. F., and Burton H. (1973) Lunar ejecta and meteorites experiment. In *Apollo 17 Preliminary Science Report*, pp. 16-1 to 16-9. NASA SP-330.

Bouquet F. L., Maag C. R., and Heggen P. M. (1988) *Design Considerations for Long-Lived Glass Mirrors for Space.* NASA JPL Invention Report NPO-17047/6564. 8 pp.

Brownlee D., Boucher W., and Hodge P. (1971) Micrometeoroid flux from Surveyor glass surfaces. *Proc. Lunar Sci. Conf. 2nd,* pp. 2781-2789.

Burns J. O. and Mendell W. W. (1988) *Future Astronomical Observatories on the Moon.* NASA CP-2489. 134 pp.

Burns J. O., Duric N., Taylor G. J., and Johnson S. W. (1990) Astronomical observatories on the Moon. *Sci. Am., 262,* 42-49.

Buvinger E. A. (1971) Replication electron microscopy on Surveyor III unpainted aluminum tubing. *Proc. Lunar Sci. Conf. 2nd,* pp. 2791-2795.

Carroll W. F. and Blair P. M. (1972) Spacecraft changes, Part A, lunar dust and radiation darkening of Surveyor 3 surfaces. In *Analysis of Surveyor 3 Material and Photographs Returned by Apollo 12,* pp. 23-29. NASA, Washington, DC.

Carroll W. F., Davis R., Goldfine M., Jacobs S., Jaffee L. D., Leger L., Milwitzky B., and Nickle N. L. (1972) Introduction. In *Analysis of Surveyor 3 Material and Photographs Returned by Apollo 12,* pp. 15-21. NASA, Washington, DC.

Clark L., ed. (1984) *The Long Duration Exposure Facility (LDEF): Mission 1 Experiments.* NASA SP-473, NASA, Washington, DC.

Coulson K. L. (1975) *Solar and Terrestrial Radiation.* Academic, New York. 322 pp.

Cour-Palais B. G. (1974) The current micrometeoroid flux at the moon for masses $<10^{-7}$ g from the Apollo window and Surveyor 3 TV camera results. *Proc. Lunar Sci. Conf. 5th,* pp. 2451-2462.

Cour-Palais B. G., Flaherty R. E., High R. W., Kessler D. J., McKay D. S., and Zook H. A. (1971) Results of Surveyor III sample impact examination conducted at the Manned Spacecraft Center. *Proc. Lunar Sci. Conf. 2nd,* pp. 2767-2780.

Cour-Palais B. G., Flaherty R. E., High R. W., Kessler D. J., McKay D. S., and Zook H. A. (1972) Results of examination of the returned Surveyor 3 samples for particulate impacts. In *Analysis of Surveyor 3 Material and Photographs Returned by Apollo 12,* pp. 158-157. NASA, Washington, DC.

Criswell D. R. (1972) Lunar dust motion. *Proc. Lunar Sci. Conf. 3rd,* pp. 2671-2680.

Fechtig H., Hartung J. B., Nagel K., and Neukum G. (1974) Lunar microcrater studies, derived meteoroid fluxes, and comparison with satellite-borne experiments. *Proc. Lunar Sci. Conf. 5th,* pp. 2463-2474.

Fernini I., Burns J. O., Taylor G. J., Sulkanen M., Duric N., and Johnson S. W. (1990) Dispersal of gases generated near a lunar outpost. *J. Spacecr. Rockets, 27,* 527-538.

Flanagan G. F. (1986) Hubble Space Telescope ground testing. In *International Test and Evaluation Association 1986 Symposium Proceedings,* pp. 68-90. Huntsville, Alabama.

Heiken G. (1975) Petrology of lunar soils. *Rev. Geophys. Space Phys., 13,* 567-587.

Hodges R. R. Jr. (1976) The escape of solar-wind carbon from the moon. *Proc. Lunar Sci. Conf. 7th,* pp. 493-500.

Hoffman J. H. and Hodges R. R. Jr. (1975) Molecular gas species in the lunar atmosphere. *Moon, 14,* 159-167.

Hoffman J. H., Hodges R. R. Jr., and Johnson F. S. (1973) Lunar atmospheric composition results from Apollo 17. *Proc. Lunar Sci. Conf. 4th,* pp. 2865-2875.

Johnson S. W. (1988) Engineering for a 21st century lunar observatory. *J. Aerosp. Engr., 1,* 35-51.

Johnson S. W. and Wetzel J. P. (1988) Advanced technology for a lunar astronomical observatory. In *Engineering, Construction, and Operations in Space,* pp. 1102-1113. American Society of Civil Engineers, New York.

Johnson S. W. and Wetzel J. P., eds. (1990) *Engineering Construction and Operations in Space II,* pp. 666-737. American Society of Civil Engineers, New York.

Keihm S. J. and Langseth M. G. Jr. (1973) Surface brightness temperatures at the Apollo 17 heat flow site: Thermal conductivity of the upper 15 cm of regolith. *Proc. Lunar Sci. Conf. 4th,* pp. 2503-2513.

Kessler D. J. (1985) Examination of returned Solar-Max surfaces for impacting orbital debris and meteorites. In *Proc. SMRM Degradation Study Workshop,* pp. 245-246. Goddard Space Flight Center, Greenbelt, MD.

Liang R. H., Oda K. L., and Chung S. Y. (1985) Degradation studies of SMRM teflon. In *Proc. SMRM Degradation Study Workshop,* pp. 337-342. Goddard Space Flight Center, Greenbelt, MD.

Nickle N. L. (1971) Surveyor III material analysis program. *Proc. Lunar Sci. Conf. 2nd,* pp. 2683-2697.

Robinson N. (1966) *Solar Radiation.* Elsevier, New York. 347 pp.

Schramm L. S., McKay D. F., Zook H. A., and Robinson G. A. (1985) Analysis of micrometeorite material captured by the Solar-Max satellite. In *Proc. SMRM Degradation Study Workshop,* pp. 243-244. Goddard Space Flight Center, Greenbelt, MD.

Scott R. F. and Zuckerman K. A. (1971) Examination of returned Surveyor III surface samples. *Proc. Lunar Sci. Conf. 2nd,* pp. 2743-2751.

Silberberg R., Tsao C. H., Adams J. H. Jr., and Letaw J. R. (1985) Radiation transport of cosmic ray nuclei in lunar materials and radiation doses. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 663-669. Lunar and Planetary Institute, Houston.

SMRM (Solar Maximum Recovery Mission) (1985) *Proc. SMRM Degradation Study Workshop.* NASA Goddard Space Flight Center, Greenbelt, MD. 356 pp.

Taylor G. J. (1988) Geological considerations for lunar telescopes. In *Future Astronomical Observatories on the Moon* (J. O. Burns and W. W. Mendell, eds.), pp. 21-27. NASA CP-2489.

Watson J. K., Dickinson D. W., and Rokhlin S. I. (1988) Survey of Soviet space welding technology development. In *Engineering, Construction, and Operations in Space* (S. W. Johnson and J. P. Wetzel, eds.), pp. 820-831. American Society of Civil Engineers, New York.

Zook H. (1978) Dust, impact pits, and accreta on lunar rock 12054. *Proc. Lunar Planet. Sci. Conf. 9th,* pp. 2469-2484.

LUNAR BASE ACTIVITIES AND THE LUNAR ENVIRONMENT

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The Moon is an attractive site for astronomical observatories and other facilities because of the absence of a substantial lunar atmosphere and the stability of the lunar surface. The present lunar atmosphere is sufficiently transparent that there is no significant image distortion due to absorption or refraction. This thin atmosphere results from a combination of small sources and prompt losses. The major source that has been identified is the solar wind, whose total mass input into the lunar atmosphere is approximately 50 gm/sec. The major components of the solar wind are light elements (H and He) that promptly escape from the lunar surface by exospheric evaporation (Jeans' escape). The principal atmospheric loss mechanism for heavier gases is photoionization within a period of weeks to months, followed by immediate loss to the solar wind. Lunar base activities will modify the lunar atmosphere if gas is released at a larger rate than that now occurring naturally. Possible gas sources are rocket exhaust, processing of lunar materials, venting of pressurized volumes, and astronaut life support systems. For even modest lunar base activity, such sources will substantially exceed natural sources, although effects are expected to be localized and transient. The Apollo database serves as a useful reference for both measurements of the natural lunar environment and its modification by lunar base activities.

INTRODUCTION

The absence of a substantial lunar atmosphere makes the Moon an attractive site for astronomical observatories and for materials processing facilities that require a high vacuum. The present lunar atmosphere is sufficiently transparent that there is no significant image distortion due to absorption or refraction. Furthermore, the absence of a significant lunar ionosphere allows the use of the Moon for radio astronomy at frequencies much lower than those observed with terrestrial observatories. Other benefits to lunar base activities are the surface stability that facilitates the construction of large arrays, the lack of erosion or wind loading, and the availability of ample shielding from space radiation. These unique conditions at the lunar surface will enable the operation of science facilities not feasible elsewhere (*Mendell,* 1985).

To evaluate quantitatively the operational capabilities of lunar science facilities, it is necessary both to specify the natural lunar environment and to assess the alterations to it that might result from lunar base activities. Fortunately, many of the physical environmental conditions at the lunar surface were measured during the Apollo program. In particular, several sensors measured the neutral and ionized gases at the lunar surface. Although these measurements were limited both spatially and temporally, they have been used to identify the sources of the natural lunar atmosphere and ionosphere, as well as the loss processes that are effective in the lunar environment.

Lunar base activities can modify the lunar atmosphere by gas release at a larger rate than is now occurring naturally. Possible gas sources, such as rocket exhaust and astronaut life support systems, can be estimated from the experiences of the Apollo program. The Apollo database serves as a useful reference for both measurements of the natural lunar environment and its modifications by manned activities. Although no specific experiments were performed and measurements were not comprehensive, data do exist for several artificial releases such as the lunar module liftoff, S-IVB impacts, and cabin vents. During the Apollo missions rocket exhaust was a substantial, although transient, addition to the lunar atmosphere.

This paper reviews the characteristics of the natural lunar atmospheres and ionospheres, including known sources and loss processes. The gas sources expected to be associated with lunar base activities are identified, and the expected release rates are calculated. The Apollo data relevant to atmospheric alteration are briefly discussed. Finally, several recommendations are made for investigations that need to be made in support of a lunar base program.

CHARACTERISTICS OF THE LUNAR ATMOSPHERE

Prior to the Apollo program little was known about the lunar atmosphere except that it is very transparent to both light and radiowaves, implying that surface densities of both neutral and ionized gases are very small (*Johnson,* 1971).

Several kinds of atmospheric and ionospheric monitors were placed on the lunar surface as part of the Apollo Lunar Surface Experiments Package (ALSEP) during the Apollo program. The Cold Cathode Gauge Experiment (CCGE) and the Lunar Atmospheric Composition Experiment (LACE) measured the density and flux of neutral gases, and the Suprathermal Ion Detector Experiment (SIDE) measured the flux of atmospheric ions. Surface photoelectrons and the lunar plasma environment

were detected by the Charged Particle Lunar Environment Experiment (CPLEE) and the Solar Wind Spectrometer (SWS). The locations of these instruments are summarized in Table 1.

TABLE 1. ALSEP atmosphere and ionosphere observations.

Instrument	Missions	Phenomenon
CCGE	12, 14, 15	Total gas pressure
LACE	17	Nightside atmospheric composition
SIDE	12, 14, 15	Atmospheric ions
CPLEE	14	Surface photoelectrons
SWS	12, 15	Solar wind plasma

The CCGE was a total pressure monitor that provided the first estimates of the number density of neutral gases at the lunar surface (*Johnson et al.,* 1972). The observed nighttime concentration was about $2 \times 10^5\,cm^{-3}$. The daytime value was $10^7\,cm^{-3}$, but was thought to be an upper limit because the measurements were presumed to be affected by local contamination.

The most sensitive instrument for detection of lunar atmospheric gases was the LACE, which was deployed only on the last Apollo mission. This neutral gas mass spectrometer was able to detect atmospheric gases with excellent resolution and high sensitivity (*Hoffman et al.,* 1973). During the nine months of operation, useful data for most gases were generally obtained only during the lunar night. During the entire night, measurements were made of Ar, Ne, and He. Gases that condense on the lunar darkside, such as CH_4, NH_3, and CO_2, were measured only for a limited time just prior to sunrise (*Hoffman and Hodges,* 1975). Their daytime abundance (see Fig. 1) can be calculated from the solar wind source intensity and the modeling of atmospheric dynamics (*Hodges,* 1976).

The final instrument providing atmospheric information was the SIDE. Each SIDE contained two ion detectors that monitored ions originating from the lunar atmosphere and from gases released during Apollo missions, as well as ions from the Earth's magnetosphere and the solar wind (*Freeman et al.,* 1971, 1973; *Lindemann et al.,* 1973). An important feature of the SIDE network was its ability to monitor the local atmosphere density by a variety of methods (*Freeman and Benson,* 1977). Also, since the detectors relied on an exterior electric field to accelerate the ions, they were less sensitive to contamination due to internal instrument degassing. The SIDE measurements on the dayside were consistent with the gas concentrations measured by the CCGE and inferred from the LACE data, assuming that molecular species are present on the dayside in addition to the inert gases (*Vondrak et al.,* 1974).

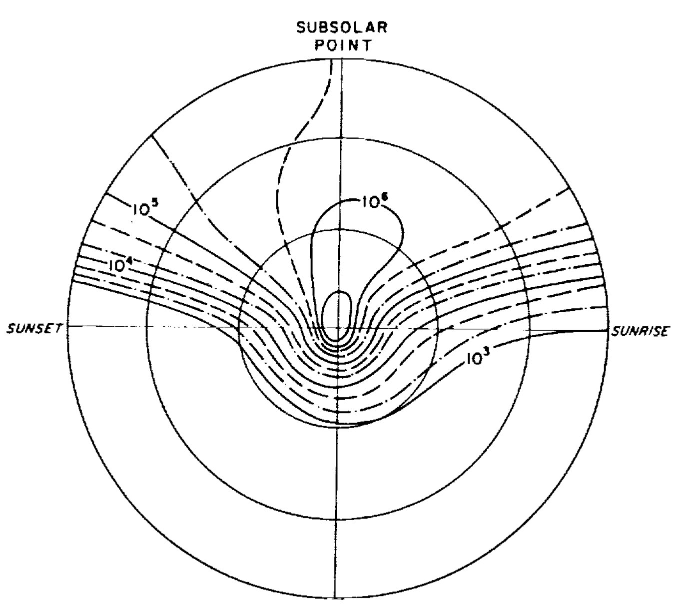


Fig. 1. Stereographic projection of the distribution of carbon dioxide (CO_2) on the northern hemisphere of the Moon (*Hodges,* 1976).

Sodium and K have recently been detected in the lunar atmosphere by ground-level spectroscopic observations (*Potter and Morgan,* 1988). The surface densities of these elements are quite small (less than $10^2\,cm^{-3}$), and are thought to originate from the vaporization of surface minerals by meteoric impact or solar wind sputtering.

The concentrations of the lunar atmospheric species are indicated in Table 2. The predominant dayside species is CO_2, and Ne is the predominant nightside gas. The total atmospheric mass globally integrated for each species is shown in Table 3. The quantities of gases temporarily adsorbed on the lunar surface exceed these amounts by factors of about 1000. The quantities shown in Tables 2 and 3 are based on measurements and inferences from the Apollo and other data. However, it should be kept in mind that the limited observations result in significant uncertainties, especially for the molecular gases. In particular, the quantities shown for each of the carbon compounds are overestimates because the calculations by *Hodges* (1976) assumed that all the solar wind carbon was the source of each particular compound.

The quantities of gases in the lunar atmosphere are quite small. For comparison, the terrestrial atmosphere concentrations are approximately $10^{19}\,cm^{-3}$ at the Earth's surface and are as little as 10^5 to $10^7\,cm^{-3}$ at altitudes between 500 km and 1500 km.

The thin lunar atmosphere results from a balance between small sources and rapid losses. The primary source of the lunar atmosphere is the solar wind. Because the Moon lacks any

TABLE 2. Lunar atmosphere concentrations.

Species	Daytime (cm^{-3})	Night (cm^{-3})
H_2	4×10^3	1.2×10^4
^4He	2×10^3	4×10^4
^{20}Ne	4×10^3	1×10^5
^{36}Ar	1×10^2	3×10^3
^{40}Ar	1.6×10^3	4×10^4
CO	3×10^5	$<10^3$
CO_2	6×10^5	$<10^3$
CH_4	7×10^4	$<10^3$
Na	6.7×10^1	
K	1.5×10^1	

TABLE 3. Total atmospheric mass.

Species	Mass (kg)
CO_2	5200
CO	2000
CH_4	640
Ne, Ar	<500

substantial intrinsic magnetic field or thick atmosphere, the solar wind flows directly onto the lunar surface. The flux and composition of the solar wind have been measured by spacecraft instruments (e.g., *Bochsler,* 1989; *Bochsler et al.,* 1986). The typical total flux of solar wind species on the sunward hemisphere of the Moon is listed in Table 4. The solar wind ions impact the lunar surface, are thermalized and perhaps chemically altered there, and are then released as neutral atoms or molecules that travel along collision-free ballistic trajectories between impacts with the surface. Although the total solar wind mass input to the lunar surface is about 50 gm/sec, the major components are the light elements H and He. The thermalized H and He have velocities greater than the lunar escape velocity and are rapidly lost from the lunar environment. The total mass input by the heavier elements in the solar wind that reside longer in the lunar atmosphere is less than 1 gm/sec.

Other possible sources of the lunar atmosphere are material released or weathered from the lunar surface, or gases released from the lunar interior. As discussed above, surface weathering results in small quantities of Na and K. The only endogenous gases that have been conclusively identified are Ar and Rn, decay products of radioactive materials in the lunar interior. About 10% of the He in the lunar atmosphere is thought to be of radiogenic origin (*Hodges,* 1977). The most important is ^{40}Ar, originating from K. The total source rate of ^{40}Ar is about 0.06 gm/sec, and is about an order of magnitude larger than the solar wind input of ^{36}Ar (*Hodges and Hoffman,* 1974). Although larger sources due to the localized release of gases have been suggested in association with lunar transient phenomena (*Middlehurst,* 1967), none has been conclusively identified or measured. The LACE found evidence for a time variation in the abundance of ^{40}Ar, possibly related to lunar teleseismic events (*Hodges and Hoffman,* 1974), but found no unambiguous transient increase in the abundance of other gases. An extensive analysis of SIDE data likewise failed to uncover any transient increases in the lunar atmosphere (*Freeman and Benson,* 1977), and these data were used to set upper limits to transient outgassing from specific sites on the lunar surface (*Vondrak,* 1977).

The neutral atmosphere of the Moon is lost into space by two mechanisms: loss to the solar wind and thermal escape. Mass is lost to the solar wind because atmospheric ions are driven into space by the interplanetary electric field that results from the flow of the interplanetary magnetic field past the Moon (*Manka and Michel,* 1971). Such ion pickup has been observed at Halley's Comet and in artificial chemical releases in space. The time constant for loss by this process is determined by the photoionization lifetime (typically 10^6 to 10^7 sec). Thermal escape results from the fact that the gases in the lunar atmosphere travel along collisionless ballistic trajectories in the lunar gravitational field until they impact the surface or escape into space. Gases that do impact the surface are briefly adsorbed by the surface material (*Cadenhead et al.,* 1972). The gases come into thermal equilibrium with the lunar surface before desorbing. When they are emitted from the surface, a fraction has sufficient thermal energy to escape from the lunar gravity field. This loss of the faster-moving exospheric gases is referred to as evaporative loss or Jeans' escape (*Jeans,* 1923). The atmospheric loss rate by this process is dependent upon the mass and temperature of the gases. For the two lightest atmospheric species, H and He, the thermal escape time constant is several thousand seconds. For all other atmospheric gases the thermal escape time constant is very long (e.g., 190 years for Ne and 10^{10} years for Ar), much longer than the time constant for solar-wind-induced mass loss (*Johnson,* 1971).

TABLE 4. Solar wind flux to lunar surface.

Element	Flux (gm/sec)
Hydrogen (^{1}H)	40
Helium (^{4}He)	8
Oxygen (^{16}O)	0.2
Carbon (^{12}C)	0.1
Neon (^{20}Ne)	0.07
Nitrogen (^{14}N)	0.03
Silicon (^{28}Si)	0.05
Argon (^{36}Ar)	0.004

THE LUNAR IONOSPHERE AND PLASMA ENVIRONMENT

Extensive investigations of the lunar plasma environment made during the Apollo program have shown that, like the Earth, the Moon is surrounded by a region of ionized plasma. However, unlike the terrestrial ionosphere, the lunar ionosphere is directly coupled to the solar wind by the interplanetary electric field. As a result, the lunar atmospheric ion fluxes are both directional and variable and the ions have a nonthermal energy distribution. Collisions within the ionosphere are unimportant, and interactions with the lunar surface control the ionospheric chemistry. In addition to this ionosphere of lunar origin, the region surrounding the Moon is permeated by the interplanetary and magnetospheric plasma.

The lowest part of the lunar ionosphere consists of a thin, dense sheath surrounding the Moon as a result of the electric potential of the lunar surface. Measurements by the SIDE (*Fenner et al.,* 1973) and CPLEE (*Reasoner and Burk,* 1972) have demonstrated that the surface potential is positive on the dayside, at a value of about 30 V with respect to the solar wind. There the sheath consists of a photoelectron layer with an electron density at the surface of about 10^4/cm^3 and an altitude extent of several hundred meters (*Reasoner and Burke,* 1972). Near the sunrise and sunset terminators and on the nightside the lunar surface potential becomes negative (*Manka and Michel,* 1971).

The most extensive part of the lunar ionosphere consists of ions produced from the lunar atmosphere. The neutral atmospheric gases are ionized by the solar ultraviolet photons and by charge transfer with the solar wind in a time of typically 10^6 to 10^7 sec (*Siscoe and Mukherjee,* 1972). These ions are then accelerated by the interplanetary electric field (typically 1-3 mV/m) and are driven either into space or into the lunar surface. As shown in Fig. 2, the initial motion of an ion is in the direction of the solar wind electric field. Ions formed on the dayside in the lower hemisphere (with respect to the electric field) are generally driven into the lunar surface. Ions formed in the upper hemisphere escape into the solar wind because the magnetic force curves the ions in the direction of the solar wind flow with a resulting cycloidal orbit that carries them away from the Moon. Thus, on the lunar dayside the interplanetary electric field drives approximately 50% of the atmospheric ions into space (*Manka and Michel,* 1971; *Vondrak et al.,* 1974).

The flux and energy distribution of the atmospheric ions can be easily calculated for the region where ions are driven directly into the lunar surface by the interplanetary electric field. As shown in Fig. 3, we assume that the neutral atmosphere is distributed

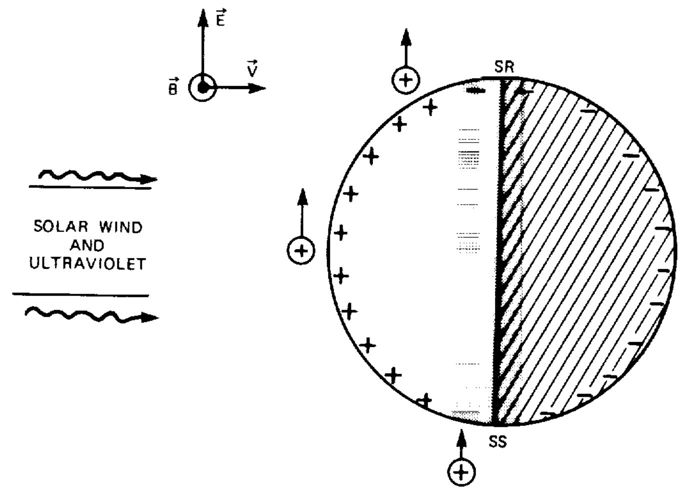
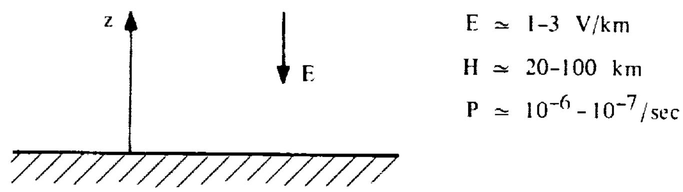


Fig. 2. Schematic representation of the sources of the electric fields near the northern hemisphere of the Moon. The interplanetary electric field is due to the motion of the solar wind magnetic field, and the lunar surface electric field arises from charges on the lunar surface. The shaded area near the terminator is where the potential is negative and the lunar atmosphere is exposed to ionizing solar radiation (*Vondrak et al.,* 1974).



PRODUCTION RATE = $P n(z)$

ENERGY DISTRIBUTION: $F(\mathcal{E}) = \frac{P n_0}{qE} e^{-z/H} e^{-\mathcal{E}/qEH}$

INTEGRAL FLUX = $P n_0 H e^{-z/H}$

DENSITY: $\rho = P n_0 \sqrt{\frac{\pi}{2}\frac{mH}{qE}}\; e^{-z/H}$

Fig. 3. Flux and density of lunar atmospheric ions at the lunar surface when the interplanetary electric field is pointed downward.

exponentially in altitude with a scale height H. The scale height H for a gas of mean molecular mass m is equal to kT/mg, where T is the temperature, k is Boltzmann's constant, and g is the surface gravitational acceleration. This distribution applies to both collisional atmospheres and thin exospheres (*Johnson,* 1971). If the ions are accelerated by a uniform interplanetary field of magnitude E, then they will have an altitude distribution that is also exponential according to the neutral gas scale height H (*Benson et al.,* 1975). The ions will have a residence time of only a few seconds before impacting the surface. The resulting energy distribution is strongly nonthermal and the differential energy flux for an ion of charge q is an exponential distribution with a folding energy equal to qEH (typically 20-300 eV). The SIDE measurements of the differential energy distribution of the lunar atmospheric ions are consistent with such an exponential altitude distribution. The production rate of these atmospheric ions is proportional to the product of neutral number density and ionization rate. Consequently, the density and flux of the atmospheric ions vary, being directly responsive to changes in the solar wind flux and transient outgassing from the lunar surface. The altitude variation of the density of lunar atmospheric ions is shown in Fig. 4, where we have assumed E = 1 mV/M and H = 50 km. The subsolar photoelectron layer is shown for comparison (*Reasoner and Burke,* 1972). Because of the electric potential of the lunar surface, there is a lack of charge neutrality in the plasma near the lunar surface.

In addition to the ionospheric plasma of lunar origin (the photoelectron layer and lunar atmospheric ions), the Moon is exposed to plasma of extralunar origin (the solar wind, mangetospheric plasma, and cosmic rays). Because the Moon lacks shielding by either a strong magnetic field or dense atmosphere, these extralunar plasmas penetrate directly to the surface where they have been regularly observed by the SIDE, CPLEE, and SWS. As shown in Fig. 5, during each orbit the Moon spends about 20 days in the solar wind, 4 days in the magnetosheath (thermalized solar wind plasma), and 4 days within the magnetospheric plasma sheet.

ALTERATION OF THE LUNAR ATMOSPHERE BY LUNAR BASE ACTIVITIES

Because the lunar atmosphere and ionosphere is so tenuous, it is susceptible to alteration by human activity at the lunar surface. Lunar base activities can modify the lunar atmosphere by gas release at a larger rate than is now occurring naturally. Possible gas sources are rocket exhaust, gas release from processing lunar materials, venting of pressurized volumes, and astronaut life support systems. These rates can be estimated from the experience of the Apollo program (see Table 5).

For Apollo, rocket exhaust was a substantial, although transient addition to the lunar atmosphere (see Fig. 6). Each Apollo mission deposited nearly 10^4 kg of rocket exhaust into the lunar

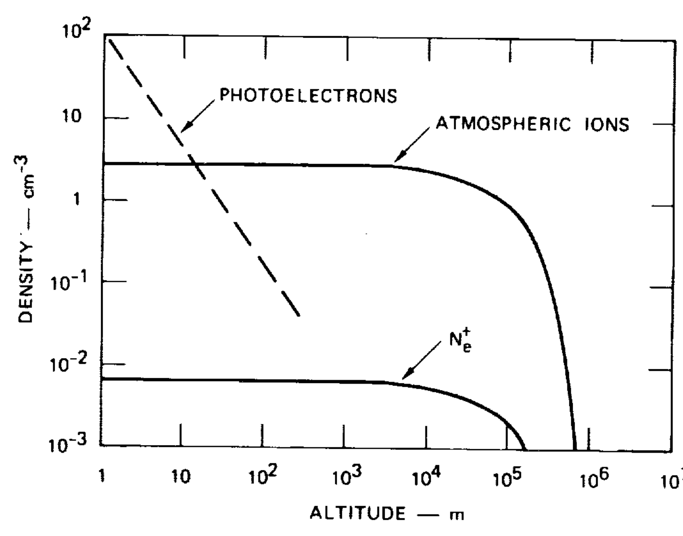


Fig. 4. Altitude variations of photoelectrons and lunar atmospheric ions.

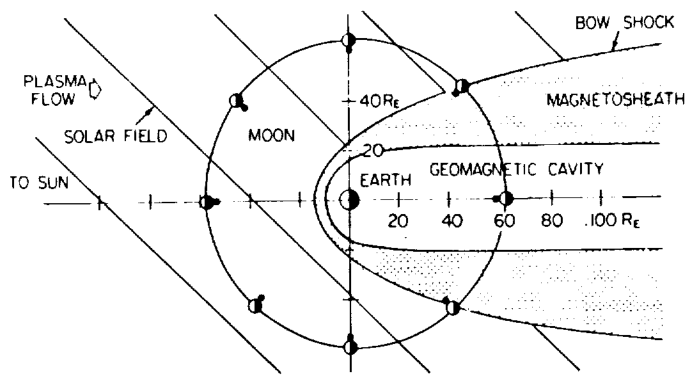


Fig. 5. Schematic diagram showing the three plasma regions traversed by the Moon. The plane of the figure is essentially the ecliptic plane.

TABLE 5. Artificial sources of atmospheric gases.

Source	Quantity
Lunar module landing/ascent	3000 kg
Apollo CSM return	5000 kg
Lunar module cabin venting	3 kg
Airlock venting (1000 ft^3)	27 kg
Astronaut suit	0.3 gm/sec (H_2O)
Structural leakage	0.1 mg/m^2-sec
Trapped gases in soil	0.1 g/kg

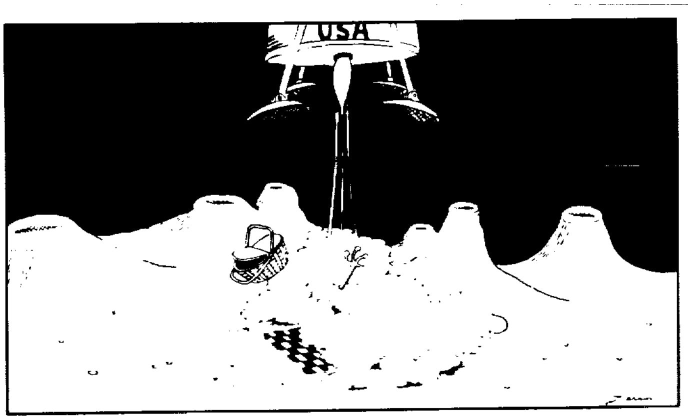


Fig. 6. Conceptual representation of the effect of Apollo lunar module exhaust gases on the lunar environment. The FAR SIDE cartoon by Gary Larson is reprinted by permission of Chronicle Features, San Francisco, California.

environment. However, since Apollo missions occurred infrequently, no long-lived increase in the lunar atmosphere was produced. A permanent lunar base might result in gas release at a rate equivalent to 10^{-2} kg-sec^{-1} per person, assuming supply traffic equal to one Apollo mission month^{-1} per person. Therefore, even small lunar basees would add gases to the lunar environment at a greater rate than natural sources. Another potential source of gas contamination is mining of the lunar surface. Examination of the lunar samples indicates that about 10^{-4} of the mass of the lunar soil consists of trapped gases. Perhaps 10% of these gases will be released during upheaval and heating of the soil in normal mining operations. Construction of large structures in space has been proposed by *O'Neill* (1974) that would require the removal of about 10^9 kg of soil from the lunar surface. Mining this amount would yield only about 10^4 kg of gases, so this does not appear to be a significant source of atmospheric contamination. However, vigorous lunar base activity for mining of rocket propellant or He^3 could result in more substantial release rates.

Gas leakage rates from pressurized structures are difficult to estimate. The Gemini and Apollo space capsules had passive leakage rates of 10 and 30 mg/sec, respectively. These leakage rates were due to continuous diffusion through the cabin walls and outgassing through small leaks, and do not include gas release from thruster firings and cabin ventings. These and Skylab leakage measurements, as well as space station design specifications, indicate a minimum passive leakage rate of about 0.2 mg/m^2-sec for current-technology spacecraft. Improved sealing technology and system design during the next 30 years are expected to reduce the rate by not more than a factor of 10 (*Nishioka et al.,* 1973). To exceed the natural source rate of 10 gm/sec, the surface area of the lunar base pressurized areas would have to be more than 50,000 m^2. The use of partially pressurized airlocks would possibly be a more significant source of gas venting than leakage from the sealed structures.

The consequences of lunar atmospheric modification are a localized increase in atmosphere absorption and light scattering. Because of the reduced gravity on the Moon, the effects of even localized releases spread out to distances of hundreds of kilometers. Dispersal beyond that distance depends on the degree of gas absorption by the lunar regolith. The manner in which neutral gases are transported across the lunar surface has been analyzed by several methods (*Vogel,* 1966; *Milford and Pomilla,* 1967; *Hodges,* 1972; *Hodges et al.,* 1972; *Hall,* 1973; *Burns et al.,* 1991). All these transport calculations assume that no collective plasma effects are important and that the gases move on individual trajectories because the mean free path is greater than the hop distance. The dispersal rate and spatial distribution depend on the source intensity and duration, and the surface retention characteristics. The analyses also indicate that the propagation of gas clouds can be separated into two regions: the direct flux near the source, which describes the initial gas flow, and the diffusive flux whereby gases propagate to greater distances. The direct flux will extend laterally to about two scale heights, typically 100 km, which is the limit of significant gas transport if all gases stick to the lunar surface. If the gases are only adsorbed and subsequently reemitted, they will gradually spread to much larger distances. Because the gases spread rapidly over large areas, gas releases associated with moderate injection rates (approximately 1 kg/sec) are not expected to be a detriment to astronomical observations and high-vacuum materials processing (*Burns et al.,* 1991).

LONG-LIVED ALTERATION OF THE LUNAR ATMOSPHERE

Lunar base activities described in the preceding section are not significant because the solar wind is able to rapidly remove gases added to the thin lunar atmosphere, with a photoionization time constant of weeks to months. However, if the lunar atmosphere were to become more dense, loss to the solar wind would be ineffective and thermal escape from the exosphere would become

the dominant loss mechanism. The lunar atmosphere would then be long-lived since thermal escape lifetimes are at least several hundred years for gases heavier than He (*Johnson,* 1971).

The solar wind loss mechanism is the dominant process only as long as the solar wind has direct access to the majority of the atmosphere. As the atmosphere becomes more dense, newly formed ions of atmospheric origin load down the solar wind and cause it to be diverted around the Moon (see Fig. 7). The effectiveness of the lunar atmosphere loss mechanisms for varying atmospheric total density has been evaluated quantitatively by *Vondrak* (1974, 1976) for an oxygen atmosphere. The exospheric loss rate to the solar wind was calculated by assuming that the total ionization rate was 5×10^{-6} ions per atom sec^{-1} and that half of the exospheric mass was on the dayside. The limit to mass lost to the solar wind for a thick atmosphere was taken as equal to the solar wind mass flux through the lunar cross section ($\sim$50 g-sec^{-1}), since critical mass loading of the solar wind occurs if ions are added at a rate comparable to the solar wind flow. Venus and Mars each lose about 10 g-sec^{-1} to the solar wind (*Michel,* 1971; *Cloutier et al.,* 1974; *Breus et al.,* 1989), which is 1% (Venus) and 20% (Mars) of the mass flux of the solar wind through their cross-sectional areas. The thermal escape rate for a thin exosphere in contact with the surface was calcualted with

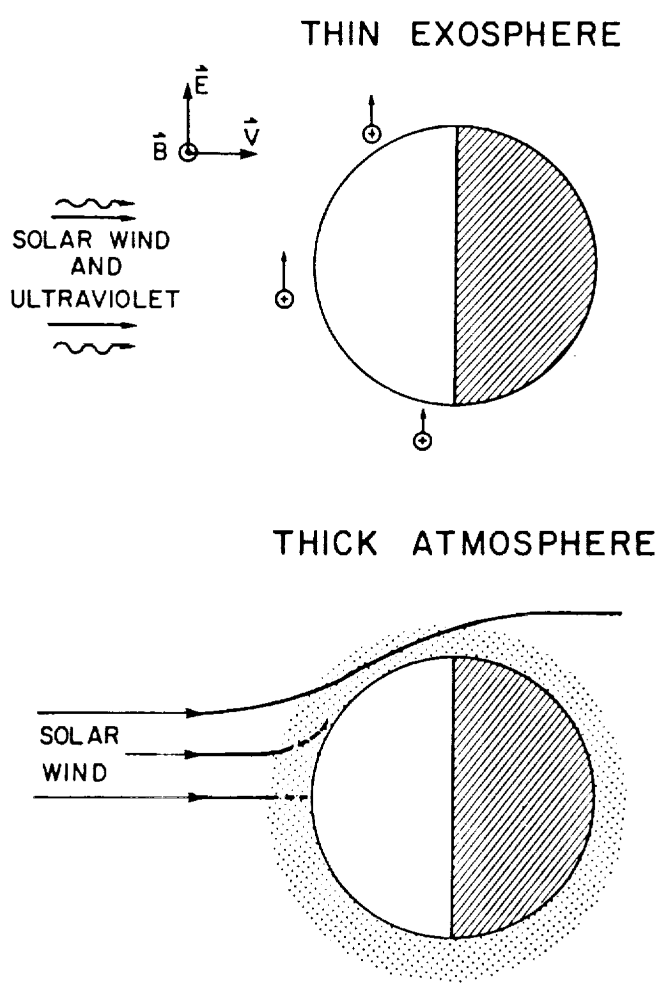


Fig. 7 Interactions between the solar wind and a thin atmosphere with ion pickup, and a thick atmosphere that diverts the solar wind flow.

300 K as the weighted average of the lunar surface temperature. Absorption of the solar wind and ultraviolet by a thick atmosphere results in exospheric heating and more effective thermal evaporation.

As the atmospheric density is increased, the exospheric base rises above the surface and the mass lost to thermal evaporation is no longer a constant fraction of the entire atmospheric mass. The mass loss rate is then a fraction of only the exospheric mass because the atmosphere below the exosphere does not participate in the thermal escape. The transition to a long-lived atmosphere in which thermal escape is the primary loss mechanism requires a total mass of 10^8 kg, and requires a source rate estimated to be about 60 kg-sec^{-1}. However, the surface density at this transition would still be low compared to densities in the terrestrial atmosphere. For atmospheres in excess of 10^8 kg, the atmosphere will grow if the source rate exceeds about 30 kg-sec^{-1}. For thicker atmospheres the mass loss rate depends on the escape velocity at the base of the exosphere. Therefore, the loss rate increases slowly because the exosphere rises to higher altitudes as a result of the increase in total mass.

The growth of the artificial atmosphere is ultimately limited by the large escape rate that occurs when the exosphere rises to an altitude where the thermal velocity is comparable to the escape velocity. In this case the exosphere is in a state of rapid escape, referred to as blowoff, in which the loss of gas ceases to be evaporative and becomes a hydrodynamic outflow (*Fahr and Shizgal,* 1983). The altitude at which this occurs depends on the exospheric composition and temperature. For an exosphere of atomic oxygen, the thermal and escape velocities are equal at an altitude of 13.6 lunar radii for an exospheric temperature of 400 K and 6.8 lunar radii for a temperature of 800 K. The surface densities of such extended atmospheres would greatly exceed that of the Earth. If the artificial atmosphere consisted of an atmosphere of molecular gases and a surface density equal to the Earth's atmosphere, the base of the exosphere would be at an altitude of about 3 lunar radii. In this case, the total escape rate of the exosphere (assumed to be atomic oxygen at a temperature of 700 K) would be about 60 kg/sec.

An obvious speculation is the feasibility of creating an artificial lunar atmosphere that would be "breathable" or as dense as the terrestrial surface atmosphere. Such a lunar atmosphere would have a total mass of about 2×10^{18} kg. Obtaining this much oxygen by vaporization of lunar soil requires an amount of energy equal to 5×10^{25} J, or 2×10^9 GW-yr. If nuclear devices were used, the equivalent of 2×10^{11} kt of TNT would be needed since *Ehricke* (1974) estimates that a 1-kt device will yield 10^7 kg of oxygen. Because this is approximately 10^4 times larger than the total U.S. stockpile of nuclear weapons, it seems impractical that such an amount of gas could be generated by current technology. There are no known natural gas reservoirs on the Moon, although permanently shaded regions near the lunar poles might contain as much as 10^{14} kg of condensed water (*Arnold,* 1979). Therefore, it would be necessary to import gases. For example, a cometary nucleus of radius 80 km contains 2×10^{18} kg of oxygen and could be used, in principle, to supply the man-made atmosphere.

Although the artificial generation of a lunar atmosphere can be considered as another potential method for modification of planetary environments, the desirability of intentionally increasing the density of the lunar atmosphere is highly questionable because the primary application of a lunar laboratory involves utilization of the present lunar "vacuum."

APOLLO MEASUREMENTS OF ARTIFICIAL GAS RELEASES

The Apollo database serves as a useful reference for both measurements of the natural lunar environment and its modifications by manned activities. Although no specific experiments were performed and measurements were not comprehensive, data do exist for several artificial releases such as lunar module liftoff, S-IVB impacts, and cabin vents. These data are consistent with the theoretical expectations for transient release dynamics, although some puzzles have been identified. Unfortunately, no measurements of Apollo activities were made by LACE because it was deployed on the last mission.

The lifetime of gases in the lunar atmosphere has been measured by observations of exhaust gases released into the lunar atmosphere by the Apollo vehicles. Figure 8 shows an exhaust spectrum detected by the SIDE mass analyzer soon after an Apollo landing (*Freeman et al.,* 1973). The mass distribution agrees well with laboratory measurements of the lunar module fuel combustion products. Also shown in Fig. 8 are observations of the exhaust density that were made approximately two months later. It is seen that the gas density decreases with approximately a one-month exponential decay time, consistent with the photoionization lifetime.

The Apollo 15 SIDE measured the differential fluxes of magnetosheath ions during the ascent of the Apollo 15 lunar module (*Hills et al.,* 1972). At LM ascent, a strong decrease occurred in the magnetosheath ion fluxes being detected at the time. This decrease lasted approximately eight minutes. It could be attributed to energy loss in the relatively dense exhaust gas, to losses by charge exchange, or to temporary deviations of the magnetosheath ion-flow direction caused by the exhaust gas. Full recovery of the fluxes to preascent intensities had a puzzling time-dependent variation, and variations were also seen on the ALSEP magnetometer. These could have been signatures of collective plasma phenomena or may have been results of complicated magnetosheath flow.

On April 15, 1970 the Apollo 13 S-IVB stage impacted the nighttime lunar surface approximately 140 km west of the Apollo 12 ALSEP site and 410 km west of the dawn terminator (see Fig. 9). Beginning 20 sec after impact, the SIDE and the SWS observed a large flux of positive ions (maximum flux $\sim 3 \times 10^8$ ions/cm^{2}sec-sr) and electrons (*Lindeman et al.,* 1974). Two separate streams of ions were observed: a horizontal flux that appeared to be deflected solar wind ions and a smaller vertical flux of predominantly heavy ions (>10 amu), which were probably material vaporized from the S-IVB stage. An examination of the data shows that collisions between neutral molecules and hot electrons (50 eV) were probably an important ionization mechanism in the impact-produced neutral gas cloud. These electrons, which were detected by the SWS, are thought to have been energized in a shock front or some form of intense interaction region between the cloud and the solar wind. Thus, strong ionization and acceleration are seen under conditions approaching a collionless state. This observation indicates that plasma interactions need to be taken into account when evaluating the effects of large gas releases.

CONCLUSIONS

Although the basic features of the lunar environment were surveyed during the Apollo program, our present understanding

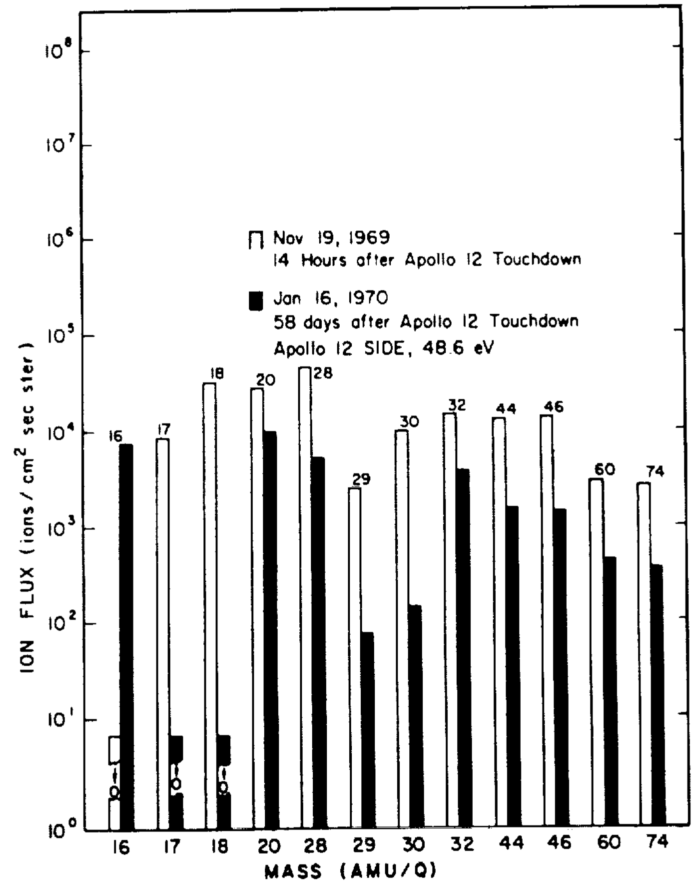


Fig. 8. Observed decrease in exhaust gas fluxes over a 58-day period (*Freeman et al.,* 1973).

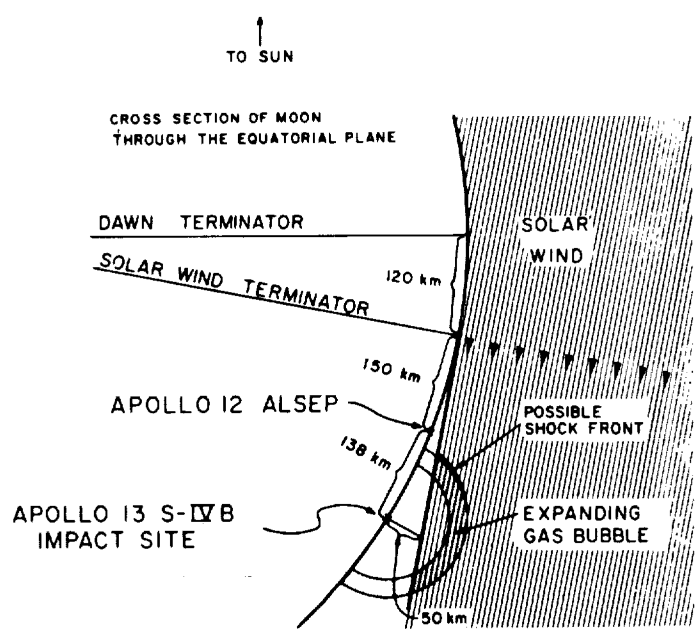


Fig. 9. An equatorial cross-sectional view of the Moon showing the location of the Apollo 13 S-IVB impact with respect to the Apollo 12 ALSEP site and the dawn and solar wind terminators (*Lindeman et al.,* 1974).

is somewhat fragmented and incomplete. The major uncertainties concern the composition and concentration of the dayside lunar atmosphere, the possible presence of transient lunar degassing, and the dynamics of the lunar ionosphere.

Based on our current understanding of the lunar atmosphere processes, small lunar bases are not expected to produce any significant long-lived alteration of the lunar atmosphere. Although the artificial gas release rate can easily exceed the natural rate, the principal effects should be both localized and transient.

Because the lunar atmosphere and ionosphere are susceptible to alteration by human activities, it is important that these be extensively surveyed prior to any lunar surface return program. In particular, measurements need to be made of the lunar atmosphere on the dayside. Also, better measurements need to be made of lunar atmospheric ions. The SIDE ion mass spectrometer had a narrow, fixed field of view and low sensitivity. Modern ion sensors can measure more completely the full three-dimensional plasma distribution with much higher sensitivity.

For astronomical applications, an important early measurement is the optical quality ("seeing conditions") at the lunar surface. Although the atmosphere is expected to be nearly transparent, there have been suggestions that dust may be present, due to measurements made by Surveyor 7, the Apollo crew, and the Lunokhod photometer (*Rozenberg*, 1970; *McCoy and Criswell*, 1974). Such measurements of atmospheric quality need to be continued after a lunar base is established, in order to monitor any alteration of the atmosphere.

Finally, the response of the lunar atmosphere to gas releases needs to be determined. Controlled releases of gases should be made, perhaps of tracer gases not naturally present in the lunar environment. Monitoring of their transport to other locations will validate our understanding of lunar environmental dynamics. Such experiments may also identify the location of any endogenous sources of volatiles. The discovery of such volatiles would greatly facilitate the establishment of a lunar base.

REFERENCES

Arnold J. (1979) Ice in the lunar polar regions. *J. Geophys. Res., 84,* 5659.

Benson J., Freeman J., and Hills H. (1975) The lunar terminator ionosphere. *Proc. Lunar Sci. Conf. 6th,* pp. 3013-3021.

Bochsler P. (1989) Velocity and abundance of silicon ions in the solar-wind. *J. Geophys. Res., 94,* 2365-2373.

Bochsler P., Geiss J., and Kunz S. (1986) Abundances of carbon, oxygen, and neon in the solar wind during the period from August 1978 to June 1982. *Sol. Phys., 103,* 177-201.

Breus T., Bauer S., Krymskii A., and Mitnitskii V. (1989) Mass loading in the solar-wind interaction with Venus and Mars. *J. Geophys. Res., 94,* 2375-2382.

Burns J. O., Fernini I., Sulkanen N., Duric N., Taylor J., and Johnson S. (1992) Artificially generated atmosphere near a lunar base. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Cadenhead D. A., Wagner N. J., Jones B. R., and Stetter J. R. (1972) Some surface characteristics and gas interactions of Apollo 14 fines and rock fragments. *Proc. Lunar Sci. Conf. 3rd,* pp. 2243-2257.

Cloutier P., Daniell R., and Butler D. (1974) Atmospheric ionwakes of Venus and Mars in the solar wind. *Planet. Space Sci., 22,* 967-990.

Ehricke K. A. (1974) Lunar industries and their value for the human environment on earth. *Acta Astronaut., 1,* 585-622.

Fahr H. and Shizgal B. (1983) Modern exospheric theories and their observational relevance. *Rev. Geophys. Space Phys., 21,* 75-124.

Fenner M. A., Freeman J. W., and Hills H. K. (1973) The electric potential of the lunar surface. *Proc. Lunar Sci. Conf. 4th,* pp. 2877-2888.

Freeman J. W. and Benson J. L. (1977) A search for gaseous emissions from the moon. *Phys. Earth Planet. Inter., 14,* 276-281.

Freeman J. W., Fenner M. A., Hills H. K., Lindeman R. A., Medrano R., and Meister J. (1971) Suprathermal ions near the moon. *Icarus, 16,* 328.

Freeman J. W., Hills H. K., Lindeman R. A., and Vondrak R. R. (1973) Observations of water vapor ions at the lunar surface. *Moon, 8,* 115-128.

Hall F. G. (1973) Role of pressure transients in the detection and identification of lunar surface gas sources. *J. Geophys. Res., 78,* 2111-2132.

Hills K., Meister J., Vondrak R., and Freeman J. (1972) Suprathermal ion detector experiment. In *Apollo 15 Preliminary Science Report,* pp. 12-1 to 12-10. NASA SP-289.

Hodges R. R. (1972) Response of lunar atmosphere to volcanic gas releases. *Planet. Space Sci., 20,* 1849-1864.

Hodges R. R. (1975) Formation of the lunar atmosphere. *Moon, 14,* 139-157.

Hodges R. R. (1976) The escape of solar-wind carbon from the moon. *Proc. Lunar Sci. Conf. 7th,* pp. 493-500.

Hodges R. R. (1977) Release of radiogenic gases from the moon. *Phys. Earth Planet. Inter., 14,* 282.

Hodges R. R. and Hoffman J. H. (1974) Implications of atmospheric ^{40}Ar escape on the interior structure of the moon. *Proc. Lunar Sci. Conf. 6th,* pp. 3039-3047.

Hodges R. R., Hoffman J., Yeh T., and Chang G. K. (1972) Orbital search for lunar vulcanism. *J. Geophys. Res., 77,* 4079-4085.

Hoffman J. H. and Hodges R. R. (1975) Molecular gas species in the lunar atmosphere. *Moon, 14,* 159-168.

Hoffman J. H., Hodges R. R., Johnson F. S., and Evans D. E. (1973) Lunar atmospheric composition results from Apollo 17. *Proc. Lunar Sci. Conf. 4th,* pp. 2865-2875.

Jeans J. (1923) *The Dynamical Theory of Gases.* Cambridge Univ., New York.

Johnson F. (1971) Lunar atmosphere. *Rev. Geophys. Space Phys., 9,* 813.

Johnson F. S., Carroll J. M., and Evans D. E. (1972) Lunar atmosphere measurements. *Proc. Lunar Sci. Conf. 3rd,* pp. 2231-2242.

Lindeman R. A., Freeman J. W., and Vondrak R. R. (1973) Ions from the lunar atmosphere. *Proc. Lunar Sci. Conf. 4th,* pp. 2889-2896.

Lindeman R. A., Vondrak R. R., Freeman J. W., and Snyder C. W. (1974) The interaction between an impact-produced neutral gas cloud and the solar wind at the lunar surface. *J. Geophys. Res., 79,* 2287.

Manka R. H. and Michel F. C. (1971) Lunar atmosphere as a source of lunar surface elements. *Proc. Lunar Sci. Conf. 2nd,* pp. 1717-1728.

McCoy J. and Criswell D. (1974) Evidence for a high altitude distribution of lunar dust. *Proc. Lunar Sci. Conf. 5th,* pp. 2991-3005.

Mendell W. W., ed. (1985) *Lunar Bases and Space Activities of the 21st Century.* Lunar and Planetary Institute, Houston. 865 pp.

Michel F. (1971) Solar-wind-induced mass loss from magnetic field free planets. *Planet. Space Sci., 19,* 1580-1582.

Middlehurst B. M. (1967) An analysis of lunar events. *Rev. Geophys. Space Phys., 5,* 173-179.

Milford S. N. and Pomilla F. R. (1967) A diffusion model for the propagation of gases in the lunar atmosphere. *J. Geophys. Res., 72,* 4533-4545.

Nishioka K., Arno R., Alexander A., and Slye R. (1973) *Feasibility of Mining Lunar Resources for Earth Use: Circa 2000 A.D.* NASA TMX-62267.

O'Neill G. K. (1974) The colonization of space. *Phys. Today, 27,* 32-38.

Potter A. E. and Morgan T. H. (1989) Discovery of sodium and potassium vapor in the atmosphere of the moon. *Science, 241,* 675-680.

Reasoner D. L. and Burke W. J. (1972) Direct observation of the lunar photoelectron layer. *Proc. Lunar Sci. Conf. 3rd,* pp. 2639-2654.

Rozenberg G. (1970) The atmospheric pressure on the moon according to the Surveyor 7 twilight photography. *Sov. Astron., 14,* 361-363.

Siscoe G. and Mulcherjee M. (1972) Upper limits on the lunar atmosphere determined from solar-wind measurements. *J. Geophys. Res., 77,* 6042-6051.

Vogel U. (1966) Molecular fluxes in the lunar atmosphere. *Planet. Space Sci., 14,* 1233-1252.

Vondrak R. R. (1974) Creation of an artifical lunar atmosphere. *Nature, 248,* 657.

Vondrak R. R. (1976) Creation of an artificial atmosphere on the moon. *Adv. Eng. Sci., 3,* 1215.

Vondrak R. R. (1977) Upper limit to gas emission from lunar transient phenomena sites. *Phys. Earth Planet. Inter., 14,* 293.

Vondrak R. R., Freeman J. W., and Lindeman R. A. (1974) Measurement of lunar atmospheric loss rate. *Proc. Lunar Sci. Conf. 5th,* pp. 2945-2954.

AN ARTIFICIALLY GENERATED ATMOSPHERE NEAR A LUNAR BASE

N93-17456

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We discuss the formation of an artificial atmosphere generated by vigorous lunar base activity in this paper. We developed an analytical, steady-state model for a lunar atmosphere based upon previous investigations of the Moon's atmosphere from Apollo. Constant gas-injection rates, ballistic trajectories, and a Maxwellian particle distribution for an oxygen-like gas are assumed. Even for the extreme case of continuous He^3 mining of the lunar regolith, we find that the lunar atmosphere would not significantly degrade astronomical observations beyond about 10 km from the mining operation.

INTRODUCTION

At present, the Moon has a tenuous, low-density atmosphere (10^5 particles/cm^3, <100 ions/cm^3) with a vacuum that is better than that in the best ground-based vacuum chambers. This atmosphere arises from impacts of solar wind particles, internal degassing, and meteoritic volatilization. Gas removal mechanisms such as thermal evaporation and solar wind electrodynamic stripping are sufficiently effective to prevent the growth of a substantial atmosphere from the natural sources of gas injection. The resulting environment is highly attractive for astronomical observations (*Burns and Mendell,* 1988) and for materials processing that requires high vacuum.

An important question to address is whether the gas and dust injected into the lunar atmosphere by exploration and colonization could overwhelm its removal and lead to a significant contamination of the lunar environment. This question was first addressed by *Vondrak* (1974) more than a decade ago. He considered the atmosphere removal rates by solar wind electrodynamic effects and thermal evaporation for a large range of atmospheric densities. He found that a long-lived "thick" atmosphere (total atmospheric mass of 10^6 kg, which is about 100 times more than at present) could form from activity generated by a vigorous lunar colony (see also *Vondrak,* 1991). This atmosphere would not dissipate for hundreds of years. It would also be optically thick to ultraviolet light, effectively rendering the lunar surface unusable for astronomy at these wavelengths.

In this paper, we present a further investigation of the artificial lunar atmosphere question. In particular, we study the growth of an atmosphere within a 200-km diameter of a lunar base. Neutral gas is injected at a constant rate, a fraction of which ionizes from solar radiation, and is allowed to evolve in time. We find that a steady state is reached with gas injection rates balancing removal rates from thermal evaporation and surface adsorption. We argue that collective plasma effects shield the atmosphere from single-particle solar wind electrodynamic forces, thus reducing ion stripping. Densities and optical depths are presented for one possible gas injection scenario.

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SOURCES AND SINKS OF GAS

Table 1 is a listing of the potential sources and sinks of gas that one might anticipate to be present in and around a thriving lunar base. The sources are described in more detail in a paper by *Taylor* (1991). The rates of gas injection are averaged over a year.

Lunar atmospheric gases are depleted by the three mechanisms listed in Table 1. Thermal evaporation occurs when the velocity of a gas particle exceeds the escape velocity of the Moon (2.4 km/sec). Thermal evaporation is not effective for particles with masses greater than He. If a gas particle has insufficient speed to escape, it will return to the Moon's surface on a parabolic ballistic trajectory (since collisions are negligible). Upon striking the surface, the gas particle may be adsorbed or re-emitted.

The solar wind flows by the Moon with an average velocity of 300 km/sec. The magnetic field embedded within the wind generates an effective electric field given by $-\vec{v} \times \vec{B}$. A single electron or ion will experience an electrodynamic force ($q\vec{E}$) that will accelerate the charged particle in a direction parallel (or antiparallel depending upon the sign of the charge) to the solar wind E-field direction. The charged particle will be quickly accelerated up to the velocity of the solar wind (within 1 Larmor period). This acceleration can be directed toward the lunar surface (at a time when the wind velocity is parallel to a tangent line of the surface at the lunar base) or parallel to the surface (when it is solar noon) in the two extremes. Thus, the wind could serve to strip charged particles out of the atmosphere and into the wind or drive them back into the surface. It is interesting to note that the acceleration of a single oxygen ion produced by the solar wind electric field is nearly 10^4 times greater than that produced by lunar gravity. *Manka and Michel* (1971) were among the first to recognize this as an important mechanism for removing gas from the lunar atmosphere.

The above scenario is applicable to situations where the density of ions is low. However, higher densities can generate important collective plasma effects that may shield the lunar atmospheric

TABLE 1. Sources and sinks of gas near a lunar base.

Sources	Rate (kg/sec)	Sink	Effectiveness
Meteoritic Volatilization	2×10^{-3}	Thermal Escape	Low to Moderate
Internal Degassing	$<3 \times 10^{-4}$	Solar Wind	Moderate
Solar Wind	5×10^{-2}	Soil Adsorption	Low to Moderate
Rocket Exhaust	10^{-2} - 10^{-1}		
Habitat Venting	5×10^{-4}		
Mining and Manufacturing			
Oxygen Production	10^{-3}		
Helium Mining	1		
Glass Production	10^{-5}		

gas from the solar wind electric field. The Debye shielding length is given by (e.g., *Chen*, 1984)

$$\lambda_D = 6.9(T/n_i)^{1/2}\ \text{cm} \tag{1}$$

For ambient temperatures, T, of 300 K and densities as low as those in current lunar ionosphere ($n_i = 100$ ions cm^{-3}), λ_D is 12 cm. As shown below, the density of an artificial atmosphere is expected to be considerably larger than that of the current atmosphere near the source and thus the Debye length will also be much smaller. Thus, even using these conservative figures, solar wind electric fields will be shielded out on distances short in comparison to the scale height of the atmosphere. For neutral particle injection rates of 10^{-3} kg/sec (e.g., oxygen production in Table 1) and a solar ionization rate of 5×10^{-6} ions/atom/sec (*Freeman et al.*, 1973), there will be 2×10^{18} ions in the atmosphere within 1 sec. This will be more than adequate to produce strong plasma shielding effects.

The above analysis is applicable for a constant electric field. However, the interaction of the solar wind and charged particle gases near a lunar base is variable in time. A more appropriate quantity is the dielectric constant of the plasma given by (e.g., *Chen*, 1984)

$$\epsilon = 1 + (c/v_A)^2 \tag{2}$$

where c is the speed of light and v_A is the Alfven speed in the plasma cloud. If the embedded magnetic field is taken to be that of the solar wind (5×10^{-5} G), then the dielectric constant is $7.6 \times 10^6 n_i$. The resulting electric field that the ions will feel inside the cloud is E_{sw}/ϵ where E_{sw} is the intrinsic solar wind E-field. Thus, for all intents and purposes, the ionic gas cloud will not be affected by the external electric field.

Both the electric and magnetic fields from the solar wind will be diverted around the expanding gas cloud near a lunar base. This will produce a macroscopic pressure (Maxwell stresses of the fields) that will tend to accelerate the cloud in bulk in the direction of the solar wind. One way of describing how this momentum transfer can occur is to construct a characteristic time, t_{acc}, during which the cloud will be accelerated to the solar wind speed (*Haerendel et al.*, 1986)

$$\frac{dv_c}{dt} = \frac{v_{sw} - v_c}{\tau_{acc}} \tag{3}$$

This characteristic time is given by

$$\tau_{acc} = \frac{\rho_c l_c}{4\kappa \rho_{sw} v_A} \tag{4}$$

where ρ_c and ρ_{sw} are the cloud and solar wind densities, respectively, v_A is the solar wind Alfven speed (50 km/sec), l_c is the cloud diameter, and κ is a magnetic field compression factor on the leading edge of the cloud (generally between 5 and 20). Equation (4) can be rewritten in terms of parameters similar to those applicable to the artificial atmosphere discussed in the next section

$$\tau_{acc} = 300\ \text{yr} \left[\frac{n_c}{10^8\ \text{cm}^{-3}}\right] \left[\frac{l_c}{10\ \text{km}}\right] \left[\frac{\kappa}{10}\right]^{-1} \tag{5}$$

Thus, timescales for such bulk acceleration will be hundreds of years and, to first order, solar wind stripping is not a factor at these densities.

This collective plasma shielding was recently demonstrated to be effective in an environment similar to that on the surface of the Moon. A cloud of 2 kg of Ba vapor was released into the solar wind in 1984 and 1985 (*Valenzuela et al.*, 1986). The Ba ionized very quickly allowing the researchers to study the interaction of a highly supersonic, dilute, magnetized plasma (solar wind) with a stagnant, unmagnetized cloud. Both plasmas were collisionless. Measurements of densities, velocities, and magnetic fields demonstrated that the shielding effects discussed above became effective in <1 min. However, ion extraction produced by an antisunward-pointing polarization electric field was more effective than had been anticipated. This resulted in a smaller lifetime of the Ba plasma cloud than had been calculated.

The interaction of the solar wind with an ionized cloud is clearly more complex than simple single-particle electric field interactions that have been previously used. For the preliminary calculations that we describe below, the solar wind interactions are ignored. This results in an upper limit to the artifical atmosphere density, and thus strengthens our conclusions described below that pollution of the lunar atmosphere is not a serious problem.

FORMATION OF AN ARTIFICIAL ATMOSPHERE

We now consider a simple analytical model for the formation of an artificial lunar atmosphere. There is a large body of work that exists on the generation of atmospheres on the Moon and Mercury (e.g., *Hodges*, 1974, 1975; *Hartle and Thomas*, 1974; *Lindeman et al.*, 1974; *Potter and Morgan*, 1988). These models have been very successful in reproducing basic observations such as the gas ion flux observed by the Apollo Suprathermal Ion Detector Experiment following the impact of the Apollo 13 upper stage on the lunar surface. We have attempted to use relevant components of these previous models and apply them directly to

an assessment of the effects of atmospheric modification on astronomical observations. The details of these calculations appear elsewhere (*Fernini et al.,* 1990). In this paper, we describe an overview of the model assumptions and present examples of how the lunar atmosphere will be affected by continuous mining and manufacturing near a lunar base.

We have made the following assumptions in our model calculations. First, the gas injection rate is taken to be continuous in time. This is an approximation of a scenario in which habitat venting, mining, or manufacturing processes on the lunar surface occur over long periods of time at nearly a constant rate. As is described below, this assumption leads to a steady-state atmosphere near the lunar base.

Second, the ejected gas is assumed to be collisionless. Gas particles will have ballistic trajectories through the exosphere under the lunar gravitational force. For an oxygen-type atmosphere, the mean free path is about $10^{16}n^{-1}$ cm, where n is the atmospheric density. At present, the mean free path is about 10^{11} cm. So, even for large increases in atmospheric density, the atmosphere is expected to remain collisionless.

Third, only a neutral atmosphere is considered. Nonthermal escape mechanisms such as photoionization and removal by the solar wind are neglected. This will be true near the source where the plasma shielding is most effective. However, further out where the density drops, electrodynamic stripping by the solar wind electric field will become important. Thus, our model represents an upper limit to the expected atmospheric density.

Fourth, the gravitational acceleration is assumed to be constant (i.e., flat Moon approximation). This approximation has been used extensively in past models (e.g., *Hodges,* 1972; *Lindeman et al.,* 1974) where localized sources of gas have been considered. Within the volume surrounding the lunar base that we consider (a box centered on the gas source with dimensions of 200 km along the lunar surface and an altitude of 100 km), the real gravitational acceleration of the Moon is within 10% of the constant value of 1.62 m/sec^2 that we assume. *Hodges and Johnson* (1968) have shown that this assumption limits the applicability of results to gases with smaller scale heights; however, this limitation affects only lunar hydrogen and helium. Since an oxygen-like atmosphere is the only one being considered in our models, we believe that this approximation is reasonable. We emphasize here that we are only interested in the atmospheric conditions near a lunar base. Thus, the model predictions are limited to the vicinity of the base.

Fifth, the particle distribution is Maxwellian characterized by the source or surface temperature. The particles are also assumed to have initially isotropic trajectories. For our models, the temperature is assumed to be constant in the calculation.

In our model calculations, we considered two types of gas transport. The first, which we term direct transport, involves simple direct flux transport of particles released from a point source on the lunar surface. The particles are assumed to be premanently adsorbed by the surface on the first contact. Here we follow the techniques outlined by *Hodges et al.* (1972) and *Lindeman et al.* (1974). In this model, the rate of atmosphere growth increases precipitously for the first few seconds, levels off, then sharply drops toward zero. This implies that a steady-state balance is reached between the rate of particle injection and particle loss through adsorption and escape from our grid. This equilibrium is reached about 20 minutes after the source is turned on. (Since the gas is collisionless, this timescale is independent of gas injection rate.)

The second transport mechanism is termed diffusion. In this model, the particles are allowed to "bounce" off the surface an indefinite number of times. Since the adsorption lifetime of an oxygen-like molecule is short ($<10^{-2}$ sec at T = 100 K), repeated bounces of a given molecule into the atmosphere are likely. *Hall* (1973) recognized this process on the Moon to be similar to a one-dimensional diffusion problem as originally described by *Chandrasekhar* (1943). We assume that upon adsorption, the particle is immediately re-emitted isotropically at a temperature characteristic of the lunar surface.

We find that for moderate mass molecules (i.e., oxygen) and for small distances from the injection point (<20 km), direct transport is the dominant process at T = 100 K. For higher temperatures, direct transport dominates out to larger distances (e.g., 50 km for T = 300 K). For larger distances, smaller temperatures, and heavier molecules, diffusion dominates. So, for a complete model of an artificial atmosphere, one must consider both transport processes.

DISCUSSION

We now apply the above models to a particular gas injection scenario. We choose the most extreme example of which we are aware, namely that of helium production (*Wittenberg et al.,* 1986). In this scenario, large amounts of regolith must be mined to obtain enough helium for fusion energy back on Earth. The rate listed in Table 1 assumes that 10% of the gas in the mining process is lost to the atmosphere. This produces an injection rate of 2×10^{25} part/sec. For this example, we also consider the outgassing to consist of molecular oxygen (although a range of other gas products is expected, oxygen is representative) and the surface/source temperature to be 100 K (lunar night).

In Table 2, we list the gas density, column density (approximated as the nH where H is the atmospheric scale height, which is 30 km for O_2 at 100 K), and optical depth at ultraviolet wavelengths for three distances from the source, 1 km, 5 km, and 60 km. It is clear from Table 2 that beyond a few to 10 km from a lunar base, the atmosphere will be significantly denser than at present, but it remains sufficiently transparent to conduct optical/ultraviolet astronomy.

We also list in Table 2 the plasma frequency of the lunar ionosphere for the above model. We have roughly estimated the density of ions/electrons in the atmosphere by assuming a constant rate of ionization from solar photons, 5×10^{-6} ions/atom/sec (e.g., *Freeman et al.,* 1973). The density of the ionosphere will determine the minimum frequency for radio observations from the surface. The ionosphere will reflect radio waves with frequencies less than $9n_e^{1/2}$ kHz. For densities $>10^4$ electrons/cm^3, the ionosphere will not transmit radio waves with frequencies below 1 MHz. This is a crucial wavelength window for radio astronomical observations from the lunar farside (*Douglas and Smith,* 1985). It would appear, then, that a very low frequency radio astronomy observatory should be located beyond about 5 km from the helium production facility.

CONCLUSIONS

Our analysis of the growth of an artificial lunar atmosphere suggests that the astronomical environment beyond about 10 km from mining or manufacturing operations will be relatively unaffected by the resulting outgassing. The transparency of the atmosphere remains high for very low frequencies and ultraviolet wavelengths.

We believe that our simple atmospheric models are reasonable order-of-magnitude estimates of the lunar gas densities. In the limits of smaller injection rates, our models agree very well with measurements from Apollo experiments. However, our models are only approximations since we have made several simplifying assumptions including constant source/surface temperatures and no losses due to stripping by the solar wind. The latter, however, can only strengthen our conclusion.

Our models are also limited to the direct vicinity of a lunar base. We have not considered global additive effects such as multiple mining operations. These collective effects could potentially increase the overall atmospheric density, but this appears unlikely unless there are large numbers of such facilities with increased outgassing as assumed by *Vondrak* (1974). According to our models, each such facility is ineffective near the boundaries of our grid, with outer densities nearly equal to that of the current atmospheric density. Diffusion of gas between the individual sites and around the Moon is not likely to be important since solar wind stripping becomes more effective for longer particle lifetimes (greater probability of ionization) and at lower densities (less shielding by collective plasma effects). This process has been effective in the past in keeping the lunar atmospheric density low in spite of multiple sites of meteoritic volatilization.

In summary, it appears that a vigorous lunar base will not impede astronomical observations from the lunar surface. Our conclusions can be tested quantitatively by gas release experiments during lunar precursor or early lunar base missions.

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REFERENCES

Burns J. O. and Mendell W. W., eds. (1988) *Future Astronomical Observatories on the Moon.* NASA CP-2489. 134 pp.

Chandrasekhar S. (1943) Stochastic problems in physics and astronomy. *Rev. Mod. Phys., 15,* 1-89.

Chen F. F. (1984) *Plasma Physics and Controlled Fusion,* 2nd edition. Plenum Press, New York. 421 pp.

Douglas J. N. and Smith H. J. (1985) A very low frequency radio astronomy observatory on the Moon. In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 301-306. Lunar and Planetary Institute, Houston.

Fernini I., Burns J., Taylor J., Sulkanen M., Duric N., and Johnson S. (1990) Disposal of gases generated near a lunar outpost. *J. Spacecr. Rockets, 27,* 527-538.

Freeman J. W., Hills H. K., Lindeman R. A., and Vondrak R. R. (1973) Observations of water vapor ions at the lunar surface. *Moon, 8,* 115-128.

Haerendel G., Paschmann G., Baumjohann W., and Carlson C. W. (1986) Dynamics of the AMPTE artificial comet. *Nature, 320,* 720.

Hall F. G. (1973) Role of pressure transients in the detection and identification of lunar surface gas sources. *J. Geophys. Res., 78,* 2111-2132.

Hartle R. E. and Thomas G. E. (1974) Neutral and ion exosphere models for lunar hydrogen and helium. *J. Geophys. Res., 79,* 1519-1526.

Hodges R. R. (1972) Response of lunar atmosphere to volcanic gas releases. *Planet. Space Sci., 20,* 1849-1864.

Hodges R. R. (1974) Model atmospheres for Mercury based on a lunar analogy. *J. Geophys. Res., 79,* 2881.

Hodges R. R. (1975) Formation of the lunar atmosphere. *Moon, 14,* 139.

Hodges R. R. and Johnson F. S. (1968) Lateral transport in planetary exospheres. *J. Geophys. Res., 73,* 7307.

Hodges R. R., Hoffman J. H., Yeh T. T. J., and Chang G. K. (1972) Orbital search for lunar volcanism. *J. Geophys. Res., 77,* 4079-4085.

Lindeman R. A., Vondrak R. R., Freeman J. W., and Snyder C. W. (1974) The interaction between an impact-produced neutral gas cloud and the solar wind at the lunar surface. *J. Geophys. Res., 79,* 2287-2296.

Manka R. H. and Michel F. C. (1971) Lunar atmosphere as a source of lunar surface elements. *Proc. Lunar Sci. Conf. 2nd,* pp. 1717-1728.

Potter A. E. and Morgan T. H. (1988) Discovery of sodium and potassium vapor in the atmosphere of the Moon. *Science, 241,* 675-680.

Taylor G. J. (1992) Astronomy on the Moon: Geological considerations. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Valenzuela A., Haerendel G., Föppl H., Melzner F., Neuss H., Rieger E., Stöcker J., Bauer O., Höffner H., and Loidl J. (1986) The AMPTE artificial comet experiments. *Nature, 320,* 700-703.

Vondrak R. R. (1974) Creation of an artificial lunar atmosphere. *Nature, 248,* 657-659.

Vondrak R. R. (1992) Lunar base activities and the lunar environment. In *The Second Conference on Lunar Bases and Space Activities of the 21st Century,* this volume.

Wittenberg L. J., Santarius J. F., and Kulcinski G. L. (1986) Lunar source of ^{3}He for commercial fusion power. *Fusion Technol., 10,* 167.

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A MANNED EXOBIOLOGY LABORATORY BASED ON THE MOON

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INTRODUCTION

Establishment of an exobiology laboratory on the Moon would provide a unique opportunity for exploration of extraterrestrial materials on a long-term, ongoing basis, for elucidation of exobiological processes and chemical evolution.

A major function of the lunar exobiology laboratory would be to examine samples collected from other planets (e.g., Mars) for the presence of extant or extinct life. By establishing a laboratory on the Moon, preliminary analyses could be conducted away from Earth, thus establishing that extraterrestrial materials are benign before their return to Earth for more extensive investigations.

Unmanned missions to other planets will have a capability for detecting extant life (biology experiments), but less capability for detecting extinct life or for determining the level of prebiotic chemical evolution. Prior to return of samples from another planet to the Earth or lunar laboratory, data from instrumentation on the unmanned missions will be available for analysis. The results from experiments to detect life will be available, as well as data on the organic carbon content of the material to be returned. A spectrum of possibilities then exists concerning these data, from results indicating extant life to data strongly suggesting the absence of significant chemical evolution (e.g., mass spectrometric data indicating the absence of organic carbon).

In the case that extant life is indicated, or that substantial amounts of organic carbon of relatively high molecular weight are found, the return of the sample to a Moon-based laboratory for preliminary study would provide a buffer to terrestrial exposure until the returned samples have been characterized and proven to be nonhazardous. In addition, a wealth of soil samples and cores would be available for study by the laboratory. Immediately at hand would be a wide variety of lunar sample types and amounts as well as meteoritic material; thus, there would be no restriction on sample availability for analysis at the part per billion (ppb) level.

EXPERIMENTS

Initial studies would focus on the search for extant life and would include an array of biological, physical, and chemical studies. If extant life is found in planetary returned samples, the lunar laboratory would perform a variety of investigations to characterize the life form(s) prior to return of the material to laboratories on Earth.

The Moon-based exobiology laboratory would have three major components for study of samples returned from other planets.

1. *The search for extant life.* This component would focus on the detection and identification of life forms using biological, physical, and chemical methods.
2. *The search for extinct life.* This component would concentrate on identification of extinct life using micropaleontological physical and chemical means.
3. *The search for evidence of chemical evolution.* This component would be devoted to the detection and identification of molecules revealing prebiotic chemical evolution. Amino acids/peptides, nucleobases/nucleosides/nucleotides/nucleic acids, and other classes of biologically important molecules would be sought and characterized. This component would also be an important resource in the investigations for extant and extinct life.

INSTRUMENTATION

Although analytical techniques will advance considerably by the date of development of the Moon-based laboratory, the major techniques of chromatographies interfaced with mass spectrometry will be used to separate, identify, and measure the molecules and isotopes of exobiological interest. Miniaturization of these and other analytical instrumentations will provide scientific investigators in the lunar exobiology laboratory with the necessary research tools for identification and structural characterization of organic molecule classes in a wide array of extraterrestrial samples.

BACKGROUND

Our studies (1969-74) of the returned lunar samples (*Gehrke et al.,* 1972, 1975, 1987; *Rash et al.,* 1972) and those of others (*Hamilton and Nagy,* 1972; *Fox et al.,* 1973) have shown the importance of acquiring pristine samples that have not been exposed to terrestrial contamination. Indeed, the question of contamination is continually an important aspect of studies of meteorites or the returned lunar samples, especially with regard to investigations of organic compounds. The establishment of a manned lunar exobiology laboratory, equipped with appropriate instrumentation and a complement of scientists would present a unique scientific opportunity for study of extraterrestrial samples from various sources.

REFERENCES

Fox S., Harada K., and Hare P. (1973) Accumulated analyses of amino acid precursors in returned lunar samples. *Proc. Lunar Sci. Conf. 4th,* pp. 2241-2248.

Gehrke C.W., Zumwalt R.W., Kuo K.C., Ponnameruma C., and Shimoyama A. (1972) Research for amino acids in lunar samples. *Space Life Sci., 3,* 439.

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Gehrke C. W., Zumwalt R. W., Kuo K. C., Ponnameruma C., and Shimoyama A. (1975) Search for amino acids in returned lunar soil. *Origins Life, 6,* 541.

Gehrke C. W., Kuo K. C., Zumwalt R. W., Ponnamperuma C., and Shimoyama A. (1987) Search for amino acids in Apollo returned lunar soil. In *Amino Acid Analysis by Gas Chromatography*, Vol. 2 (R. Zumwalt, K. Kuo, and C. Gehrke, eds.), pp. 151-163. CRC Press, Boca Raton, Florida.

Hamilton P. B. and Nagy B. (1972) Problems in the search for amino acids in lunar fines. *Space Life Sci., 3,* 432.

Rash J. J., Zumwalt R. W., Gehrke C. W., Kuo K. C., Ponnameruma C., and Shimoyama A. (1972) GLC of amino acids: A survey of contamination. *J. Chromatogr. Sci., 10,* 444.

POSSIBLE BIOMEDICAL APPLICATIONS AND LIMITATIONS OF A VARIABLE-FORCE CENTRIFUGE ON THE LUNAR SURFACE: A RESEARCH TOOL AND AN ENABLING RESOURCE

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Centrifuges will continue to serve as a valuable research tool in gaining an understanding of the biological significance of the inertial acceleration due to gravity. Space- and possibly lunar-based centrifuges will play a significant and enabling role with regard to the human component of future lunar and martian exploration, both as a means of accessing potential health and performance risks and as a means of alleviating these risks. Lunar-based centrifuges could be particularly useful as part of a program of physiologic countermeasures designed to alleviate the physical deconditioning that may result from prolonged exposure to a ⅙-g environment. Centrifuges on the lunar surface could also be used as part of a high-fidelity simulation of a trip to Mars. Other uses could include crew readaptation to 1 g, waste separation, materials processing, optical mirror production in situ *on the Moon, and laboratory specimen separation.*

HUMANS AND GRAVITY LEVELS OF LESS THAN 1 *g*

Spaceflight removes living organisms from the *only* environmental factor whose strength and direction have remained constant throughout life's tenure on Earth: gravity (*Gordon and Chen,* 1970; *Halstead,* 1987; *Pestov and Gerathewohl,* 1975; *Parfyonov,* 1983; *NASA Office of Space Science and Applications,* 1987). Living organisms exhibit a wide range of sensitivities to a variably perceptible gravitational/accelerative vector. This range of responses is seen both between individual species and among various physiologic systems within individual organisms (*Pestov and Gerathewohl,* 1975; *NASA Office of Space Science and Applications,* 1987; *Fuller et al.,* 1987; *Schopf,* 1988).

Of particular interest to the human exploration of the Moon (and eventually Mars and other planets) is the effect of prolonged exposure both to weightlessness (or, more accurately, the accelerative unloading that occurs upon exposure to a microgravity environment) during cislunar transit and to gravitational levels greater than microgravity but less than 1 *g* (hypogravity) upon a planetary surface (*Nicogossian and Parker,* 1982; *Ride,* 1987).

Current knowledge regarding the effects of prolonged exposure to microgravity has a temporal limit in the 326-day exposure of one cosmonaut (*Schneider et al.,* 1988) and that of two other cosmonauts who spent 365 days in orbit aboard the MIR space station (*Covault,* 1988). The data regarding shorter exposures to microgravity (on the order of days or weeks), while greater in quantity, still leave us with more questions than answers (*Schneider et al.,* 1988; *Covault,* 1988). In the matter of prolonged exposure to lunar gravity, the data are limited to that gained from the exposure of 14 individuals with a maximum exposure (during the Apollo 17 mission) of 2 individuals to 75 hr on the lunar surface, of which 22.1 hr were spent in extravehicular activity (EVA) (*Pestov and Gerathewohl,* 1975; *Nicogossian and Parker,* 1982; *NASA Johnson Space Center,* 1973).

While the physiological aspects of microgravity exposure have been only partially investigated, even less is known about the gravitational regime that lies between microgravity and 1 *g* (*NASA Office of Space Science and Applications,* 1987; *Fuller et al.,* 1987; *Schopf et al.,* 1988; *Nicogossian and Parker,* 1982; *Schneider et al.,* 1988). Significant questions remain as to what exposure times and level(s) of gravity or other forms of inertial acceleration are necessary for the maintenance of normal physiologic activity (*Schopf et al.,* 1988; *Schneider et al.,* 1988; *National Research Council,* 1987a; *Smith et al.,* 1986). In the course of further research it may be discovered that some activities such as neurovestibular function adapt readily to ⅙ *g* and remain pliant enough to allow full readaptation to 1 *g*. Other activities such as bone mineral homeostasis, muscle function and maintenance, and cardiovascular function may be found to maladapt to ⅙ *g* and may not completely readapt once an astronaut returned to a 1-*g* environment.

CENTRIFUGES AS RESEARCH TOOLS

Identifying the physiologic threshold(s) of gravitational loading sensitivity, the character and course of response, perturbation, adaptation, and the development of countermeasures requires the capability to expose humans and other species to variable levels of inertial acceleration. The effects of gravity must also be able to be removed as a variable in order to identify the involvement of other spaceflight factors (e.g., radiation, vehicle disturbances,

and environmental contaminants) in observed phenomena (*NASA Office of Space Science and Applications,* 1987; *Fuller et al.,* 1987; *Schopf et al.,* 1988; *Schneider et al.,* 1988; *National Research Council,* 1987a). There are three ways to do this: (1) conduct research on a planet with the desired gravity level (impractical and, in the case of humans, unethical and potentially hazardous); (2) place specimens in a spacecraft and then accelerate them in a linear trajectory at a constant velocity (impractical and prohibitively expensive); or (3) use centrifugation. Centrifuges have been, and will continue to be, the favored means of producing an "artificial" gravity inasmuch as they can provide a practically available substitute in space for the level of inertial acceleration due to gravity experienced upon the Earth's surface, with the added capability to expose organisms to hypogravity environments, which are not possible to produce upon the Earth's surface (*Fuller et al.,* 1987).

To date, only small-radius centrifuges have been flown on far too few occasions as on-orbit controls, e.g., on the German D-1 Spacelab Mission (STS 61-A) (*Fuller et al.,* 1987; *Schopf et al.,* 1988; *Schneider et al.,* 1988; *National Research Council,* 1987a; *Mesland,* 1988). This will change dramatically with the advent of a 2.5-m centrifuge aboard space station *Freedom.* A variety of centrifuges will soon be available that will allow specimens up to the size of a rhesus monkey to be exposed to variable levels of artificial gravity (*NASA Office of Space Science and Applications,* 1987; *Schopf et al.,* 1988; *Schneider et al.,* 1988; *NASA Ames Research Center,* 1986; *NASA,* 1987). The capability to provide prolonged exposure of humans to variable levels of artificial gravity will have to wait until much larger radii centrifuges become available (*Schopf et al.,* 1988; *Schneider et al.,* 1988; *National Research Council,* 1987a; *Healy et al.,* 1988; *Sanford et al.,* 1988; *Burton,* 1988). One such facility, the proposed Variable Gravity Research Facility (VGRF), would be a tethered vehicle with a radius of 100 m or more, capable of exposing humans to various levels of artificial gravity for prolonged periods of time. Research aboard such a VGRF or other facility equipped with large-radii centrifuges would make it possible to identify the effects of long-term lunar (and martian) gravity and would allow countermeasures to be developed before humans return to the Moon or explore Mars (*Schopf et al.,* 1988; *Schneider et al.,* 1988; *National Research Council,* 1987a; *Healy et al.,* 1988; *Sanford et al.,* 1988; *Lemke,* 1988; *National Commission on Space,* 1986).

CENTRIFUGES AS COUNTERMEASURES AGAINST PHYSICAL DECONDITIONING

Centrifuges may have a second role in addition to their importance as a research tool, that of a countermeasure to lessen or eliminate the deleterious effects of prolonged exposure to microgravity and hypogravity (*Fuller et al.,* 1987; *Schopf et al.,* 1988; *Schneider et al.,* 1988; *Healy et al.,* 1988; *Sanford et al.,* 1988; *Nicogossian and McCormack,* 1987). Having such a countermeasure available in Earth orbit or on the Moon may serve as an enabling factor that greatly extends the otherwise limited ability of humans to work for long periods on the lunar surface. Two possible approaches that might be taken are (1) intermittent exposure to 1 g (or higher levels) for short periods, which might be sufficient to eliminate or reduce deconditioning to a manageable level, or (2) continuous exposure at 1 g or less, which may also produce the same effect (*Lemke,* 1988). Before either approach can be implemented a number of questions need to be answered.

1. What thresholds exist at which gravity directs or influences different physiologic functions? In other words, what levels of centrifugation might be needed (e.g., 0.5 g, 1 g, or more than 1 g) to counteract deconditioning (*Burton,* 1988)?

2. What exposure times (e.g., occasional periods of prolonged centrifugation vs. intermittent, acute exposures; *Burton,* 1988) produce the desired counteractions to deconditioning?

3. What gravity gradients, force vectors, coriolis forces, and cross-coupled accelerations would be acceptable to humans, and what countermeasures (e.g., exercise, nutritional supplements, pharmacological agents) might need to be used in conjunction with centrifugation (*Pestov and Gerathewohl,* 1975; *NASA Office of Space Science and Applications,* 1987; *Fuller et al.,* 1987; *Schopf et al.,* 1988; *Nicogossian and Parker,* 1982; *Ride,* 1987; *Schneider et al.,* 1988; *Healy et al.,* 1988; *Sanford et al.,* 1988; *Nicogossian and McCormack,* 1987)?

Research aboard space station *Freedom,* and later aboard a potential VGRF-like facility, should provide information as to the course and character of any deconditioning that may occur after prolonged exposure to variable gravity levels such as $1/6$ g. If it is determined that chronic exposure to $1/6$ g does have deleterious consequences that can be ameliorated by centrifugation, the question then arises as to what centrifugation capabilities need to be provided.

LIMITATIONS TO THE USE OF CENTRIFUGES WITH HUMANS

A number of potentially undesirable factors that result from centrifugation need to be taken into account when considering the use of centrifugation as a countermeasure. The biological effects of "artificial" gravity produced by centrifugation, while indeed similar to those experienced by organisms undergoing constant linear acceleration on a planetary body or within a spacecraft, also differ in many significant ways. Organisms undergoing centrifugation experience a constant change in their direction without a change in their rate of motion, as opposed to what occurs within a planetary gravitational field where an organism experiences a change in its motion without a change in direction (*Fuller et al.,* 1987; *Smith,* 1975). The size of a centrifuge must be carefully considered when large animals such as humans are concerned. As the radius of a centrifuge is decreased, the rate at which it must rotate to maintain a given force at its rim must be increased. The smaller the radius, the more pronounced the gravity gradient from its axis of rotation outward to its rim becomes. For upright humans with their feet on the inside of a centrifuge's rim, a gradient will always exist with the greatest force felt at the subjects' feet and the least force felt at their heads. Again, the smaller the centrifuge radius, the more pronounced and troublesome the gradient becomes (*Fuller et al.,* 1987; *Smith,* 1975; *Graybiel,* 1975). If a centrifuge is to be designed to mimic the accelerations experienced by a human on a planetary surface (where a gravity gradient, while present, is infinitesimally small and probably not detectable), then the radius must be extremely large. Such would be the case in a facility like the proposed VGRF (*Lemke,* 1988).

There are, however, other factors to be considered when designing a centrifuge for human use. Troublesome problems arise when humans are exposed to radial acceleration: coriolis forces. If a person were to attempt to move tangentially or radially in a linear fashion (i.e., walk) within the rotating environment of a centrifuge, they would experience a force (depending upon

whether they were walking with or against the rotation of the centrifuge) that would cause them to tend to veer off at an angle to the straight line that they would otherwise expect to travel. The slower the rate of rotation, the less of a problem this becomes. Coriolis effects could be expected to affect all crew movements within a centrifuge (*Nicogossian and McCormack,* 1987; *Graybiel,* 1975).

A more severe problem would arise if a person attempted angular motions, most notably moving his head out of the axis of the centrifuge's rotation. The angular acceleration of the centrifuge becomes cross coupled with that produced by head movement and often leads to motion sickness (*Nicogossian and McCormack,* 1987). The faster a centrifuge rotates, the more severe these effects become. It is uncertain at present if humans can adapt to such vestibular perturbations (*Nicogossian and McCormack,* 1987; *Graybiel,* 1975). This problem is also an argument for reducing the rotation rate as much as possible, which, if significant gravity levels are desired, requires large-radii centrifuges.

If it is deemed necessary to expose humans to frequent centrifugation to counteract microgravity- or hypogravity-induced deconditioning, then it will be important to minimize the effects of gravity gradients, coriolis forces, and cross-coupled angular accelerations, especially if the persons within the centrifuge need the freedom to move and work. As such, an extremely long radius (100 m or more) at low rotation rates (1 rpm) would be required (*Lemke,* 1988; *Nicogossian and McCormack,* 1987). Such is the case with a facility like the proposed VGRF or a large, rotating interplanetary spacecraft (*Schneider et al.,* 1988). If deconditioning from chronic exposure to $\frac{1}{6}g$ can only be treated with continual centrifugation (which would make human presence on the lunar surface impractical), the other preventative or therapeutic countermeasures will need to be developed such as pharmacological agents, dietary supplements, or exercise regimens.

CENTRIFUGES AS PART OF A LUNAR HEALTH MAINTENANCE CAPABILITY

It has been suggested, based upon microgravity- and hypergravity- (where gravity is $>1\,g$) based research that exercise and pharmacological countermeasures, combined with intermittent exposure to gravity levels approaching $1\,g$ (or more), may suffice to reduce problems such as bone loss and muscle atrophy to tolerable levels (*Lemke,* 1988; *Nicogossian and McCormack,* 1987). Such exposures can be provided with much smaller centrifuges (radii of several meters) rotating at higher rates (10 rpm or more) (*Nicogossian and Parker,* 1982). The use of small centrifuges would be feasible if it could be shown that the troublesome effects of gravity gradients, coriolis forces, and cross-coupled angular accelerations can be controlled. This could be accomplished by limiting exposures to periods of the day when physical activity could be minimized. This could be done during periodic rest periods or even during sleep. Evidence exists from human experiments within a ground-based, 2-m, rapidly rotating (23.2 rpm) centrifuge that sleeping (and moving while asleep) does not produce any noticeable or unpleasant side effects (*Graybiel,* 1975).

Centrifuges may therefore need to be incorporated into the design of the earliest long-term outposts on the Moon. Crews could be assigned specific "centrifuge time" during breaks in their work cycle or be required to spend a certain number of sleep periods under centrifugation (*Burton,* 1988). As with all medical treatments, a wide range of responses can exist between individuals. Frequent monitoring of crew health would be needed to determine the effectiveness of centrifugation as well as other possible adjunct countermeasures.

Living on the Moon will undoubtedly lead to crew members developing new instincts unique to living in a $\frac{1}{6}$-g environment. This was clearly evident with regard to the peculiar gait adopted by the Apollo astronauts as they worked on the lunar surface (*Pestov and Gerathewohl,* 1975). Learned behaviors acquired from long periods on the Moon such as those acquired from surface EVAs and other activities, e.g., the moving of large pieces of equipment, might be difficult to unlearn and therefore potentially hazardous once an astronaut returned to Earth. Humans have been shown to readapt to terrestrial conditions very quickly after short-term exposures to microgravity. As microgravity exposure times increase, the time needed for readaptation increases. This may also prove true with regard to extended stays on the lunar surface. Lunar-based centrifuges could also be used to help crew members readapt their motor skills to 1-g conditions before returning to Earth, thus reducing the possible time needed to readapt to terrestrial conditions after their return. Such a readaptive capability would probably require the construction of much larger centrifuges to minimize the deleterious characteristics inherent in the smaller centrifuges that might be used as part of deconditioning countermeasures. While this concern should be considered, it would probably not need to be implemented at the onset of lunar base operations, that is, at least until tours of duty became extremely long. From a logistical point of view, it may turn out to be more practical for crew members to spend some time aboard a successor to the proposed VGRF in low Earth orbit before returning to Earth rather than in a large (and no doubt expensive) centrifuge on the Moon.

USE OF A LUNAR-BASED CENTRIFUGE IN A SIMULATED MARS MISSION

Planning both the human and hardware components of any Mars mission will undoubtedly be preceded by extensive simulations (*Ride,* 1987; *Schneider et al.,* 1988). Another use for a lunar-based centrifuge could be as part of a high-fidelity simulated mission to Mars. A possible scenario might be as follows: Astronauts would spend a simulated outbound trip aboard a VGRF-like facility outfitted with a large-radius centrifuge to simulate an artificial-gravity-equipped Mars spacecraft. Crews would be transferred to lunar orbit where they would be subjected to a lunar descent with a high-gravity profile similar to that expected when landing on Mars. Crews would then be housed in a lunar-based facility containing a centrifuge capable of being spun up to the equivalent of $0.38\,g$, wherein the crew would sleep and perform other activities not requiring substantial movement. The facility would have a coupling that could be despun to allow the crew to perform EVAs on the lunar surface with spacesuits and equipment weighted down to allow crews to experience a close approximation of the work loads that would accompany surface activity on Mars at $0.38\,g$. Activities within the lunar facility requiring significant movement would likewise be performed with appropriately weighted garments. Such a simulation would be faced with the obvious human factor constraints imposed by living within a centrifuge on a planetary surface and those associated with wearing weighted garments, and would most likely find the greatest applicability with the

simulation of relatively short stay times associated with so-called Mars "Sprint" missions. The return portion of the trip could be conducted in a similar manner with crews spending additional time aboard a VGRF. Such a simulation would allow a thorough evaluation of human health and performance issues such as the need to transfer to and from an artificial gravity environment, experience transient microgravity, tolerate high-gravity accelerations after prolonged spaceflight, and perform surface activity at $0.38\,g$. Suggesting such a simulated mission presumes the existence of a vigorous program of lunar exploration with an already established infrastructure, one that is designed to incorporate evolutionary explorative activities and therefore capable of being substantially augmented for an activity of this nature (*Ride,* 1987; *National Commission on Space,* 1986).

GRAVITATIONAL BIOLOGY RESEARCH ON THE MOON

Research into the physiological mechanisms whereby plants and animals sense and respond to gravity is a prime justification for the existence of space-based research facilities (*Halstead,* 1987; *Fuller et al.,* 1987; *Ride,* 1987; *Schneider et al.,* 1988; *National Research Council,* 1987a,b; *Smith and Fuller,* 1986; *Lemke,* 1988). Extensive research will continue to be performed in space before and, indeed, in preparation for the large-scale human exploration of the Moon and Mars (*Ride,* 1987; *Schneider et al.,* 1988; *National Commission on Space,* 1986). As lunar exploration in particular proceeds, there will be an ongoing need to build upon this body of knowledge by conducting additional basic and applied research, e.g., refining the effectiveness of human countermeasures and enhancing the growth of lunar food crops (*Galston et al.,* 1988). Such work will be done by researchers, many of whom will undoubtedly be located on the lunar surface themselves. As has been the case in microgravity-based research, centrifuges will be a useful tool on the Moon. There will, however, be some limitations to their use. The ubiquitous presence of the Moon's gravity will limit the gravitational forces available to levels above $\frac{1}{6}\,g$. In addition, lunar gravity will add a potentially unwanted acceleration vector to any specimen within a lunar-based centrifuge, unwanted in the sense that many of the biological processes studied by gravitational biologists are exquisitely sensitive to gravity and can only be studied under very low and precisely defined levels of gravity.

MATERIALS PROCESSING AND SEPARATION

Centrifuges also have potential applications outside biomedical research. One such application might be separating waste materials as part of a Closed Ecological Life Support System (CELSS). Another application will certainly be the use of ultracentrifuges in a clinical setting not unlike those currently used on Earth. Additional uses might be found in manufacturing, inasmuch as many manufacturing and refining processes rely upon the separation of materials of different densities (*Waldron,* 1988; *Schlichta,* 1988). Some processes, which may prove inefficient in a gravitational field of $\frac{1}{6}\,g$, may be enhanced by the use of centrifugation at levels equal to or above $1\,g$. A particularly important application might be found in the production of lunar-based construction materials (*Khalili,* 1985), cryogenic fuel production, and perhaps the casting of large optical mirrors for lunar-based telescopes (*Anderson,* 1988).

CONCLUDING REMARKS

Humans have been using centrifugation in various forms as a tool for a thousand centuries. It probably began with a nomadic hunter's use of a hand-operated centrifuge (known today in various forms as a slingshot, bolo, etc.) that she or he may have used to accelerate a projectile. As time progressed, centrifuge-based inventions such as the potter's wheel, spin-dry washing machines, and high-speed gas centrifugation for the isolation of fissionable U isotopes have appeared. Just as the slingshot helped to increase humanity's capability to wander across the surface of this planet, it seems befitting that the descendants of this ancient tool may allow latter-day human nomads to wander and thrive upon the surfaces of other planets.

REFERENCES

Anderson I. (1988) Angel sharpens the Vatican's vision of the heavens. *New Scientist,* May 5, 1988, p. 30.

Burton R. R. (1988) A human-use centrifuge for space stations: Proposed ground-based studies. *Aviat. Space Environ. Med., 59,* 579-582.

Covault C. (1988) Record Soviet manned space flight raises human endurance questions. *Aviat. Week Space Technol, 4.*

Diamandis P. H. (1988) The need for artificial gravity on the Moon (abstract). In *Papers Presented to the 1988 Symposium on Lunar Bases and Space Activities of the 21st Century,* p. 71. Lunar and Planetary Institute, Houston.

Fuller C. A., Sulzman F. M., and Keefe J. R. (1987) Artificial gravity: The evolution of variable gravity research (abstract). In *Papers Presented to the 38th Congress of the International Astronautical Federation.* IAF-87-539.

Galston A. W., Vitousek P., and Hinkle C. R. (1988) Controlled Ecological Life Support System (CELSS). In *Exploring the Living Universe* (J. H. Bredt and M. Averner, eds.), pp. 112-123. NASA Life Sciences Strategic Planning Study Committee, Washington, DC.

Gordon S. A. and Chen M. J., eds. (1970) *Gravity and the Organism.* Univ. of Chicago, Chicago. 474 pp.

Graybiel A. (1975) Angular velocities, angular acceleration, and coriolis forces. In *Foundations of Space Biology and Medicine, Vol. 2* (J. P. Marbarger and P. V. Vasil'yev, eds.), pp. 247-304. NASA, Washington, DC.

Halstead T. W., ed. (1987) *1986-87 NASA Space/Gravitational Biology Accomplishments.* NASA TM-89951.

Halstead T. W. and Dufour P. A., eds. (1986) *Biological and Medical Experiments on the Space Shuttle 1981-1985.* NASA Life Sciences Division, Washington, DC. 176 pp.

Healy B., DeCampli W., Robbins F. C., and Lockette W. (1988) Biomedical research. In *Exploring the Living Universe* (J. H. Bredt and M. Averner, eds.), pp. 40-52. NASA Life Sciences Strategic Planning Study Committee, Washington, DC.

Khalili E. N. (1985) Magma, ceramic, and fused adobe structures generated *in situ.* In *Lunar Bases and Space Activities of the 21st Century* (W. W. Mendell, ed.), pp. 399-403. Lunar and Planetary Institute, Houston.

Lemke L. (1988) VGRF technology overview and strawman design. Presentation to the Workshop on the Role of Life Science in the Variable Gravity Research Facility, NASA Ames Research Center, Moffett Field, California.

Mesland D. A. M., ed. (1988) *Biorack on Spacelab D-1.* ESA SP-1091. 182 pp.

NASA (1987) *Reference Mission Operation Analysis Document (RMOAD) for the Life Sciences Research Facilities (Revised).* NASA TM-89604. 289 pp.

NASA Ames Research Center (1986) *Life Science Research Objectives and Representative Experiments for the Space Station (Green Book).* Preliminary draft, Biological Research Project, NASA Ames Research Center, Life Science Division, Advanced Programs Office, Moffett Field, California.

NASA Johnson Space Center (1973) *Apollo 17 Preliminary Science Report.* NASA SP-330.

NASA Office of Space Science and Applications (1987) *Life Sciences Report.* NASA Office of Space Science and Applications, Life Sciences Division. 84 pp.

National Commission on Space (1986) *Pioneering the Space Frontier.* Bantam, New York. 214 pp.

National Research Council (1987a) *A Strategy for Space Biology and Medicine for the 1980's and 1990's.* National Research Council, Washington, DC. 196 pp.

National Research Council (1987b) *Report of the Committee on the Space Station.* National Academy Press, Washington, DC. 61 pp.

Nicogossian A. E. and McCormack P. D. (1987) Artificial gravity—a countermeasure for zero gravity. In *Papers Presented to the 38th Congress of the International Astronautical Federation.* IAF-87-533.

Nicogossian A. E. and Parker J. F. (1982) *Space Physiology and Medicine.* NASA SP-447.

Parfyonov G. P. (1983) Evolutionary and physiological adaptation to gravity. *Physiologist, 26,* S-57.

Pestov I. D. and Gerathewohl S. J. (1975) Weightlessness. In *Foundations of Space Biology and Medicine, Vol. 2* (J. P. Marbarger and P. V. Vasil'yev, eds.), pp. 305-354. NASA, Washington, DC.

Ride S. K. (1987) *Leadership and America's Future in Space; Report to the Administrator.* U.S. Govt. Printing Office, Washington, DC. 63 pp.

Sanford J. P., Huntoon C. L., Bennett I. L., and Linder B. J. (1988) Operational medicine. In *Exploring the Living Universe* (J. H. Bredt and M. Averner, eds.), pp. 91-100. NASA Life Sciences Strategic Planning Study Committee, Washington, DC.

Schlichta P. J. (1988) *Centrifugation Would Purify Mercuric Iodide.* JPL Invention Report NPO 16737/6254. Jet Propulsion Laboratory, Pasadena. 4 pp.

Schneider W., Collins M., Carr G., Cowing K., Leveton L., and Hobish M. (1988) Flight programs. In *Exploring the Living Universe* (J. H. Bredt and M. Averner, eds.), pp. 154-170. NASA Life Sciences Strategic Planning Study Committee, Washington, DC.

Schopf W., Galston A., and Cowing K. (1988) Gravitational biology. In *Exploring the Living Universe* (J. H. Bredt and M. Averner, eds.), pp. 101-111. NASA Life Sciences Strategic Planning Study Committee, Washington, DC.

Smith A. H. (1975) Principles of gravitational biology. In *Foundations of Space Biology and Medicine, Vol. 2* (J. P. Marbarger and P.V. Vasil'yev, eds.), pp. 129-162. NASA, Washington, DC.

Smith A. H. and Fuller C. A., eds. (1986) *Space Station Centrifuge: A Requirement for Life Science Research.* 38 pp.

Waldron R. W. (1988) *Continuous Flow Centrifugal Separator.* MSC-21173, Rockwell, Houston. 5 pp.

Appendixes

AUTHOR INDEX

SUBJECT INDEX

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LOCATION INDEX

ACRONYM GLOSSARY

ΔV	Change of Velocity
A&R	Automation and Robotics
ABL	Anthropometry and Biomechanics Laboratory
AC	Alternating Current
ACC	Aft Cargo Carrier
ACS	Attitude Control Subsystem
AEC	Atomic Energy Commision
AI	Attitude Indicator
AL	Action Limit Autolander
ALSE	Active Lunar Seismic Experiment
ALSEP	Apollo Lunar Surface Experiment Package
ALSPE	Anomalously Large Solar Proton Event
AMCD	Annular Momentum Control Device
AOL	Airborne Oceanographic Lidar
AP	Agricultural Plant
APP	Astrofuel Production plant
APS	Ascent Propulsion System
APSA	Advanced Photovoltaic Solar Array
APT	Antarctic Planetary Testbed
ARC	Ames Research Center
ARCO	Atlantic Richfield Company
ASD	Advanced Solar Dynamic
ASE	Advanced Space Engine
ASO	Active Solar Optics
ASPS	Adaptable Space Propulsion System
ASTM	American Society for Testing Materials
BMC	Bone Mineral Content
BPC	Biomass Production Chamber
C&D	Control and Display
C&T	Communications and Tracking
CCGE	Cold Cathode Gauge Experiment
CDS	Command and Data Subsystem
CEC	Cation Exchange Capacity
CELSS	Controlled Ecological Life Support System
CER	Cost Estimating Relationship
CETF	Critical Evaluation Task Force
CL	Crew Lander
CL-AS	Crew Lander-Ascent Stage
CL-DS	Crew Lander-Descent Stage
CLEFT	Cleaved Lateral Epitaxy for Film Transfer
CLP	Closed Loop Processing
CLSI	Civil Space Leadership Initiative
CM	Center of Mass Command Module
CMG	Control Moment Gyro
CNDB	Civil Needs Data Base
COBE	Cosmic Background Explorer
COE	Cost of Energy
CPLEE	Charged Particle Lunar Environment Experiment
CPS	Capillary Plant Support
CSAR	Coherent Synthetic Aperture Radar
CSM	Command Service Module
CSTI	Civil Space Technology Initiative Civilian Space Technology Initiative
D	Deuterium
DC	Direct Current
DCE	Drive Control Electronics
DDT&E	Design, Development, Testing, and Evaluation
DG	Directional Gyro
DKC	Design Knowledge Capture
DMS	Data Management System
DOF	Degrees of Freedom
DPS	Descent Propulsion System
ECCS	Emergency Core Cooling Systems
ECCV	Earth Crew Capture Vehicle
ECLS	Environmental Control and Life Support
ECLSS	Environmental Control and Life Support System
EDP	Embedded Data Processor
EDS	Earth Departure Stage
EDX	Energy Dispersive X-Ray
EEO	Elliptical Earth Orbit Eccentric Earth Orbit
ELM	Earth Launch Mass
ELV	Expendable Launch Vehicle
EMPA	Electron Microprobe Analysis
EMT	Electromagntic Translator
EMU	Extravehicular Mobility Unit
EOI	Earth Orbit Insertion
EPS	Electrical Power System
ESECOM	Committee on Environmental, Safety, and Economic Aspects of Magnetic Fusion Energy
ETO	Earth-to-Orbit
ETR	Engineering Test Reactor
ETS	Extraterrestrial Station
ETV	Earth Transfer Vehicle
EVA	Extravehicular Activity
FBR	Fluidized Bed Reactor
FDIR	Fault Detection, Identification, and Reconfiguration
FMEA	Failure Mode and Effects Analysis
FMECA	Failure Mode and Effects Criticality Analysis
FMR	Ferromagnetic Resonance
FTA	Fault Tree Analysis
GCTCA	Ground Control Television Cameras Assembly
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation, and Control
HDAB	Hexadecyl Trimethyl Ammonium Bromide
HI	Heading Indicator
HID	High Intensity Discharge
HLLV	Heavy Lift Launch Vehicle
HLV	Heavy Lift Vehicle
HM	Habitation Module
IF	Intermediate Frequency
IIP	Imaging Impact Probe
IMF	Initial Mass Function
IMU	Inertial Measurement Unit
IOC	Initial Operational Capabilty Initial Operating Capacity
IR	Infrared
IRAS	Infrared Astronomy Satellite
IRR	Internal Rate of Return
I_s/FeO	FMR intensity normalized to total iron content (soil maturity index)
ISA	Inertial Sensor Assembly
I_{sp}	Specific Impulse
ISY	International Space Year
IVA	Intravehicular Activity
IWWMS	Integrated Waste and Water Management System
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
KREEP	Potassium, Rare-Earth Elements, and Phosphorus
KSC	Kennedy Space Center
LACE	Lunar Atmosphere Composition Experiment
LaRC	Langley Research Center
LCRU	Lunar Communications Relay Unit
LDC	Less Developed Country
LDEF	Long Duration Exposure Facility
LEO	Low Earth Orbit
LeRC	Lewis Research Center
LGO	Lunar Geoscience Orbiter
LH_2	Liquid Hydrogen
LIPO	Lunar Imaging Polar Orbiter
LLO	Low Lunar Orbit
LLOX	Lunar Liquid Oxygen
LM	Lunar Module (also LEM)
LMDE	Lunar Module Descent Engine
LO	Lunar Orbiter Lunar Orthophotomap
LOI	Lunar Orbit Insertion
LOLA	Lunar Observer Laser Altimeter
LOP	Lunar Orbital Prospector
LOTRAN	Local Transporation Vehicle
LOX	Liquid Oxygen
LRU	Line Replacable Unit
LRV	Lunar Roving Vehicle
LSA	Level of Safety Assurance

LSE	Lunar Sounder Experiment
LT	Low Titanium
LTO	Lunar Topographic Orthophotomap
LULOX	Lunar Liquid Oxygen
LUO	Lunar Orbit
LVDT	Linear Variable Differential Transformer
LVLH	Local Vertical/Local Horizontal
MACS	Modular Attitude Control System
MCC	Mission Control Center
MDM	Multiplexer/Demultiplexer
MEB	Main Electronics Box
MERI	Moon-Earth Radio Interferometer
MFV	Moon Flight Vehicle
MHD	Magnetohydrodynamic
MLI	Multilayer Insulation
MMH	Monomethyl Hydrazine
MOI	Mars Orbit Insertion
MOSAP	Mobile Surface Applications
MPD	Magnetoplasmadynamic
MPR	Mean Payback Ratio
MPS	Maximum Permissible Limit
MSFC	Marshall Space Flight Center
MSIF	Multiple System Integration Facility
MTV	Mars Transfer Vehicle
NAS	National Academy of Sciences
NASA	National Aeronautics and Space Administration
NCOS	National Commission on Space
NEP	Nuclear-Electric Propulsion
NET	New European Torus
NI	Navigational Impactor
NIOSH	National Institute of Occupational Safety and Health
NSF	National Science Foundation
NSO	Nuclear-Safe Orbit
OAET	Office of Aeronautics, Exploration, and Technology
OAST	Office of Aeronautics and Space Technology
OMA	Operations Management Application
OMGA	Operations Management Ground Application
OMS	Operations Management System
	Orbital Maneuvering System
OMV	Orbital Maneuvering Vehicle
OPP	Oxygen Production Plant
OPWC	Oxygen Plasma Waste Conversion
OSHA	Operating and Support Hazard Analysis
	Occupational Safety and Health Administration
OTSF	Orbiting (Orbital) Transfer (Transportation) and Staging Facility
OTV	Orbital Transfer Vehicle
PAR	Photosynthetic Active Radiation
PEC	Photoelectrochemical
PHA	Preliminary Hazard Analysis
PHM	Planetary Habitation Module
PIDDP	Planetary Instrument and Definition and Development Program
PLC	Programmable Logic Controller
PLG	Prism Light Guide
PLSS	Portable Life Support System
PMAD	Power Management and Distribution
PP	Power Plant
PPF	Photosynthetic Photon Flux
PPU	Power Processing Unit
PRF	Pulse Repetition Frequency
PRV	Propellant Refill Vehicle
PSO	Passive Solar Optics
PTF	Propellant Tank Farm
PV	Photovoltaic
	Pioneer Venus
PVC	Polyvinyl Chloride
PWM	Pulse Width Modulator
PZ	Piezoelectric
R&D	Research and Development
RCS	Reaction Control System
REE	Rare-Earth Elements
RF	Radio Frequency
RFC	Regenerative Fuel Cell
RFP	Request for Proposal
RI	Range Indicator

RIG	Radioisotope Thermoelectric Generator
RLG	Ring Laser Gyro
RMP	Regolith Mining Plant
RMS	Root-Mean Square
RO	Relay Orbiter
	Reverse Osmosis
ROM	Read-Only Memory
RRS	Remote Raman Spectrometer
RSM	Radar Subsurface Mapper
RTG	Radioisotope Thermoelectric Generator
RTM	Resource Transportation Module
SAB	Spacecraft Analysis Branch
SCS	Supplemental Cooling Cart
SCUBA	Self-Contained Underwater Breathing Apparatus
SD	Single Domain
	Solar Dynamic (Generator)
SDF	System Development Facility
SDP	Standard Data Processor
SEM	Scanning Electron Microscope
SHA	System Hazard Analysis
SI	Speed Indicator
SIDE	Suprathermal Ion Detector Experiment
SLAP	Shuttle Laser Altimeter Prototype
SM	Service Module
SMRM	Solar Maximum Recovery Mission
SNR	Signal-to-Noise Ratio
SO	Solar Optics
SPF	Software Production Facility
SPS	Service Propulsion System
SPU	Signal Processing Unit
SSE	Software Support Environment
SSHA	Subsystem Hazard Analysis
SSME	Space Shuttle Main Engine
STP	Standard Temperature and Pressure
STS	Space Transportation System
SWS	Solar Wind Spectrometer
T	Tritium
TCS	Thermal Control System
TDRSS	Transmission and Data Relay Satellite System
TE	Thermoelectric
TEA	Torque Equilibrium Angle
TEI	Trans-Earth Injection
TEM	Transmission Electron Microscope
TIC	Time Interval Counter
TIMES	Thermoelectric Integrated Membrane Evaporation System
TLI	Translunar Injection
TLP	Transient Lunar Phenomenon
TMI	Trans-Mars Injection
TOC	Total Organic Carbon
TTV	Tether Tip Vehicle
TV	Television
	Thrust Vector
TVS	Thermodynamic Vent System
UF	Ultrafiltration
UV	Ultraviolet
V&V	Validation and Verification
VAT	Vehicle Assembly Tent
VAX	Virtual Address Extension
VCD	Vapor Compression Distillation
VCS	Vapor Cycle System
	Vapor-Cooled Shield
VGRF	Variable Gravity Research Facility
VHK	Very High Potassium
VHT	Very High Titanium
VIMS	Visible/Infrared Mapping Spectrometer
VIS	Visible
VLA	Very Large Array
VLBI	Very Long Baseline Interferometry
VLF	Very Low Frequency
VLFA	Very Low Frequency Array
VLT	Very Low Titanium
VMS	VAX Monitoring System
	Velocity Measurement System
VPCAR	Vapor Phase Catalytic Ammonia Removal System
WDR	Waste Disposal Rating

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